Atlas®

September 2001

Launch System Mission Planner's Guide

LOCKHEED MARTIN

Revision 9 September 2001

ATLAS LAUNCH SYSTEM MISSION PLANNER'S GUIDE

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DISCLOSURE OF DATA LEGEND
These technical data are exempt from the licensing requirements of International Traffic in Arms Regulation (ITAR) under ITAR Part 125.4(b)(13) and are cleared for public release. Exemption Pending.

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FOREWORD

This *Atlas Launch System Mission Planner's Guide* presents information on the vehicle capabilities of the Atlas launch system. A range of vehicle configurations and performance levels is offered to allow an optimum match to customer requirements at low cost. The performance data are presented in sufficient detail for preliminary assessment of the Atlas vehicle family for your missions. The guide includes essential technical and programmatic data and requirements for preliminary mission planning and preliminary spacecraft design. Interfaces are in sufficient detail to assess a first-order compatibility. A brief description of the Atlas vehicles and launch facilities is also given. The *Atlas Launch Campaign Guide* defines the facilities, operations, and hardware flow for the spacecraft and Atlas vehicle leading to encapsulation, spacecraft mate to the launch vehicle, and launch countdown procedures.

Users of this guide are encouraged to contact the offices listed below to discuss the Atlas launch vehicle family and how the Atlas family may meet user needs.

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This guide is subject to change and will be revised periodically. Revision 9 has extensive updates from the *Atlas Launch System Mission Planner's Guide*, Revision 7, and the Atlas V Addendum, Revision 8. The most significant change is bringing together the contents of Revision 7 and Revision 8 with the inclusion of the Atlas V 400 and 500 series configurations. An itemized list of updates has been documented on the revisions page. This document replaces all previous documents, including the Atlas V Addendum, Revision 8.

Change pages to this printed document will not be provided, however the version on the ILS website will be maintained. The most current version of this document can be found on the Internet at: *http://www.ilslaunch.com/missionplanner*.

ATLAS LAUNCH SYSTEM MISSION PLANNER'S GUIDE REVISIONS

Revision Date	Rev No.	Change Description	Approval
September 2001	9	Section 1—Updates Include — Added Atlas V Information, Where Appropriate — Updated Atlas IIAS/IIIA/IIIB Information and Deleted Atlas II and 11-ft PLF Information, Where Appropriate — Updated Atlas Launch Vehicle Family Figure — Updates to Atlas Launch System Discussions and Figures — Updates to Atlas Mission Design and Performance Discussions and Flight Derived Guidance Accuracy Data Table — Updates to Atlas Launch System Environments Discussions — Updates to Vehicle and Ground System Interfaces Discussions — Updates to Atlas Mission Integration and Management Discussions — Updates to Spacecraft and Launch Facilities Discussions — Revised List of Atlas Enhancements — Updates to Supplemental Information Discussions	
	9	Section 2—Updates Include — Update Atlas IIAS/IIIA/IIIB Information and Deleted Atlas II Information, Where Appropriate — Added Atlas V Information, Where Appropriate	
	9	Section 3—Updates Include — Added Atlas V Information, Where Appropriate — Updates to Thermal (Prelaunch and Flight) Discussions, Tables and Figures — Updates to Radiation and Electromagnetics (Prelaunch and Flight) Discussions, Tables and Figures — Updates to Contamination Control and Cleanliness (Prelaunch and Flight) Discussions and Figures — Updates to Spacecraft Design Loads Discussions and Tables — Updates to Acoustics Discussions and Figures — Updates to Vibration Discussions and Figures — Updates to Static Pressure Discussions and Figures	
	9	Section 4—Updates Include — Updated as Needed for Atlas V and Deleted 11 ft PLF — Moved Detailed Information and Figures to Appendices D and E, as Appropriate, for the PLFs and PLAs — Revised Spacecraft to Launch Vehicle Interfaces, Atlas PLFs, Mechanical Interface—PLAs Discussions and Figures — Updates to Electrical Interfaces Discussions and Figures — Updates to Spacecraft to Ground Equipment Interfaces Discussions and Deleted Section 4.2.5 — Updates to Range And System Safety Interfaces Discussions and Figure	
	9	Section 5—Updates Include — Updated as Needed for Atlas IIAS/IIIA/IIIB/V — Updates to Integration Management Discussions Including Launch Vehicle Management, Integration Organization, Integration Program Reviews, Mission Peculiar Design Reviews, Launch Readiness Review, and Integration Control Documentation — Added Discussions of Ground Operations Working Group and Ground Operations Readiness Review — Updates to Mission Integration Analysis Discussions — Updates to Coupled Loads Analysis Discussions — Updates to Critical Clearance Analysis Discussions — Updates to Spacecraft Postseparation Clearance Analysis Discussions — Updates to EMI/EMC Analysis Discussions — Updates to Contamination Analysis Discussions — Updates to RF Link Compatibility and Telemetry Coverage Analyses (Airborne and Ground) Discussions — Updates to Stability and Control Analysis Discussions	

Revision Date	Rev No.	Change Description	Approval
		— Updates to Injection Accuracy Analysis Discussions — Updates to Launch Window Analysis Discussions — Updates to Postflight Data Analysis Discussions — Updates to Destruct System Analysis Discussions — Updates to Mission Peculiar Flight Software Discussions — Updates to Launch Vehicle Logo Discussions — Updates to Launch Scheduling Discussions and Table and Deleted Launch Scheduling Figure — Updates to Spacecraft Launch Window Options Discussions	
	9	Section 6—Updates Include — Updates to Astrotech Facility Information Including Adding Discussion of Astrotech Building 9 and Customer LAN Capability — Updates to LC-36 Facility Information — Added Discussions of LC-41 Facility Information — Updates to SLC-3E Facility Information	
	9	Section 7—Updates Include — Updates to Launch Services Discussions — Updates to Integrated Test Plan Discussions — Updates to Launch Site Prelaunch Operations Discussions and Figures — Updates to Integrated Operations Discussions and Figures — Updates to Launch Countdown Operations Discussions and Figures — Updates to Launch Capability Discussions — Updates to Launch Postponements Discussions — Added Atlas V Operations Discussions, Where Appropriate	
	9	Section 8—Updates Include — Moved Centaur Type F Adapter and Upgrades for Payload Adapter/Payload Separation System Discussions into Standard Capabilities — Added Dual Payload Carrier Discussions into Mission Unique Enhancements — Updates to Atlas Evolutionary Enhancements, Atlas V Shock Enhancements, and Atlas V Heavy-Lift Enhancements Discussions	
		Appendices—Updates Include — Appendix A • Updated as Needed for Atlas IIAS/IIIA/IIIB • Added Atlas V • Revised Vehicle Development Discussions and Figures • Revised History for Current Flight Accomplishments • Updates to Atlas And Centaur Production And Integration Discussions and Figures • Updates to Vehicle Reliability Discussion and Figure — Appendix B • Updates to Corrective and Preventive Action Process Discussions and Figures • Updates to Software Quality Program Discussions • Updates to System Operations Discussions — Appendix C • Updates to Spacecraft Design Requirements Discussion • Updates to Spacecraft Integration Inputs Discussion — Appendix D • New Appendix for Discussion of Payload Fairings — Appendix E • New Appendix for Discussion of Payload Adapters	

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1.0 INTRODUCTION

1.1 SUMMARY

The *Atlas® Launch System Mission Planner's Guide* (AMPG) is designed to provide current and potential Atlas launch services customers with information about the Atlas launch vehicle family and related spacecraft services. The Atlas family (Fig. 1.1-1) includes the flight-proven Atlas IIAS and III versions the Atlas V series 400 and 500 versions in development. A full range of technical planning data and requirements are included to allow the user to assess the compatibility of the user's payload with various interfaces that comprise the Atlas launch vehicle system.

1.2 LAUNCH SERVICES

Atlas is offered to commercial and government launch services users through International Launch Services (ILS), a joint venture between the Lockheed Martin Corporation and the Lockheed-Khrunichev-Energia International Incorporated (LKEI) joint venture. ILS markets and manages Atlas and Proton launch services using a dedicated team of technical, management, and marketing specialists to readily define and refine the optimum and most cost-effective space transportation solutions for the launch services customer.

ILS operates as a strategic alliance between the Lockheed Martin Corporation, the manufacturer and operator of Atlas, and the Khrunichev State Research and Production Space Center, the manufacturer and operator of the Proton launch vehicle, and the Rocket Space Company Energia named for S. P. Korolev, the manufacturer of the Proton K upper stage, the Block DM. Lockheed Martin Commercial Launch Services (CLS), a constituent company of ILS, operates as the legal contracting entity for all Atlas launch services. ILS's other constituent company, LKEI, operates as the legal contracting entity for Proton launch services.

The ILS organization is aligned to offer space launch services for Atlas and/or Proton. Unique features made possible by the alliance of these two mature launch systems, can also be offered and tailored to meet customer needs and business plan (e.g., schedule assurance).

1.3 LAUNCH SERVICES ORGANIZATION AND FUNCTION

Through ILS, Lockheed Martin offers a full launch service from spacecraft integration, processing, and encapsulation, through launch operations and verification of orbit. The typical launch service includes:

1) Launch vehicle;
2) Launch operations services;

Atlas IIAS	Atlas IIIA	Atlas IIIB	Atlas V 400				Atlas V 500					
			401	411	421	431	501	511	521	531	541	551
Performance to GTO, kg (lb)												
3,719	4,037	4,500	4950	5950	6830	7640	3,970	5,270	6,285	7,200	7,980	8,670
(8,200)	(8,900)	(9,920)	(10,913)	(13,18)	(15,058)	(16,843)	(8,752)	(11,618)	(13,856)	(15,873)	(17,593)	(19,114)

Figure 1.1-1 Atlas Launch Vehicle Family

3) Mission-peculiar hardware and software design, test, and production;
4) Technical launch vehicle and spacecraft integration and interface design;
5) Mission management;
6) Program management;
7) Mission Success® events
8) Launch facilities and support provisions;
9) Payload processing facilities;
10) Spacecraft support at the launch site;
11) Validation of spacecraft separation sequence and orbit;
12) Range Safety interface.

The ILS Program Director is the primary interface with the customer, coordinates all integration activities required to support a mission and ensures that technical, contractual and licensing issues are addressed in a timely manner. With authority to proceed on a launch services contract, the Atlas program immediately assigns a single point of contact, the Program Manager that acts as both mission integrator and vehicle manager. In cases that involve national security interests of the United States, separate integration management resources of the Lockheed Martin Astronautics Operations (LMAO) organization are called on to handle unique requirements typically present with integration and launch of these types of payloads.

As part of our Atlas launch services contract support, administrative guidance and assistance can be provided, when needed, in meeting government regulations, including import and export licenses, permits, and clearances from government and political entities

1.4 ADVANTAGES OF SELECTING ATLAS

All government and commercial agreements required to conduct Atlas launch services are maintained for our customer. Agreements are in place covering payload and Atlas launch vehicle processing facilities, services, and Range support at Cape Canaveral Air Force Station (CCAFS) in Florida. Similar agreements are in work for comparable services at Vandenberg Air Force Base (VAFB) in California.

Our launch vehicles and services provide the following key advantages:

1) Flight-proven Atlas IIAS and III flight vehicle and ground system hardware and processes; Atlas V flight vehicle and ground system hardware and processes that are derived from flight-proven Atlas systems;
2) Moderate, fully validated, payload launch environments for Atlas IIAS and III (e.g., shock, vibration, acoustic, thermal) that are generally lower than those of other launch vehicles; Atlas V payload launch environments (e.g., shock, vibration, acoustic, thermal) that are the same as or better for the spacecraft than those of other launch vehicles;
3) Launch pads at CCAFS (Atlas IIAS, III and V) and VAFB (Atlas IIAS only) to accommodate low- and high-inclination satellite missions;
4) Two distinct launch complexes at CCAFS for the Atlas IIAS/III and the Atlas V to ensure launch schedules and maintain commitments;
5) A streamlined Atlas V launch processing approach to ensure launch schedules and maintain commitments;
6) An experienced team that has launched more than 75 communications satellites; an experienced organization that has launched more than 480 orbital missions in almost 40 years;
7) Mission design flexibility demonstrated in a diverse array of mission types, including projected government missions in the Evolved Expendable Launch Vehicle (EELV) National Mission Model,

low-Earth orbit missions, heavy-lift, geosynchronous orbits (GSO), most U.S. planetary missions, and numerous geostationary transfer orbit (GTO) missions;

8) A flexible mission design capability providing maximum spacecraft onorbit lifetime through optimized use of spacecraft and launch vehicle capabilities;
9) The combined resources and experience of the Atlas and Proton launch services team to meet the challenging commercial launch services needs of the future.

1.5 LAUNCH SYSTEM CAPABILITIES AND INTERFACES

From the user's perspective, the Atlas launch system is comprised of a number of hardware—and software-based subsystems and engineering, manufacturing, and operations processes designed to properly interface the spacecraft with our space transportation vehicle. The following paragraphs summarize the major interface and process components of the Atlas launch system. Each subject corresponds to an appropriate section of this document where more detailed information on the same subject can be found.

1.5.1 Atlas Launch System

The Atlas IIAS and III launch vehicle system consists of the Atlas booster, the Centaur upper stage, the payload fairing (PLF), and a payload interface (Fig. 1.5.1-1 and 1.5.1-2).

The Atlas V 400 and 500 series configurations introduces a Common Core Booster™ (CCB), up to five strap-on solid rocket boosters (SRB), a stretched Centaur upper stage (CIII) with either the single-engine Centaur (SEC) or the dual-engine Centaur (DEC), and a payload fairing (PLF). Figure 1.5.1-3 illustrates key components of the Atlas V launch vehicles. The Atlas V launch vehicle family includes the Atlas V 400 configuration that will incorporate either the standard Atlas large payload fairing (LPF) or extended payload fairing (EPF), and the Atlas V 500 configuration that will incorporate the 5-m short or 5-m medium payload fairings.

The Atlas IIAS will continue to be flown through 2003. Figure 1.5.1-4 summarizes characteristics of the Atlas IIAS.

The Atlas III is limited to a total of twelve vehicles and should be phased out by 2004/2005. Figures 1.5.1-5 through 1.5.1-7 summarize characteristics of Atlas III vehicles. The prime objective of the Atlas III was to provide the intermediate "stepping stone" between the workhorse vehicles of 1990s and the workhorse of the future, the Atlas V. The Atlas III is enabling Lockheed Martin to introduce the Atlas V with an evolutionary, low-risk approach.

The Atlas V 400 and 500 series vehicles are the latest evolutionary versions in development and will be phased into service beginning in 2002. Figure 1.5.1-8 summarizes characteristics of the Atlas V 400 series; Figure 1.5.1-9 summarizes characteristics of Atlas V 500 series. A three-digit (XYZ) naming convention was developed for the Atlas V 400 and 500 series launch vehicle system to identify its multiple configuration possibilities:

1) The first digit (X) identifies the diameter class (in meters) of the PLF (4 or 5 m),
2) The second digit (Y) indicates the number of SRBs used,
3) The third digit (Z) represents the number of Centaur engines (one or two).

For example, the designation Atlas V 521 implies an Atlas V vehicle with a 5-m PLF, two SRBs and a SEC.

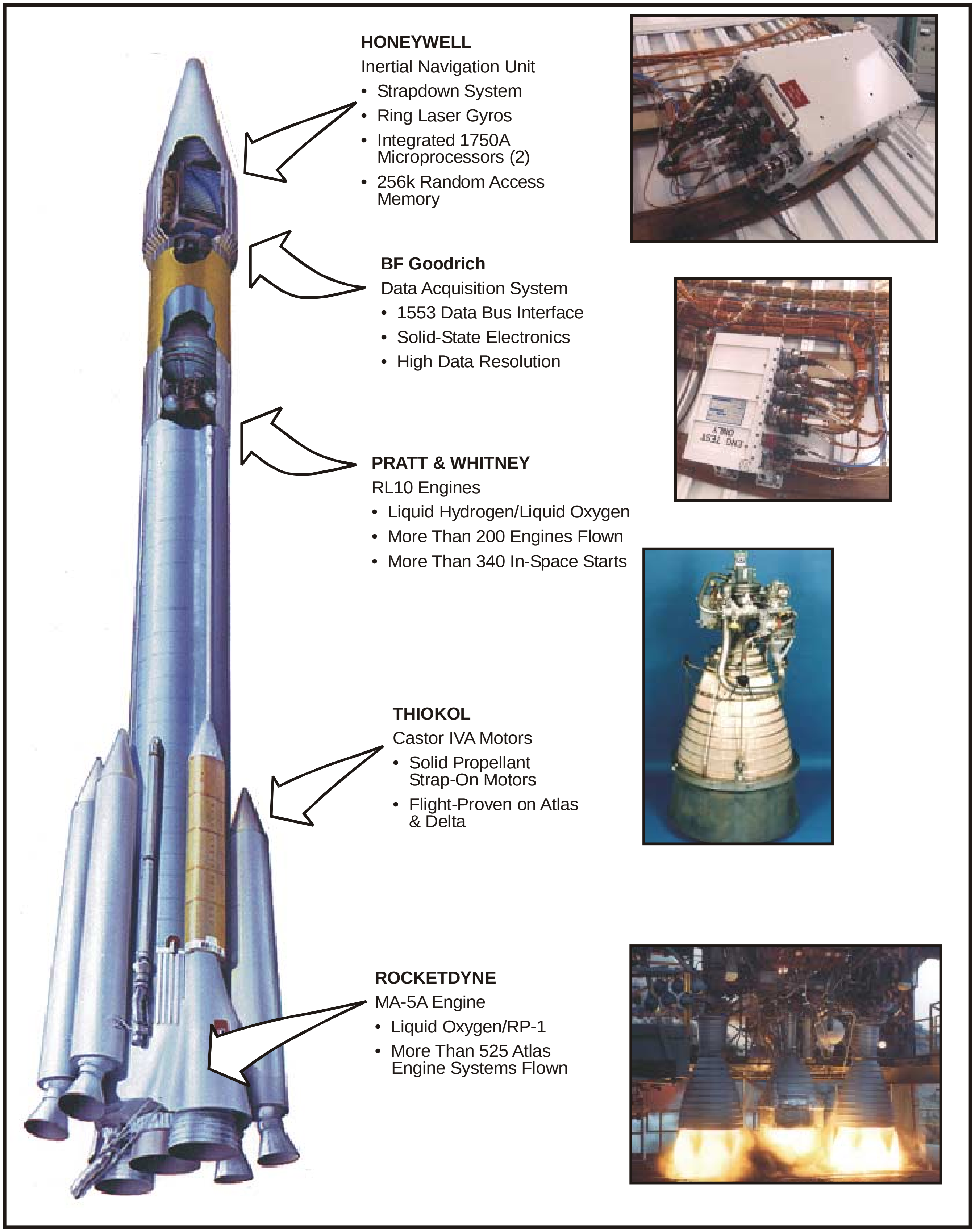


Figure 1.5.1-1 The Atlas IIAS Launch Vehicle

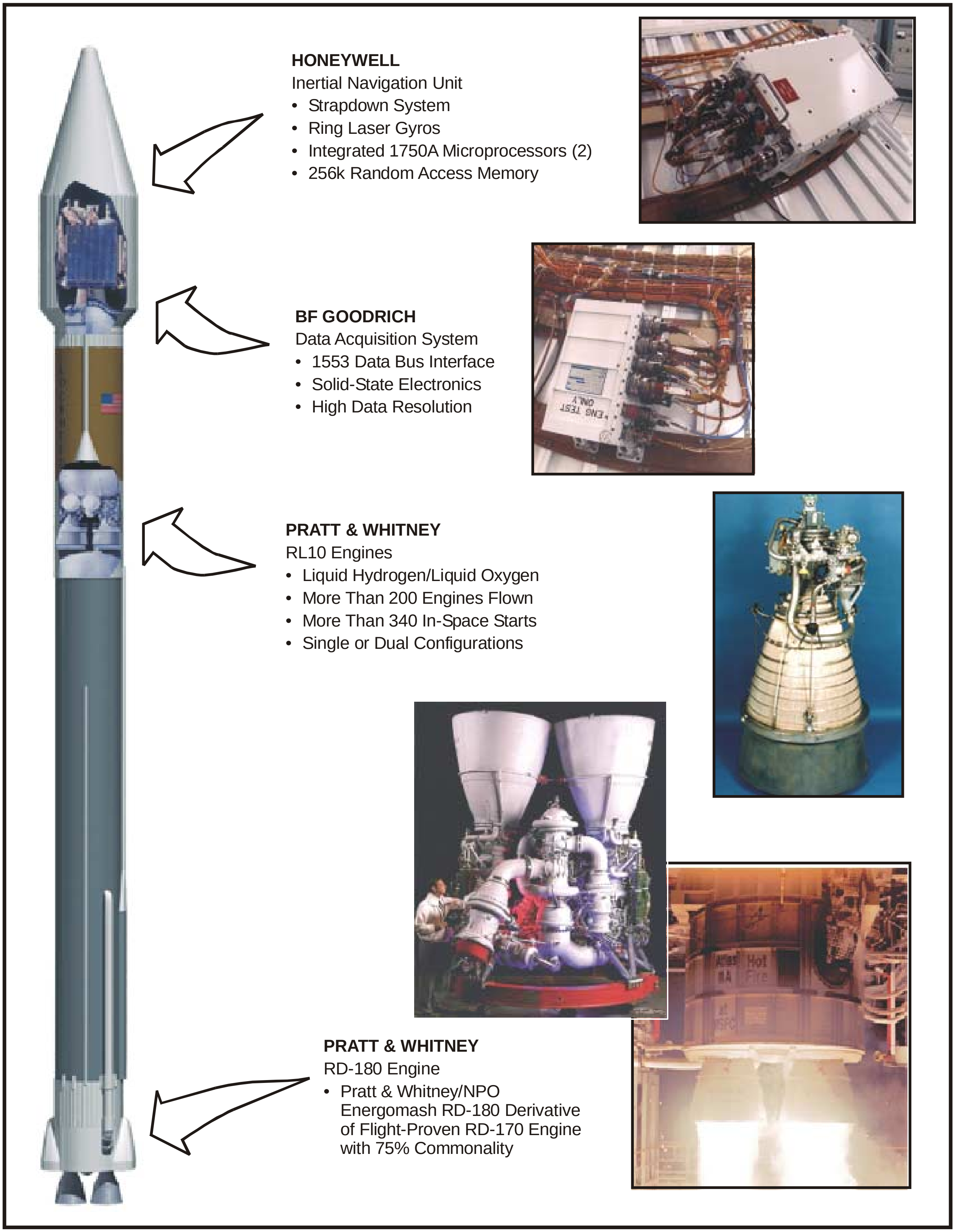


Figure 1.5.1-2 The Atlas IIIA Launch Vehicle

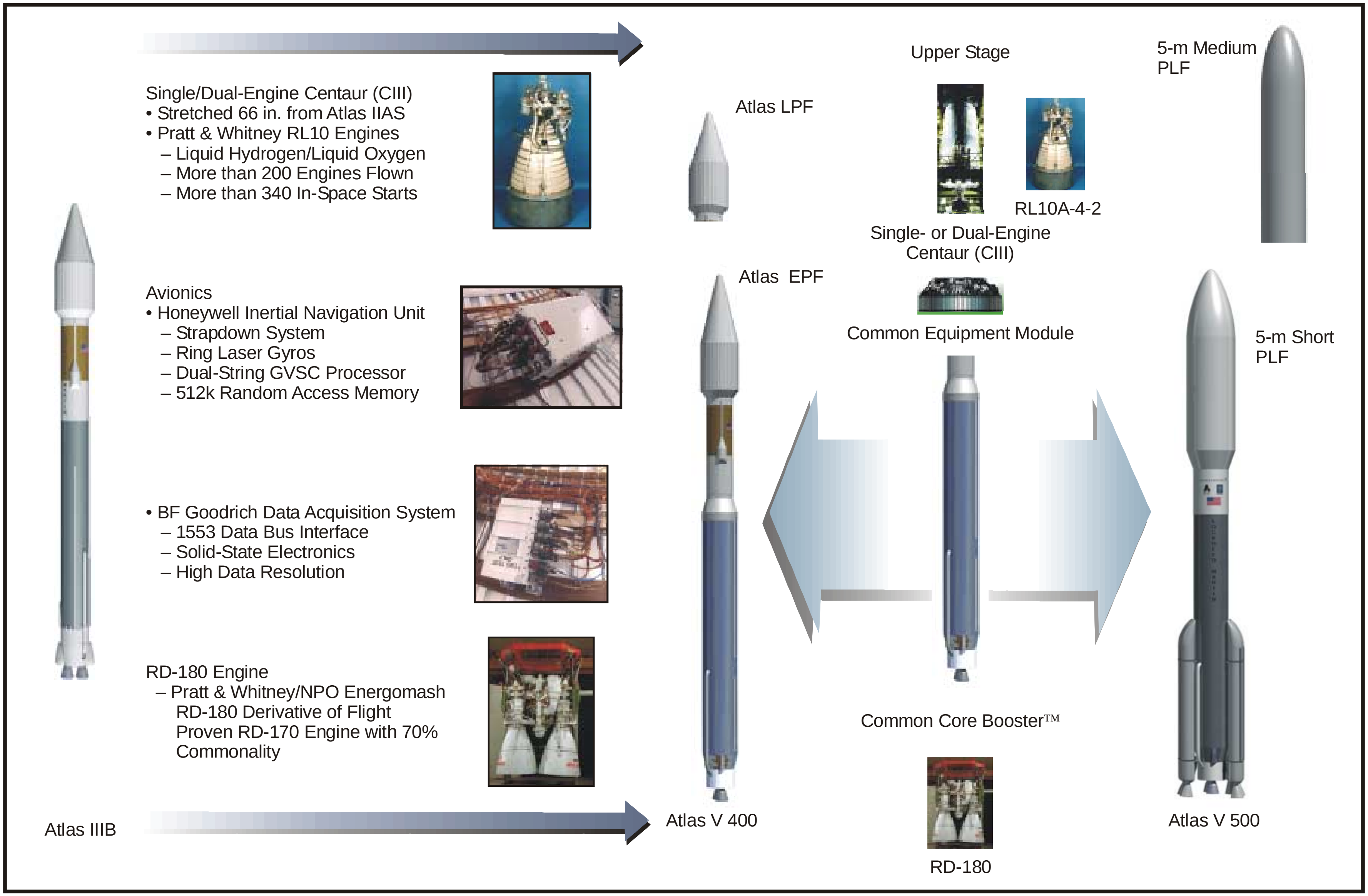
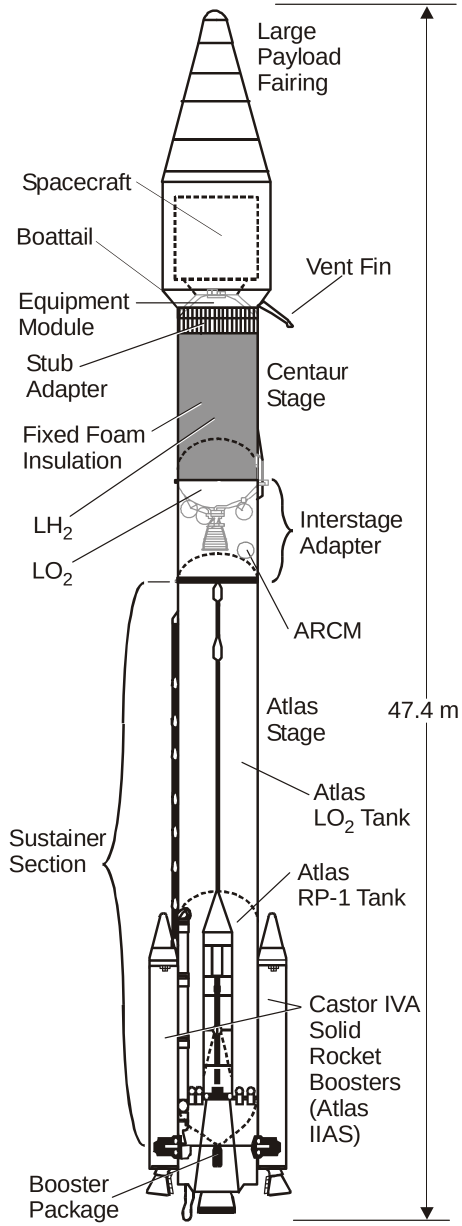


Figure 1.5.1-3 The Atlas V Launch Vehicle Family



PAYLOAD FAIRING

Features	Large (Shown)	Extended
Diameter:	4.2 m	4.2 m
Length:	12.2 m	13.1 m
Mass:	2,087 kg	2,255 kg

Subsystems	
Fairing:	Aluminum Skin Stringer & Frame Clamshell
Boattail/Split Barrel:	Aluminum Skin Stringer & Frame Clamshell
Separation:	Pyro Bolts & Spring Thrusters

CENTAUR UPPER STAGE

Features	
Size:	3.05-m Dia x 10.06-m Length
Inert Mass:	1,840 kg
Propellant:	16,930-kg LH_2 & LO_2
Guidance:	Inertial

Subsystems	
Structure:	Pressure Stabilized Stainless Steel Tanks Separated by Common Ellipsoidal Bulkhead
Propulsion:	Two Pratt & Whitney Restartable Engines
— Option:	RL10A-4-1
— Thrust:	198.4 kN
— I_{SP}:	450.5 s
	Two 51-cm Columbium Extendible Nozzles
	Twelve 27-N Hydrazine Thrusters
Pneumatics:	Helium & Hydrogen Autogenous Tank Pressurization
Hydraulics:	Provides Two Separate Systems for Gimbaling Main Engines
Avionics:	Guidance, Navigation, & Control, Vehicle Sequencing, Computer-Controlled Vent & Pressurization, Telemetry, C-Band Tracking, Range Safety Command, Electrical Power
Insulation:	Polyvinyl Chloride Foam (1.6-cm Thick), Modified Adhesive Bonding

INTERSTAGE ADAPTER (ISA)

Features	
Size:	3.05-m Dia x 3.99-m Length
Mass:	545 kg

Subsystems	
Structure:	Aluminum Skin Stringer & Frame
Separation:	Flexible Linear-Shaped Charge
Propulsion:	Atlas Roll Control Module

ATLAS STAGE

Features	
Size:	3.05-m Dia x 24.90-m Length
Propellant:	156,400-kg LO_2 & RP-1
Guidance:	From Upper Stage

Subsystems	
Structure:	Pressure-Stabilized Stainless Steel Tanks; Integrally Machined Thrust Structure
Separation:	Booster Package—10 Pneumatically Actuated Separation Latches Sustainer Section—8 Retro Rockets
Propulsion:	Rocketdyne MA-5A Booster Engine (2 Chambers) SL Thrust = 1,854 kN, I_{SP} = 262.1 s Vac Thrust = 2,065 kN, I_{SP} = 293.4 s Sustainer Engine SL Thrust = 266.0 kN, I_{SP} = 216.1 s Vac Thrust = 380.6 kN, I_{SP} = 311.0 s
Pneumatics:	Helium for Tank Pressurization, Hydraulics & Lubrication System
Hydraulics:	Provides Two Separate Systems for Main Engine Gimbal
Avionics:	Flight Control, Flight Termination, Telemetry, Computer-Controlled Atlas Pressurization System, Rate Gyro Unit

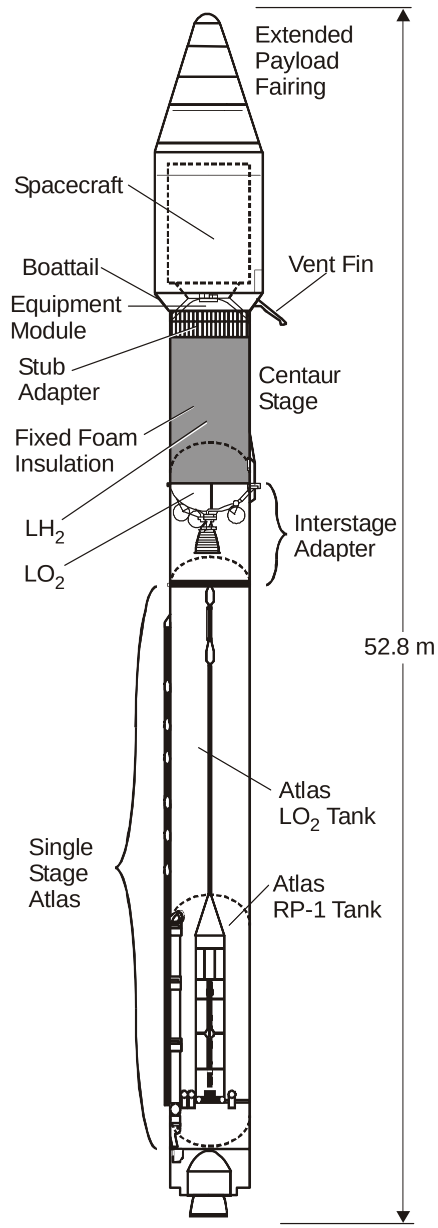
SOLID ROCKET BOOSTERS (SRB)

Four Thiokol Castor IVA

Size:	102-cm Dia x 13.6-m Length
Mass:	11,567 kg (Each Fueled)
Thrust:	433.7 kN (Each)
I_{SP}:	237.8 s

Ground-Lit Motors 11° Cant Nozzles
Air-Lit Motors 7° Cant Nozzles
Inadvertent Separation Destruct System (ISDS)

Figure 1.5.1-4 Our Atlas IIAS launch system is flight-proven and capable of meeting a wide variety of mission requirements.



PAYLOAD FAIRING

Features	Large	Extended (Shown)
Diameter:	4.2 m	4.2 m
Length:	12.2 m	13.1 m
Mass:	2,087 kg	2,255 kg

Subsystems	
Fairing:	Aluminum Skin Stringer & Frame Clamshell
Boattail/Split Barrel:	Aluminum Skin Stringer & Frame Clamshell
Separation:	Pyro Bolts & Spring Thrusters

CENTAUR UPPER STAGE

Features	
Size:	3.05-m Dia x 10.06-m Length
Inert Mass:	1,720 kg
Propellant:	16,930-kg LH_2 & LO_2
Guidance:	Inertial

Subsystems	
Structure:	Pressure Stabilized Stainless Steel Tanks Separated by Common Ellipsoidal Bulkhead
Propulsion:	One Pratt & Whitney Restartable Engine
— Type:	RL10A-4-1
— Thrust:	99.2 kN
— I_{SP}:	450.5 s
	One Electromechanically Actuated 51-cm Columbium Extendible Nozzle
	Twelve 27-N Hydrazine Thrusters
Pneumatics:	Helium & Hydrogen Autogenous Tank Pressurization
Avionics:	Guidance, Navigation, & Control, Vehicle Sequencing, Computer-Controlled Vent & Pressurization, Telemetry, C-Band Tracking, Range Safety Command, Electrical Power, Redundant Electromechanical Actuators (EMA) for Gimbal Control
Insulation:	Polyvinyl Chloride Foam (1.6-cm Thick), Modified Adhesive Bonding

INTERSTAGE ADAPTER (ISA)

Features	
Size:	3.05-m Dia x 4.45-m Length
Mass:	465 kg

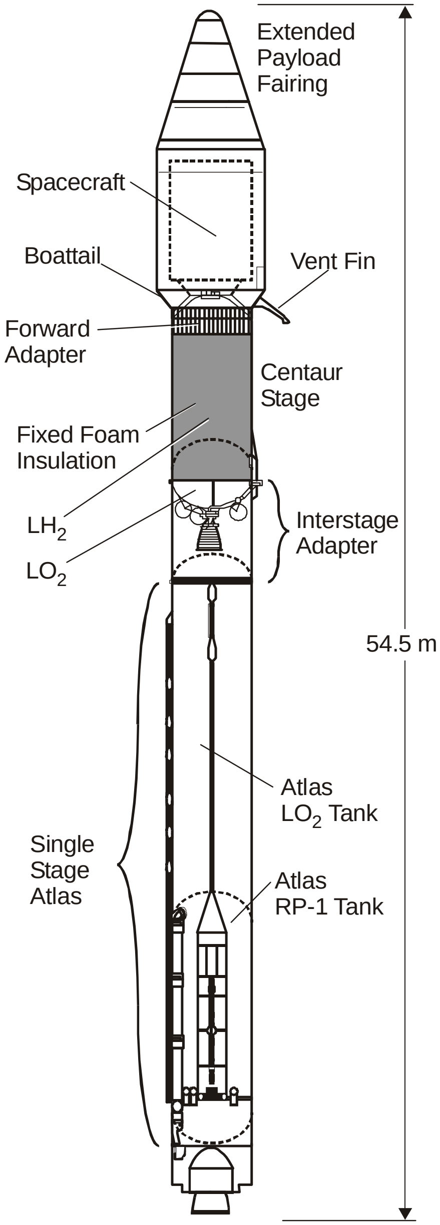
Subsystems	
Structure:	Aluminum-Lithium Skin Stringer & Frame
Separation:	Flexible Linear-Shaped Charge

ATLAS STAGE

Features	
Size:	3.05-m Dia x 28.91-m Length
Propellant:	183,200-kg LO_2 & RP-1
Guidance:	From Upper Stage

Subsystems	
Structure:	Pressure-Stabilized Stainless Steel Tanks; Integrally Machined Thrust Structure
Separation:	Single Stage Atlas—12 Retro Rockets
Propulsion:	Pratt & Whitney/NPO Energomash RD-180 Booster Engine (2 Chambers) SL 100% Thrust = 3,827 kN, I_{SP} = 311.3 s Vac 100% Thrust = 4,152 kN, I_{SP} = 337.8 s
Pneumatics:	Helium for Tank Pressurization
Hydraulics:	Integral with Engine Provides Gimbal Control
Avionics:	Flight Control, Flight Termination, Telemetry, Computer-Controlled Atlas Pressurization System, Rate Gyro Unit, Remote Control Unit

Figure 1.5.1-5 The Atlas IIIA continues the proven Atlas IIA/AS design legacy with increased performance and reduced design complexity.



PAYLOAD FAIRING

Features	Large	Extended (Shown)
Diameter:	4.2 m	4.2 m
Length:	12.2 m	13.1 m
Mass:	2,087 kg	2,255 kg

Subsystems	
Fairing:	Aluminum Skin Stringer & Frame Clamshell
Boattail/Split Barrel:	Aluminum Skin Stringer & Frame Clamshell
Separation:	Pyro Bolts & Spring Thrusters

CENTAUR UPPER STAGE

Features	
Size:	3.05-m Dia x 11.74-m Length
Inert Mass:	2,110 kg
Propellant:	20,830-kg LH_2 & LO_2
Guidance:	Inertial

Subsystems	
Structure:	Pressure Stabilized Stainless Steel Tanks Separated by Common Ellipsoidal Bulkhead
Propulsion:	Two Pratt & Whitney Restartable Engines
— Type:	RL10A-4-2
— Thrust:	198.4 kN
— I_{SP}:	450.5 s
	Two Electromechanically Actuated 51-cm Columbium Extendible Nozzles
	Eight 40-N Hydrazine Thrusters
	Four 27-N Hydrazine Thrusters
Pneumatics:	Helium & Hydrogen Autogenous (Tank Pressurization)
Hydraulics:	Provides Two Separate Systems for Gimbaling Main Engines
Avionics:	Guidance, Navigation, & Control, Vehicle Sequencing, Computer-Controlled Vent & Pressurization, Telemetry, C-Band Tracking, Range Safety Command, Electrical Power
Insulation:	Polyvinyl Chloride Foam (1.6-cm Thick), Modified Adhesive Bonding

INTERSTAGE ADAPTER (ISA)

Features	
Size:	3.05-m Dia x 4.45-m Length
Mass:	465 kg

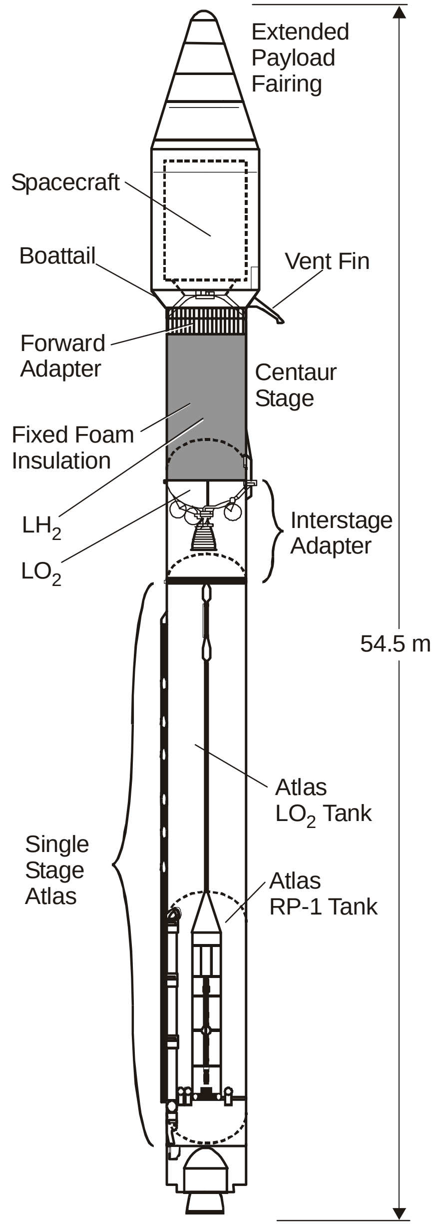
Subsystems	
Structure:	Aluminum-Lithium Skin Stringer & Frame
Separation:	Flexible Linear-Shaped Charge

ATLAS STAGE

Features	
Size:	3.05-m Dia x 28.91-m Length
Propellant:	183,200-kg LO_2 & RP-1
Guidance:	From Upper Stage

Subsystems	
Structure:	Pressure-Stabilized Stainless Steel Tanks; Integrally Machined Thrust Structure
Separation:	Single Stage Atlas—12 Retro Rockets
Propulsion:	Pratt & Whitney/NPO Energomash RD-180 Booster Engine (2 Chambers)
	SL 100% Thrust = 3,827 kN, I_{SP} = 311.3 s
	Vac 100% Thrust = 4,152 kN, I_{SP} = 337.8 s
Pneumatics:	Helium for Tank Pressurization
Hydraulics:	Integral with Engine Provides Gimbal Control
Avionics:	Flight Control, Flight Termination, Telemetry, Computer-Controlled Atlas Pressurization System, Rate Gyro Unit, Remote Control Unit

Figure 1.5.1-6 The Atlas IIIB (DEC) continues the proven Atlas IIA/AS design legacy with increased performance and reduced design complexity.



PAYLOAD FAIRING

Features	Large	Extended (Shown)
Diameter:	4.2 m	4.2 m
Length:	12.2 m	13.1 m
Mass:	2,087 kg	2,255 kg

Subsystems

Fairing:	Aluminum Skin Stringer & Frame Clamshell
Boattail/Split Barrel:	Aluminum Skin Stringer & Frame Clamshell
Separation:	Pyro Bolts & Spring Thrusters

CENTAUR UPPER STAGE

Features

Size:	3.05-m Dia x 11.74-m Length
Inert Mass:	1,930 kg
Propellant:	20,830-kg LH_2 & LO_2
Guidance:	Inertial

Subsystems

Structure:	Pressure Stabilized Stainless Steel Tanks Separated by Common Ellipsoidal Bulkhead
Propulsion:	One Pratt & Whitney Restartable Engine
— Type:	RL10A-4-2
— Thrust:	99.2 kN
— I_{SP}:	450.5 s
	One Electromechanically Actuated 51-cm Columbium Extendible Nozzle
	Eight 40-N Thrusters
	Four 27-N Thrusters
Pneumatics:	Helium & Hydrogen Autogenous Tank Pressurization
Avionics:	Guidance, Navigation, & Control, Vehicle Sequencing, Computer-Controlled Vent & Pressurization, Telemetry, C-Band Tracking, Range Safety Command, Electrical Power, Redundant Electromechanical Actuators (EMA) for Gimbal Control
Insulation:	Polyvinyl Chloride Foam (1.6-cm Thick), Modified Adhesive Bonding

INTERSTAGE ADAPTER (ISA)

Features

Size:	3.05-m Dia x 4.45-m Length
Mass:	465 kg

Subsystems

Structure:	Aluminum-Lithium Skin Stringer & Frame
Separation:	Flexible Linear-Shaped Charge

ATLAS STAGE

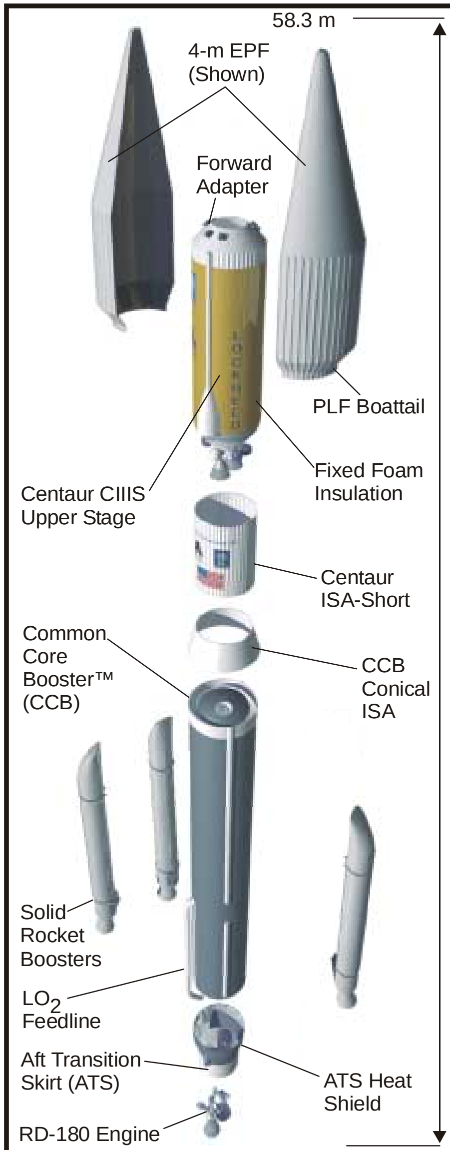
Features

Size:	3.05-m Dia x 28.91-m Length
Propellant:	183,200-kg LO_2 & RP-1
Guidance:	From Upper Stage

Subsystems

Structure:	Pressure-Stabilized Stainless Steel Tanks; Integrally Machined Thrust Structure
Separation:	Single Stage Atlas—12 Retro Rockets
Propulsion:	Pratt & Whitney/NPO Energomash RD-180 Booster Engine (2 Chambers) SL 100% Thrust = 3,827 kN, I_{SP} = 311.3 s Vac 100% Thrust = 4,152 kN, I_{SP} = 337.8 s
Pneumatics:	Helium for Tank Pressurization
Hydraulics:	Integral with Engine Provides Gimbal Control
Avionics:	Flight Control, Flight Termination, Telemetry, Computer-Controlled Atlas Pressurization System, Rate Gyro Unit, Remote Control Unit

Figure 1.5.1-7 The Atlas IIIB (SEC) continues the proven Atlas IIA/AS design legacy with increased performance and reduced design complexity.



PAYLOAD FAIRING (PLF)

Features	Large	Extended
Diameter:	4.2 m	4.2 m
Length:	12.2 m	13.1 m
Mass:	2,087 kg	2,255 kg

Subsystems
- Fairing: Aluminum Skin Stringer & Frame Clamshell
- Boattail: Aluminum Skin Stringer & Frame Clamshell
- Separation: Pyro Bolts & Spring Thrusters

CENTAUR UPPER STAGE(CIII)

Features
- Size: 3.05-m Dia x 12.68-m Length with Extended Nozzle
- Inert Mass: 1,914 kg/2,106 kg
- Propellant: 20,830-kg LH_2 & LO_2
- Guidance: Inertial

Subsystems
- Structure: Pressure Stabilized Stainless Steel Tanks Separated by Common Ellipsoidal Bulkhead
- Propulsion: One or Two Pratt & Whitney Restartable Engine(s)
  - — Model: RL10A-4-2
  - — Thrust: 99.2 kN (SEC) 198.4 kN (DEC)
  - — I_{SP}: 450.5 s
  - (SEC) One Electromechanically Actuated 51-cm Columbium Fixed Nozzle
    Four 27-N Hydrazine Thrusters
    Eight 40-N Lateral Hydrazine Thrusters
  - (DEC) Two Hydraulically Actuated 51-cm Columbium Extendible Nozzles
    Eight 40-N Hydrazine Thrusters
    Four 27-N Hydrazine Thrusters
- Pneumatics: Helium & Hydrogen Autogenous Tank Pressurization
- Avionics: Guidance, Navigation & Control, Vehicle Sequencing, Computer-Controlled Vent & Pressurization, Telemetry, Tracking, Range Safety Command, Electrical Power
- Insulation: Polyvinyl Chloride Foam (1.6-cm Thick), Modified Adhesive Bonding

SOLID ROCKET BOOSTERS (SRB)

Zero-to-Three	Ground-Lit
Size:	155-cm Dia x 19.5-m Length
Mass:	46,494 kg (Each Fueled)
Thrust:	1,361 kN (Each)
I_{SP}:	275 s
Nozzle Cant:	3°

CENTAUR INTERSTAGE ADAPTER (CISA SHORT)

Features
- Size: 3.05-m Dia x 3.13-m Length
- Mass: 374 kg

Subsystems
- Structure: Aluminum Lithium Skin Stringer & Frame
- Separation: Low-Cost Atlas Separation System

CCB CONICAL INTERSTAGE ADAPTER

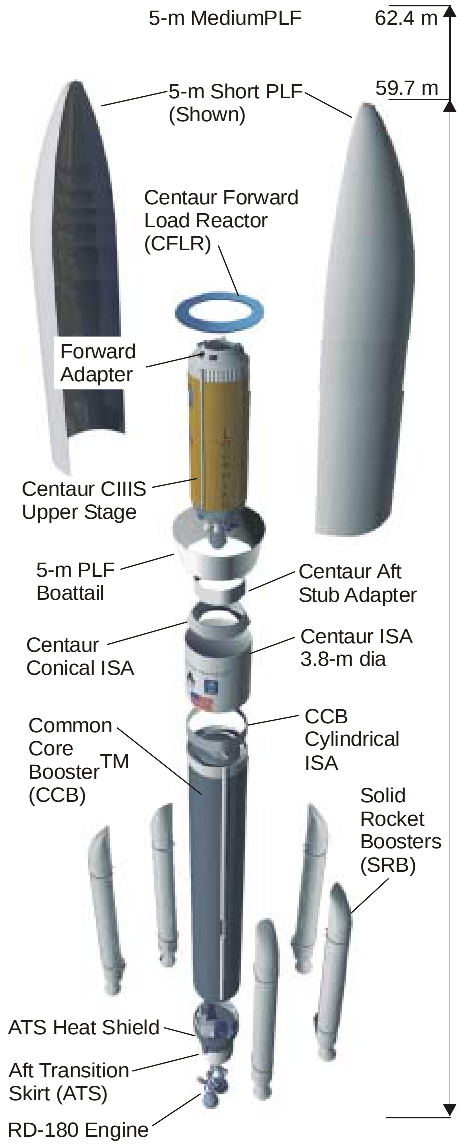
Features
- Size: 3.05-m Dia (Top) x 1.65-m Length
  3.83-m Dia (Bottom)
- Mass: 420 kg
- Structure: Composite (Graphite Epoxy) with Aluminum Ring Frames Fwd & Aft

COMMON CORE BOOSTER™ (CCB)

Features
- Size: 3.81-m Dia x 32.46-m Length
- Inert Mass: 20,743 kg
- Propellant: 284,089-kg LO_2 & RP-1
- Guidance: From Upper Stage

Subsystems
- Structure: Structurally Stable Aluminum Isogrid Tanks; Integrally Machined Aft Transition Structure; Composite Heat Shield
- Separation: 8 Retro Rockets
- Propulsion: Pratt & Whitney/NPO Energomash RD-180 Booster Engine (2 Chambers)
  SL 100% Thrust = 3,827 kN, I_{SP} = 311.3 s
  Vac 100% Thrust = 4,152 kN, I_{SP} = 337.8 s
- Pneumatics: Helium for Tank Pressurization, Computer-Controlled Pressurization System
- Hydraulics: Fluid—Integral with Engine Provides Gimbal Control
- Avionics: Flight Control, Flight Termination, Telemetry, Redundant Rate Gyros, Electrical Power

Figure 1.5.1-8 The Atlas V 400 launch system is capable of meeting a wide variety of mission requirements.



PAYLOAD FAIRING (PLF)

Features	**5-m Short (Std)**	**5-m Medium (Non-Std)**
Diameter:	5.4 m	5.4 m
Length:	20.7 m	23.4 m
Mass:	4,085 kg	4,649 kg
Subsystems		
Fairing:	Bisector; Sandwich Construction with Graphite Epoxy Face Sheets & an Aluminum Honeycomb Core	
Boattail:	Fixed, Composite Sandwich Const	
Separation:	Vertical Separation by a Linear Piston & Cylinder Activated by a Pyrotechnic Cord; Horizontal Separation by an Expanding Tube Shearing a Notched Frame, Activated by a Pyrotechnic Cord	

CENTAUR UPPER STAGE (CIII)

Features	**All Common with Atlas 400 Series**
Size:	3.05-m Dia x 12.68-m Length with Extended Nozzle
Inert Mass:	1,914-kg/2,106-kg
Propellant:	20,830-kg LH_2 & LO_2
Guidance:	Inertial
Subsystems	
Structure:	Pressure Stabilized Stainless Steel Tanks Separated by Common Ellipsoidal Bulkhead
Propulsion:	One or Two Pratt & Whitney Restartable Engine(s)
— Model:	RL10A-4-2
— Thrust:	99.2 kN (SEC) 198.4 kN (DEC)
— I_{SP}:	450.5 s
(SEC)	One Electromechanically Actuated 51-cm Columbium Fixed Nozzle Four 27-N Hydrazine Thrusters Eight 40-N Lateral Hydrazine Thrusters
(DEC)	Two Hydraulically Actuated 51-cm Columbium Extendible Nozzles Eight 40-N Lateral Hydrazine Thrusters Four 27-N Hydrazine Thrusters
Pneumatics:	Common with Atlas V 400 Series
Avionics:	Common with Atlas V 400 Series
Insulation:	Common with Atlas V 400 Series

SOLID ROCKET BOOSTERS (SRB)

Zero-to-Five	Ground-Lit
Size:	155-cm Dia x 19.5-m Length
Mass:	46,494 kg (Each Fueled)
Thrust:	1,361 kN (Each)
I_{SP}:	275 s
Nozzle Cant:	3°

CENTAUR INTERSTAGE ADAPTER (C-ISA LARGE)

Features	
Size:	3.83-m Dia x 3.81-m Length
Mass:	1,266 kg
Subsystems	
Structure:	Composite Sandwich (Aluminum Core/Graphite Epoxy Face Sheets)

CCB CYLINDRICAL INTERSTAGE ADAPTER

Features	
Size:	3.83-m Dia x 0.32-m Length
Mass:	278 kg
Subsystems	
Structure:	Aluminum Machined Rolled-Ring Forging

COMMON CORE BOOSTER™ (CCB)

Features	**All Common with Atlas V 400 Series**
Size:	3.81-m Dia x 32.46-m Length
Propellant:	284,089-kg LO_2 & RP-1
Inert Mass:	21,173 kg for 55Z Configuration
Guidance:	From Upper Stage
Subsystems	
Structure:	Structurally Stable Aluminum Isogrid Tanks; Integrally Machined Aft Transition Structure; Composite Heat Shield
Separation:	8 Retro Rockets
Propulsion:	Pratt & Whitney/NPO Energomash RD-180 Booster Engine (2 Chambers) SL 100% Thrust = 3,827 kN, I_{SP} = 311.3 s Vac 100% Thrust = 4,152 kN, I_{SP} = 337.8 s
Pneumatics:	Helium for Tank Pressurization, Computer-Controlled Pressurization System
Hydraulics:	Integral with Engine Provides Gimbal Control
Avionics:	Flight Control, Flight Termination, Telemetry, Redundant Rate Gyros, Electrical Power

Figure 1.5.1-9 The Atlas V 500 launch system is capable of meeting a wide range of mission requirements.

1.5.2 Atlas Mission Design and Performance

On more than 100 past missions, Atlas has demonstrated its capability to deliver spacecraft of various volumes and masses to precisely targeted orbits. Because the Atlas vehicle is primarily designed for dedicated, single payload missions, a number of Atlas-unique, flight-proven mission trajectory and targeting enhancement options are offered to maximize the benefits of the Atlas system to the spacecraft mission. As demonstrated by the flight-derived orbital injection accuracy data in Tables 1.5.2-1 and 1.5.2-2, Atlas provides exceptional orbit placement capabilities. Table 1.5.2-1 provides orbital insertion accuracy for Guidance Commanded Shutdown (GCS) missions. Since propellant margin is reserved to accommodate launch vehicle dispersions and ensure a guidance commanded shutdown, the flight results shown in Table 1.5.2-1 reflect the precision of the Atlas guidance and navigation hardware and flight software algorithms. For missions that incorporate Atlas mission targeting options of Inflight Retargeting (IFR), Minimum Residual Shutdown (MRS), or combination IFR/MRS, injection accuracy is provided in Table 1.5.2-2 in terms of the difference between the predicted and achieved ideal velocity required to inject from the transfer orbit to a geostationary orbit (GSO). This method provides a common comparison of flight results for the wide variety of Atlas mission optimization and targeting options. In addition, the delta-velocity to GSO that results from the achieved transfer orbit not only reflects guidance system performance, but also includes actual launch vehicle performance. As shown in Table 1.5.2-2, the near-zero statistical mean of the flight results demonstrates the accuracy of Atlas launch vehicle performance modeling and preflight performance predictions.

Section 2.0 discusses the launch vehicle mission and performance capabilities available for Atlas IIAS, III, and Atlas V 400 and 500 series missions. For certain missions, additional performance is available using a dual-engine configuration. A DEC flies on Atlas IIAS vehicles, and is manifested on Atlas III vehicles starting in 2001. The DEC performance increase is primarily from the additional thrust of the second engine, which is especially beneficial to large spacecraft flying to low-Earth orbits. The Atlas V 500 series vehicles also provide the capability for three burn Centaur missions, which allow direct insertion into geosynchronous or geostationary orbits (GSO) and significant performance improvement to geosynchronous transfer orbits (GTO). The GSO capability allows spacecraft to avoid the purchase and integration of an apogee kick motor stage. Lockheed Martin has extensive experience flying GSO missions with the Titan IV Centaur upper stage.

Section 8 discusses enhancements to the Atlas launch system currently in their early phase of implementation. In particular, ILS and Lockheed Martin will be introducing and offering a Dual Payload Carrier (DPC) and Auxiliary Payload accommodation for shared launches.

Table 1.5.2-1 Injection Accuracy Results for GCS Missions

Vehicle	Satellite	Launch Date	Achieved –Target Apogee	3σ Apogee	Achieved Apogee Accuracy (σ)	Achieved –Target Perigee	3σ Perigee	Achieved Perigee Accuracy (σ)	Achieved –Target Inclination	3σ Inclination	Achieved Inclination Accuracy (σ)	Achieved–Target Argument of Perigee	3σ Argument of Perigee	Achieved Argument of Perigee Accuracy (σ)
AC-101	MLV 1	10-Feb-92	-12.16	41.00	**-0.89**	-0.22	1.30	**-0.51**	0.003	0.020	**0.45**	0.020	0.180	**0.33**
AC-72	Galaxy V	13-Mar-92	-0.01	49.00	**0.00**	0.03	1.10	**0.08**	0.010	0.030	**1.00**	-0.030	0.210	**-0.43**
AC-105	Intelsat K	09-Jun-92	-6.02	58.18	**-0.31**	0.05	1.05	**0.14**	-0.002	0.013	**-0.46**	0.001	0.128	**0.02**
AC-103	MLV 2	02-Jul-92	-20.58	41.00	**-1.51**	-0.10	1.30	**-0.23**	0.001	0.020	**0.15**	-0.020	0.180	**-0.33**
AC-104	MLV 3	19-Jul-93	-8.04	41.00	**-0.59**	0.02	1.30	**0.05**	-0.001	0.020	**-0.15**	0.015	0.180	**0.25**
AC-106	MLV 4	28-Nov-93	-14.71	41.00	**-1.08**	-0.17	1.30	**-0.39**	-0.001	0.020	**-0.15**	0.017	0.180	**0.28**
AC-77	GOES J	23-May-95	-33.40	58.00	**-1.73**	0.22	1.10	**0.60**	-0.007	-0.016	**1.31**	-0.005	0.389	**-0.04**
AC-118	MLV 5	31-Jul-95	12.43	41.00	**0.91**	0.07	1.30	**0.16**	-0.006	0.020	**-0.90**	0.010	0.180	**0.17**
AC-122	Inmarsat 3	03-Apr-96	0.54	68.50	**0.02**	0.07	1.20	**0.18**	0.000	0.043	**0.00**	0.000	0.279	**0.00**
AC-78	SAX	30-Apr-96	0.90	2.50	**1.08**	0.00	3.60	**0.00**	0.007	0.019	**1.11**	NA	NA	**NA**
AC-129	Inmarsat 3	17-Dec-96	10.60	68.50	**0.46**	0.13	1.20	**0.33**	-0.001	0.043	**-0.07**	0.010	0.279	**0.11**
AC-79	GOES K	25-Apr-97	11.00	57.70	**0.57**	0.50	1.00	**1.50**	-0.011	0.016	**-2.06**	0.030	0.250	**0.36**
AC-131	MLV 6	24-Oct-97	8.10	41.00	**0.59**	0.18	1.30	**0.42**	0.000	0.020	**0.00**	0.010	0.180	**0.17**
AC-151	Intelsat 806	27-Feb-98	-18.90	56.20	**-1.01**	0.49	1.00	**1.47**	0.005	0.010	**1.50**	0.015	0.315	**0.14**
AC-153	Intelsat 805	18-Jun-98	-31.10	56.20	**-1.66**	-0.11	1.00	**-0.33**	0.004	0.010	**1.20**	0.020	0.315	**0.19**
AC-141	EOS Terra	18-Dec-99	-0.59	3.80	**-0.47**	0.02	1.35	**0.04**	-0.007	0.100	**-0.21**	NA	NA	**NA**
AC-138	MLV 8	20-Jan-00	0.50	41.00	**0.04**	0.03	1.30	**0.07**	0.001	0.020	**0.15**	-0.002	0.180	**-0.03**
AC-137	GOES-L	03-May-00	28.12	80.53	**1.05**	0.25	0.94	**0.80**	-0.006	0.035	**-0.51**	0.018	0.236	**0.23**
AC-140	MLV 9	19-Oct-00	-0.05	41.00	**0.00**	0.22	1.30	**0.51**	0.003	0.020	**0.45**	-0.005	0.180	**-0.08**
AC-157	MLV 11	05-Dec-00	11.33	58.6	**0.58**	0.04	0.97	**0.12**	-0.002	0.011	**-0.56**	0.000	0.191	**0.00**

Table 1.5.2-2 Injection Accuracy Results for IFR, MRS, and IFR/MRS Missions

Vehicle	Mission	ILC	Type	Predicted V to GSO (m/s)	Achieved V to GSO (m/s)	Delta (m/s)
AC-102	EUTELSAT II F3	7-Dec-91	IFR	1,542.4	1,539.3	-3.1
AC-75	UHF F2	3-Sept-93	MRS	2,349.0	2,409.0	60.0
AC-108	Telstar 4	15-Dec-93	IFR	1,774.4	1,740.2	-34.2
AC-73	GOES-I (8)	13-Apr-94	MRS	1,732.3	1,740.2	7.9
AC-76	UHF F3	24-June-94	MRS	2,325.3	2,375.8	50.5
AC-107	DirecTV / DBS-2	3-Aug-94	MRS	1,776.0	1,765.8	-10.2
AC-111	INTELSAT 703	6-Oct-94	MRS	1,802.2	1,764.8	-37.4
AC-110	Orion Atlantic F1	29-Nov-94	IFR	1,472.4	1,472.2	-0.2
AC-113	INTELSAT 704	10-Jan-95	MRS	1,758.2	1,740.8	-17.4
AC-112	EHF F4	28-Jan-95	MRS	1,961.6	1,959.0	-2.6
AC-115	INTELSAT 705	22-Mar-95	MRS	1,769.7	1,748.6	-21.1
AC-114	AMSC-1 / MSAT	7-Apr-95	MRS	1,770.4	1,748.4	-22.0
AC-116	EHF F5	31-May-95	MRS	1,969.6	1,983.6	14.0
AC-117	JCSAT 3	28-Aug-95	IFR/MRS	1,525.9	1,531.9	6.0
AC-119	EHF F6	22-Oct-95	MRS	1,955.6	1,958.2	2.6
AC-120	Galaxy-IIIR	14-Dec-95	MRS	1,835.2	1,828.2	-7.0
AC-126	Palapa-C1	31-Jan-96	IFR/MRS	1,507.0	1,499.4	-7.6
AC-125	EHF F7	25-July-96	MRS	1,962.9	1,966.3	3.4
AC-123	GE-1	8-Sept-96	MRS	1,638.2	1,628.9	-9.3
AC-124	Hot Bird 2	21-Nov-96	IFR	1,730.2	1,740.0	9.8
AC-127	JCSAT 4	16-Feb-97	IFR/MRS	1,511.3	1,501.4	-9.9
AC-128	TEMPO	8-Mar-97	MRS	2,132.6	2,121.7	-10.9
AC-133	Superbird-C	27-July-97	IFR/MRS	1,543.1	1,529.1	-14.0
AC-146	GE-3	4-Sept-97	IFR	1,600.6	1,599.7	-0.9
AC-135	EchoStar III	5-Oct-97	IFR/MRS	1,756.9	1,752.7	-4.1
AC-149	Galaxy VIII-i	8-Dec-97	MRS	1,712.4	1,685.7	-26.7
AC-132	UHF F/O F8	16-Mar-98	MRS	2,081.9	2,057.9	-24.0
AC-134	HOT BIRD 5	9-Oct-98	IFR	1,754.7	1,763.2	8.5
AC-130	UHF F/O F9	20-Oct-98	MRS	1,992.9	2,003.4	10.5
AC-152	JCSAT 6	15-Feb-99	IFR/MRS	1,483.7	1,492.8	9.1
AC-154	Eutelsat W3	12-Apr-99	IFR/MRS	1,607.3	1,605.6	-1.7
AC-155	EchoStar V	23-Sept-99	MRS	1,714.2	1,711.2	-3.0
AC-136	UHF F/O F10	23-Nov-99	MRS	2,000.5	1,992.1	-8.4
AC-158	Hispasat-1C	3-Feb-00	IFR	1,595.4	1,593.5	-1.9
AC-201	Eutelsat W4	24-May-00	IFR	1,585.5	1,607.6	22.1
AC-139	TDRS-H	30-June-00	MRS	1,908.0	1,957.5	49.5
AC-161	EchoStar VI	14-July-00	MRS	1,762.8	1,770.2	7.4
					Mean	**-0.4**
					Standard Deviation	**20.9**

1.5.3 Atlas Launch System Environments

The Atlas IIAS/III/V launch system provides spacecraft preflight and flight environments that are typically more benign than those available with other launch systems. All environments specified for the Atlas IIAS and III launch system (e.g., shock, vibration, acoustic, thermal, electromagnetic) are based on engineering analyses and have been fully validated with test and flight telemetry data. Verification that the customer's flight environments remained within specified levels can be obtained with use of additional instrumentation with an optional digital telepak. This hardware enables telemetering of high-frequency measurements concerning the spacecraft interface and environment. These supplemental measurements, when added to our standard low-frequency flight telemetry, can allow verification of interface control document (ICD) flight environments for each Atlas mission (mission satisfaction option).

All environments specified for the Atlas V launch system (e.g., shock, vibration, acoustic, thermal, electromagnetic) are based on engineering analyses of existing and evolved hardware, and are in the process of being validated by test. Flight telemetry data from Atlas V vehicle configurations will be used to update these environments, as required. As with the Atlas IIAS/III family, verification that the customer's flight environments remained within specified levels can be obtained with use of additional instrumentation that will measure spacecraft environments.

The environments to which spacecraft are exposed are fully discussed in Section 3.0.

1.5.4 Vehicle and Ground System Interfaces

The Atlas launch system offers a broad range of launch vehicle and ground processing hardware and facility options to meet spacecraft requirements. Primary interfaces between the launch vehicle and spacecraft consist of the payload adapter and payload fairing. The payload adapter supports the spacecraft on top of the launch vehicle and provides mounting provisions for the payload separation system, spacecraft-to-launch vehicle electrical interfaces, spacecraft purge system connections, and environmental measurement instrumentation systems, as required. The Atlas program has a variety of standard payload adapters, all compatible with all Atlas configurations, which have been designed to meet identified spacecraft interface requirements. Additional payload adapters can be developed to meet mission-specific needs of future spacecraft. In addition, the user has the option to provide the payload adapter and separation system. Atlas payload adapter systems are described in Section 4.1.2 and Appendix E.

The payload fairing encloses and protects the spacecraft during ground operations and launch vehicle ascent. The payload fairing also incorporates hardware to control thermal, acoustic, electromagnetic, and cleanliness environments for the spacecraft and may be tailored to provide access and radio frequency (RF) communications to the encapsulated spacecraft. The Atlas program offers two 4-m diameter payload fairings configurations, the LPF and the EPF, that are available for Atlas IIAS, III, and Atlas V 400 missions. A 5-m diameter payload fairing is available for Atlas V 500 missions in a short (20.7-m/68-ft) or medium (23.4-m/77-ft) length configuration. Atlas payload fairing systems are described in Section 4.1.1 and Appendix D.

The Atlas program also provides facilities, hardware, and services necessary to support the spacecraft during ground operations and processing. These items are discussed in Section 4.2.

1.5.5 Atlas Mission Integration and Management

The Atlas mission integration and management process is designed to effectively use engineering and production talents of Lockheed Martin Astronautics Operations and spacecraft contractor organizations to integrate the customer's spacecraft onto the Atlas launch vehicle. Section 5.0 is an overview of the mission integration process and launch services management functions in place for commercial and government missions. For typical communications spacecraft missions, the 12-month integration schedule is discussed. Our management approach and a summary of integration analysis tasks are provided to enable the customer to fully understand the mission integration process.

1.5.6 Spacecraft and Launch Facilities

Upon arrival at the launch site, most spacecraft require the use of payload and/or hazardous processing facilities for fueling and final checkout of onboard systems before launch. Section 6.0 summarizes facilities available for final spacecraft processing. In addition, operational capabilities and interfaces of the Atlas launch complexes in operation at CCAFS in Florida and at VAFB in California are defined.

1.5.7 Atlas Launch Operations

Atlas launch operations processes require involvement of the launch services customer and spacecraft contractor. Section 7.0 provides an overview of our operations processes, discussing issues that the potential customer may wish to consider early in the mission integration process.

1.5.8 Atlas Enhancements

Section 8.0 is designed to provide insight to the Atlas customer community of Lockheed Martin's plans for enhancing the Atlas vehicle to meet launch services needs of the 21st Century. Enhancements include:

1) Spacecraft dispensers and universal spacecraft separation node,
2) Dual Payload Carrier (DPC)
3) Auxiliary Payload Service
4) Atlas V 441 launch vehicle development and performance data,
5) Heavy Lift Vehicle (HLV) development and performance data.

Contact information for additional information requests can be found in the Foreword.

1.5.9 Supplemental Information

Five appendices are provided in this document to address various items in more detail:

1) Appendix A discusses the history of the Atlas and Atlas/Centaur launch vehicle. A more detailed description of Atlas and Centaur stages and subsystems is provided. The reliability growth method for evaluating mission and vehicle reliability also is summarized.
2) Appendix B details our mission success philosophy and quality assurance process at our facilities and at our subcontractors and suppliers.
3) Appendix C defines spacecraft technical data requirements to support the mission integration process. In addition, a discussion of the type and format of technical data required by the Atlas program is summarized to provide insight into the exchange of information between the spacecraft contractor, launch services customer, and ourselves that occurs during typical mission integration.
4) Appendix D describes the Atlas payload fairings.
5) Appendix E describes the Atlas payload adapters.

1.6 CONCLUSION

Members of the Atlas team are eager to assist in definition and development of potential future Atlas missions. Potential launch services customers may refer to the Foreword of this document for information regarding the appropriate ILS representative to contact for their mission needs.

2.0 ATLAS MISSION DESIGN AND PERFORMANCE

Over the past three decades, Atlas and Centaur stages have flown together as the Atlas/Centaur and with other stages (e.g., Atlas/Agena and Titan/Centaur) to deliver commercial, military, and scientific payloads to their target orbits. Based on our experience with more than 550 Atlas launches, performance for each launch vehicle is determined by engineering analysis of developed and new hardware, emphasizing conservative performance prediction to ensure each vehicle meets design expectations. As all Atlas IIAS and Atlas IIIA configurations are flight-proven, engineering estimates of Atlas family performance capabilities reflect flight-qualified hardware performance characteristics and our improved knowledge of developed hardware. Lockheed Martin has significantly increased the performance levels available to the Atlas launch services customer with the introduction of the new Atlas V configurations that are evolved from the same flight-proven components. Table 2-1 illustrates the performance capability of the Atlas IIAS; Table 2-2 provides the performance capabilities of the Atlas IIIA/IIIB family and Table 2-3 shows the performance capabilities of the Atlas V 400/500 series.

As shown by the table, Atlas IIAS is capable of being launched from either Cape Canaveral Air Force Station (CCAFS) in Florida or Vandenberg Air Force Base (VAFB) in California. The Atlas III and V vehicles only have CCAFS capability at this time. This section further describes the Atlas family mission and performance options available with East and West Coast launches.

2.1 MISSION PERFORMANCE-LEVEL PHILOSOPHY

As Tables 2-1, 2-2 and 2-3 illustrate, Lockheed Martin offers a broad range of performance levels for the Atlas family of launch vehicles. In addition, Lockheed Martin can meet performance requirements by customizing (or standardizing) mission and trajectory designs to meet specific mission desires. To meet evolving commercial satellite mission launch requirements, Lockheed Martin offers performance capability levels (as opposed to explicit hardware configurations) as part of its standard launch services package.

Besides offering the performance-level quotes that are associated with each vehicle, other performance requirements can be met in several ways. With spacecraft missions that may require less performance than a specific configuration may offer, additional mission constraints will be used that will use excess performance to benefit the launch services customer. For example ascent trajectory designs can be shaped to improve ground coverage of major events or reduce the energy

Table 2-1 Atlas IIAS Performance Capability Summary

PSWC, kg (lb)	
Atlas IIAS	**Fairing**
Geosynchronous Transfer	
167x35,786 km (90x19,323 nmi), i = 27.0°, ω_p = 180°, 99% GCS	
3,719 (8,200)	LPF
Low-Earth Orbit from CCAFS	
185 km (100 nmi) Circular Orbit, i = 28.5°, 99.87% GCS	
8,618 (19,000)	LPF
Low-Earth Orbit from VAFB	
185 km (100 nmi) Circular Orbit, i = 90°, 99.87% GCS	
8,618 (19,000)	LPF
Note: PSWC: Payload Systems Weight Capability GCS: Guidance Commanded Shutdown LPF: Large 4.2-m (14-ft) Payload Fairing	

Table 2-2 Atlas IIIA/IIIB Performance Capabilities Summary

PSWC, kg (lb)		
Atlas IIIA	**Atlas IIIB**	**Fairing**
Geosynchronous Transfer 167x35,786 km (90x19,323 nmi), i = 27.0°, ωp = 180°, 99% GCS		
4,037 (8,900)	4,477 (9,870)*	EPF
4,060 (8,950)	4,500 (9,920)*	LPF
—	4,119 (9,080)	LPF
Low-Earth Orbit from CCAFS 185-km (100-nmi) Circular Orbit, i = 28.5°, 99.87% GCS		
8,641 (19,050)	10,718 (23,630)	EPF
8,686 (19,150)	10,759 (23,720)	LPF
Note: PSWC: Payload Systems Weight Capability GCS: Guidance Commanded Shutdown EPF: Extended 0.91-m (36-in.) 4.2-m (14-ft) Payload Fairing LPF: Large 4.2-m (14-ft) Payload Fairing		
* Dual Engine Centaur (IIIB DEC)		

Table 2-3 Atlas V 400/500 Performance Capabilities Summary

Orbit Type	Atlas V 400				Atlas V 500					
	Number of Solids				Number of Solids					
	0	1	2	3	0	1	2	3	4	5
	PSWC, kg (lb)									
GTO	4,950 (10,913)	5,950 (13,117)	6,830 (15,057)	7,640 (16,843)	3,970 (8,752)	5,270 (11,618)	6,285 (13,856)	7,200 (15,873)	7,980 (17,593)	8,670 (19,114)
GSO	N/A	N/A	N/A	N/A	N/A	N/A	2,680 (5,908)	3,190 (7,033)	3,540 (7,804)	3,810 (8,400)
LEO 28.5° inc	12,500* (27,558)*	N/A	N/A	N/A	10,300* (22,707)*	12,590* (27,756)*	15,080* (33,245)*	17,250* (38,029)*	18,955* (41,788)*	20,520* (45,238)*

Note:
- PSWC: Payload Systems Weight Capability
- GCS: Guidance Commanded Shutdown, 2.33 sigma (99%)
- GTO: ≥167x35,786 km (90x19,323 nmi), I = 27.0°, wp = 180°
- GSO: 35,786-km Circ (19,323-nmi Circ), I = 0°
- LEO 28.5° Inclination: 185-km Circ (100-nmi Circ), I = 28.5°
- GTO, GSO & LEO 28.5° Launch from CCAFS

Atlas V 500:
- GTO & GSO Performance Is with Single Engine Centaur (SEC)
- LEO 28.5° Performance Is with Dual Engine Centaur (DEC)
- Quoted Performance Is with 5-m Short PLF

Atlas V 400:
- GTO Performance Is with SEC
- LEO 28.5° Performance Is with DEC (402 Only)
- Quoted Performance Is with EPF

Note: *Payload Systems Weight Above 9,072 kg (20,000 lb) May Require Mission-Unique Accommodations.

required by the spacecraft to reach its final orbit. The ascent profile can be standardized to reduce mission integration analyses and/or schedules. These options can allow a more cost-effective solution for cases where maximum vehicle performance is not required.

2.2 MISSION DESCRIPTIONS

Atlas is a reliable and versatile launch system, capable of delivering payloads to a wide range of elliptical, low- and high-circular orbits; and Earth-escape trajectories. The Atlas IIAS, Atlas III and Atlas 400 series launch vehicles are available with either a large- or extended-length large payload fairing (LPF, or EPF), and are dedicated to a single payload. LMA is analyzing a 3-ft extension of the EPF, which will be denoted as the XEPF. The Atlas V 500 series launch vehicles with greater performance and larger fairings are well suited for launching single large satellites, multiple smaller satellites populating LEO constellations, or a mix of primary and secondary satellites. The trajectory design for each mission can be specifically tailored to optimize the mission’s critical performance parameter (e.g., maximum satellite orbit lifetime, maximum weight to transfer orbit) while satisfying satellite and launch vehicle constraints.

Atlas mission ascent profiles are developed using one or more Centaur upper-stage main engine burns. Each mission profile type is suited for a particular type of mission.

Direct Ascent Missions—With a one Centaur-burn mission design, the Centaur main engines are ignited just after Atlas/Centaur separation and the burn is continued until the Centaur and spacecraft are placed into the targeted orbit. Centaur/spacecraft separation occurs shortly after the burn is completed. Direct ascents are primarily used for low-Earth circular orbits and elliptic orbits with orbit geometries (i.e., arguments of perigee and inclinations) easily reached from the launch site. Orbits achievable with little or no launch vehicle yaw steering and those that can be optimally reached without coast phases between burns are prime candidates for the direct ascent mission design. Atlas/Centaur has flown 15 missions using the direct ascent design.

Parking Orbit Ascent Missions—The parking orbit ascent, used primarily for geosynchronous transfer missions, is the most widely used Atlas trajectory design. Performance capabilities are based on two Centaur burns injecting Centaur and the satellite into a transfer orbit selected to satisfy mission requirements. The first Centaur burn starts just after Atlas/Centaur separation and is used to inject the Centaur/spacecraft into a mission performance-optimal parking orbit. After a coast to the desired location the second Centaur main engine burn provides the impulse to place the satellite into the transfer and/or final orbit. If targeted to a transfer orbit, the satellite then uses its own propulsion system to achieve the final mission orbit. Missions requiring circular final orbits will use the second Centaur burn to circularize the satellite at the desired altitude and orbit inclination. More than 87 Atlas/Centaur missions have flown using the parking orbit ascent mission profile.

Geosynchronous Orbit (GSO) Mission—The GSO mission is a three-burn Centaur mission profile. This mission can only be flown by an Atlas V 500 configuration. This type of profile combines the parking orbit ascent to a geosynchronous transfer burn with a long coast followed by the third burn. The first Centaur burn starts just after Common Core Booster™ (CCB)/Centaur separation and is used to inject the Centaur/spacecraft into a mission performance-optimal parking orbit. After a coast to the desired location for transfer orbit injection, the second Centaur engine burn provides the impulse to place the spacecraft into the transfer orbit. A long transfer orbit coast follows the second burn. The Centaur main engine is ignited for a third time near the apogee of the transfer orbit. This final burn provides the energy to circularize at synchronous altitude and reduces the inclination to 0°.

2.3 ATLAS ASCENT PROFILES

Atlas IIAS and III

To familiarize users with Atlas and Centaur mission sequences, information is provided in the following paragraphs regarding direct and parking orbit ascent mission designs. Figures 2.3-1 and 2.3-2 show ascent sequences-of-events for a typical parking orbit ascent mission. Table 2.3-1 shows mission sequence data for Atlas IIAS and III vehicles for a typical geosynchronous transfer (GTO) mission. These data are representative; actual sequencing will vary to meet requirements of each mission. Atlas can be launched at any time of day to meet spacecraft mission requirements.

Atlas V 400 and 500

To familiarize users with Atlas V mission sequences, information is provided in the following paragraphs regarding typical Atlas V mission designs. Figures 2.3-3, 2.3-4, and 2.3-5 show sequences-of-events for a typical Atlas V 401 GTO, Atlas V 551 GTO, and Atlas V 552 low-Earth orbit (LEO) park orbit ascent missions, respectively. Table 2.3-2 shows the corresponding mission sequence data for the illustrated Atlas V ascent profiles. These data are representative; actual sat any time of day to meet spacecraft mission requirements.

2.3.1 Booster Phase

2.3.1.1 Atlas IIAS—Before liftoff, the booster ascent phase begins with ignition of the Rocketdyne MA-5A engine system. After passing the engine health-check the first pair of Thiokol Castor IVA solid rocket boosters (SRB) is ignited.

During the short vertical rise away from the pad, the vehicle rolls from the launch pad azimuth to the appropriate flight azimuth. At a vehicle-dependent altitude between 215 m (700 ft) and 305 m (1,000 ft), the vehicle begins pitching over into the prescribed ascent profile. At about 2,438 m (8,000 ft), the vehicle enters a nominal zero pitch and yaw angle of attack profile phase to minimize aerodynamic loads.

The ground-lit pair of SRBs burns out at about 54 seconds into flight. Ignition of the air-lit pair is governed by structural loading parameters. Ground-lit pair jettison occurs when range safety parameters are met. The air-lit pair is jettisoned shortly after burnout.

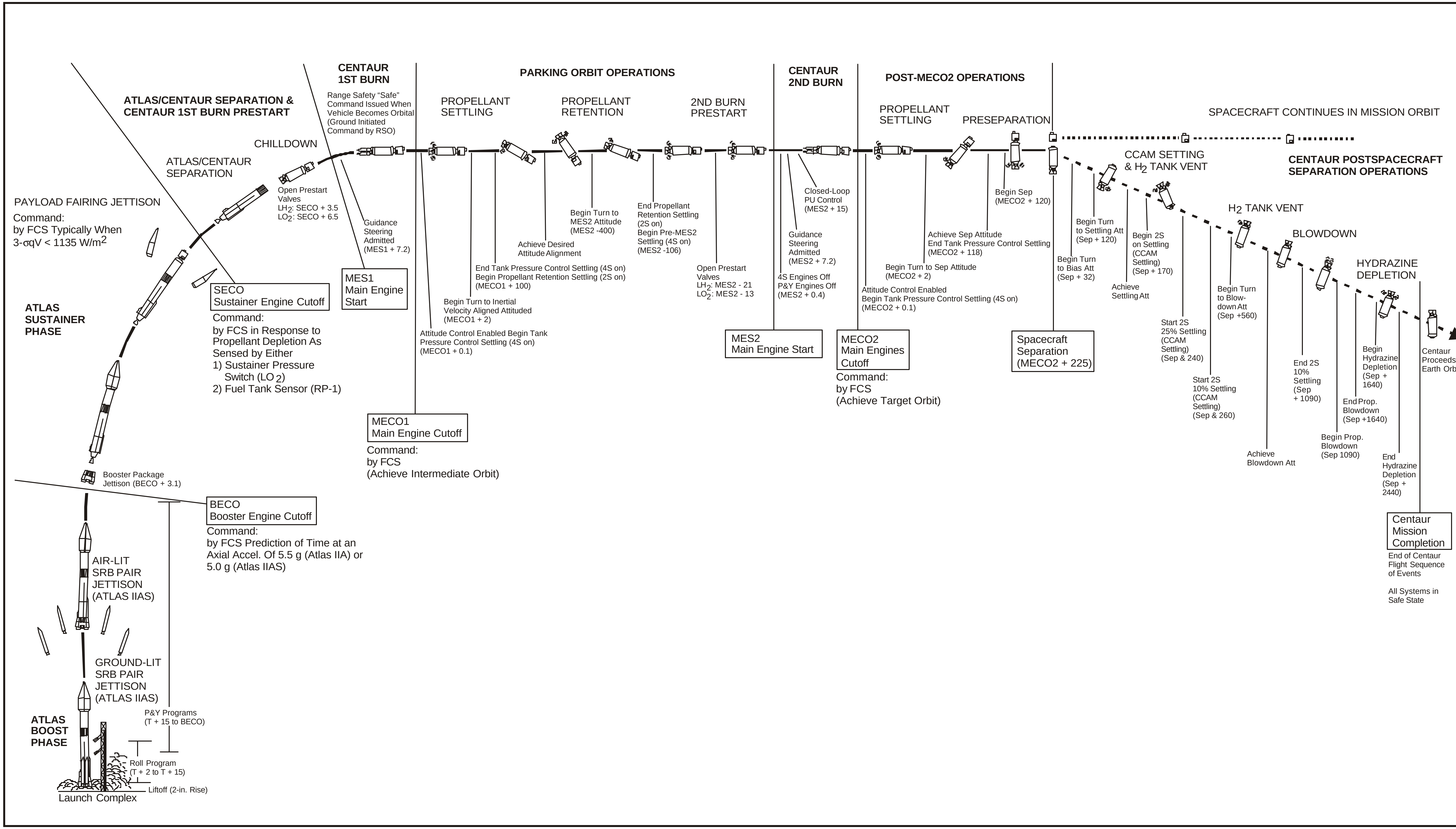
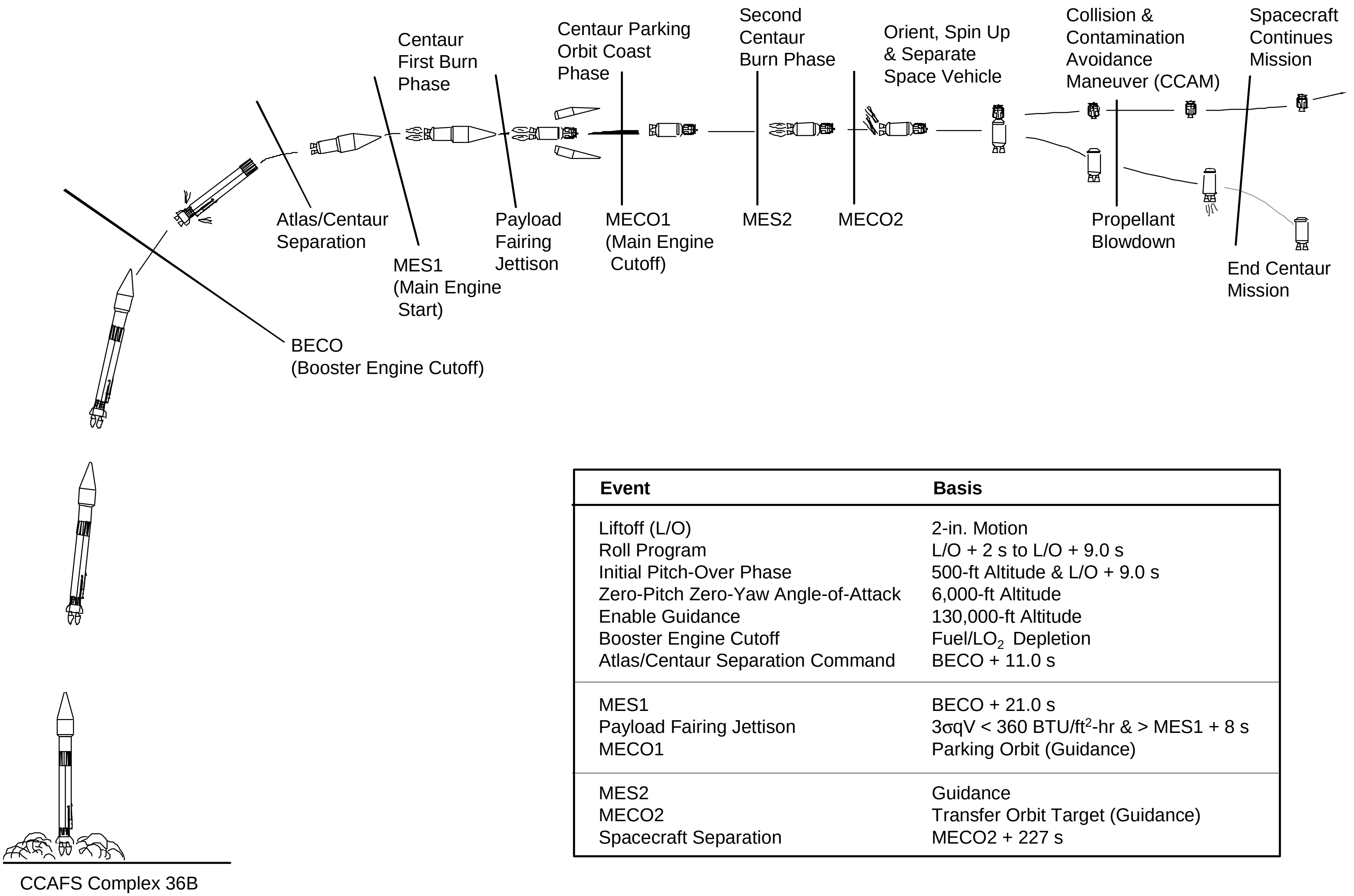


Figure 2.3-1 Typical Atlas IIA/IIAS Ascent Profile



Event	Basis
Liftoff (L/O)	2-in. Motion
Roll Program	L/O + 2 s to L/O + 9.0 s
Initial Pitch-Over Phase	500-ft Altitude & L/O + 9.0 s
Zero-Pitch Zero-Yaw Angle-of-Attack	6,000-ft Altitude
Enable Guidance	130,000-ft Altitude
Booster Engine Cutoff	Fuel/LO_2 Depletion
Atlas/Centaur Separation Command	BECO + 11.0 s
MES1	BECO + 21.0 s
Payload Fairing Jettison	$3\sigma qV$ < 360 BTU/ft^2-hr & > MES1 + 8 s
MECO1	Parking Orbit (Guidance)
MES2	Guidance
MECO2	Transfer Orbit Target (Guidance)
Spacecraft Separation	MECO2 + 227 s

Figure 2.3-2 Typical Atlas III Ascent Profile

Table 2.3-1 Typical Atlas IIAS/III GTO Mission Launch Vehicle Sequence Data

Atlas IIAS and III Events/Time from Liftoff	Atlas IIAS	Atlas IIIA	Atlas IIIB (DEC)	Atlas IIIB (SEC)
Guidance Go-Inertial	-11.0	-11.0	-11.0	-11.0
Ground-Lit SRB Pair Ignition (Atlas IIAS)	-0.5	—	—	—
Liftoff	0.0	0.0	0.0	0.0
Ground-Lit Pair Burnout (IIAS)	54.7	—	—	—
Air-Lit SRB Pair Ignition (IIAS)	60.0	—	—	—
Ground-Lit SRB Pair Jettison (IIAS)	77.1	—	—	—
Air-Lit SRB Pair Burnout (IIAS)	115.3	—	—	—
Air-Lit SRB Pair Jettison (IIAS)	117.2	—	—	—
Atlas Booster Engine Cutoff (BECO)	163.3	184.0	182.0	181.9
Booster Package Jettison (BPJ)	166.4	—	—	—
Payload Fairing Jettison (PFJ) (IIA/IIAS)	214.5	—	—	—
Atlas Sustainer Engine Cutoff (SECO)	289.2	—	—	—
Atlas/Centaur Separation	293.3	195.0	193.0	192.9
Begin Extendible Nozzle Deployment (When Used)	294.8	196.5	194.5	194.4
End Extendible Nozzle Deployment	301.8	203.5	201.5	201.4
Centaur Main Engine Start (MES1)	309.8	212.9	202.9	202.8
Payload Fairing Jettison (PFJ) (IIIA, IIIB DEC/SEC)	—	220.9	225.5	210.8
Centaur Main Engine Cutoff (MECO1)	584.8	767.7	541.7	929.3
Start Turn to MES2 Attitude	1,180.8	1,101.2	982.0	1,085.1
Centaur Main Engine Start (MES2)	1,475.8	1,401.2	1,380.0	1,483.1
Centaur Main Engine Cutoff (MECO2)	1,571.9	1,585.3	1,494.8	1,663.9
Start Alignment to Separation Attitude	1,573.9	1,587.3	1,496.8	1,665.9
Begin Spinup	1,691.9	1,717.3	1,626.8	1,795.9
Separate Spacecraft (Sep)	1,798.9	1,810.3	1,719.8	1,888.9
Start Turn to CCAM Attitude	1,918.9	1,930.3	1,839.8	2,008.9
Centaur End of Mission	4,239.9	4,250.3	4,159.8	4,328.9

The booster phase steering profile is implemented through our launch-day ADDJUST wind-steering programs, which enhance launch availability by controlling wind-induced flight loads. A zero total angle-of-attack flight profile is used from 2,438 m (8,000 ft) to minimize aerodynamic loads while in the low atmosphere. A small alpha-bias angle-of-attack steering technique is used after reaching 36,580 m (120,000 ft) until booster engine cutoff (BECO) to reduce gravity and steering losses.

Booster staging occurs when axial acceleration of 5.0 g attained. Earlier staging for reduced maximum axial acceleration and/or optimum mission design is easily accomplished with minor associated changes in performance.

After jettison of the booster engine and associated thrust structure, flight continues in the sustainer phase. Sustainer engine cutoff (SECO) occurs when all available sustainer propellants are consumed. Selected atmospheric ascent data from liftoff through first upper-stage burn accompany performance data for each launch vehicle (Ref Figs. 2.7-1 and 2.8-1).

For typical Atlas IIAS missions, the payload fairing is jettisoned before SECO, when 3-sigma free molecular heat flux falls below 1,135 W/m^2 (360 Btu/ft^2-hr). For sensitive spacecraft, payload-fairing jettison can be delayed later into the flight with some performance loss.

2.3.1.2 Atlas IIIA/IIIB—The Atlas IIIA/IIIB consists of a single-booster stage, unlike the Atlas IIAS, which is a one and one-half stage booster. On the ground, the RD-180 engine system (a single engine with two thrust chambers) is ignited and a health-check completed just before liftoff. The flight begins with a short vertical rise, during which the vehicle rolls to the flight azimuth. At an altitude of 152 m (500 ft) and time from liftoff greater than 9.0 seconds, the vehicle begins its initial pitch-over phase. At approximately 1,829 m (6,000 ft), the vehicle enters into a nominal zero-pitch and zero-yaw angle-of-attack phase to minimize aerodynamic loads. Zero-pitch and zero-yaw angle-of-attack and

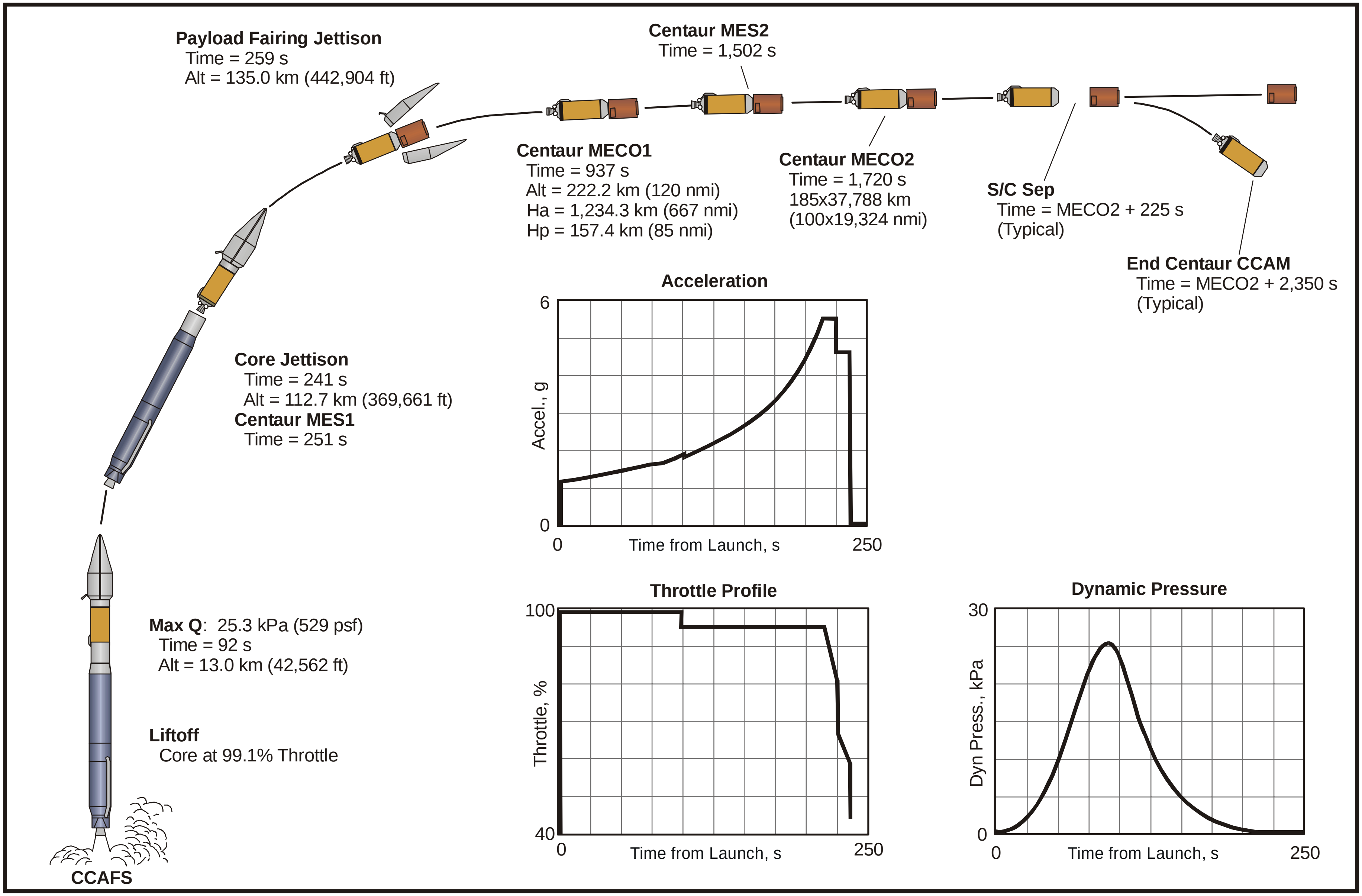


Figure 2.3-3 Typical Atlas V 401 GTO Ascent Profile

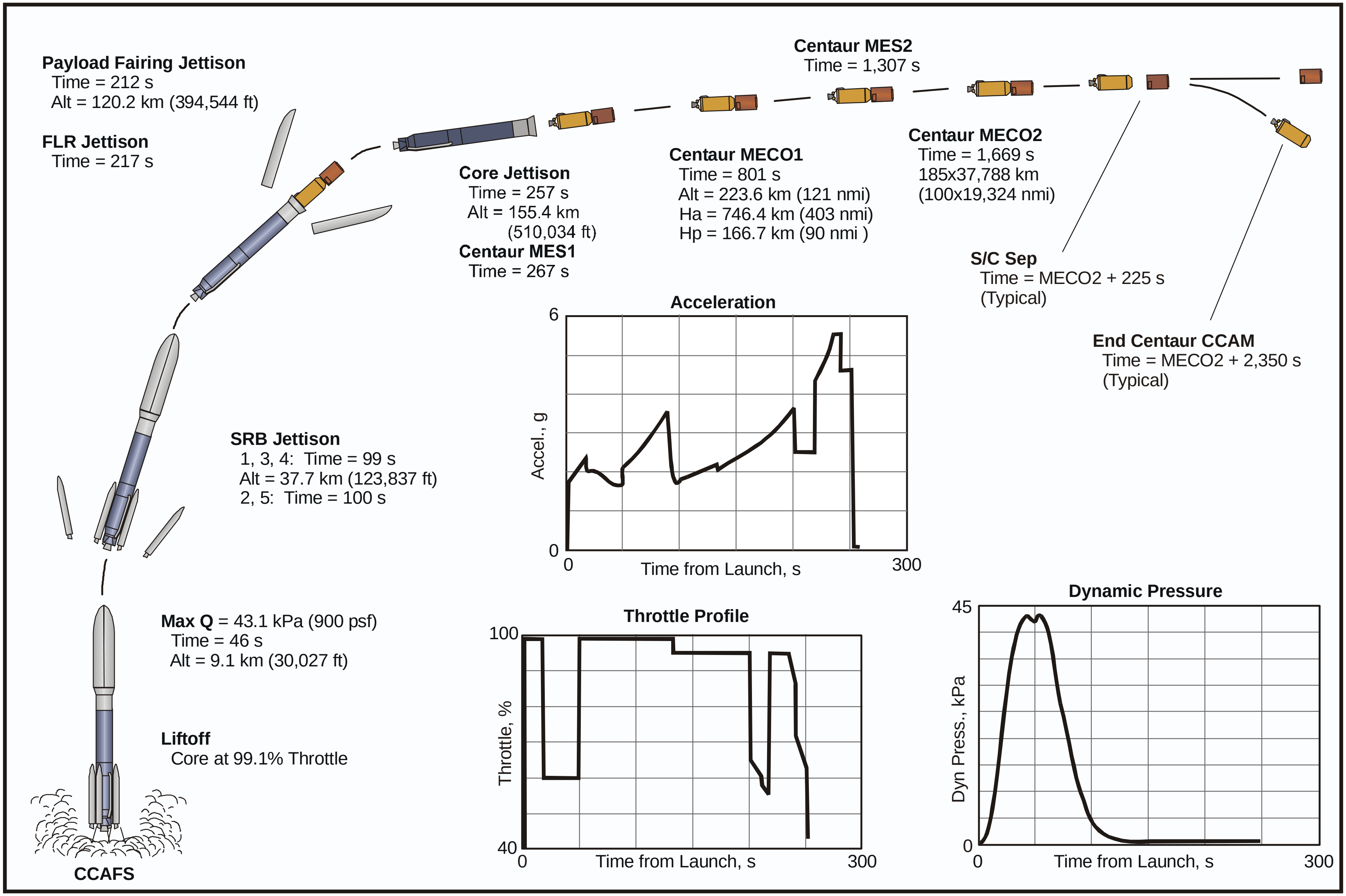


Figure 2.3-4 Typical Atlas V 551 GTO Ascent Profile

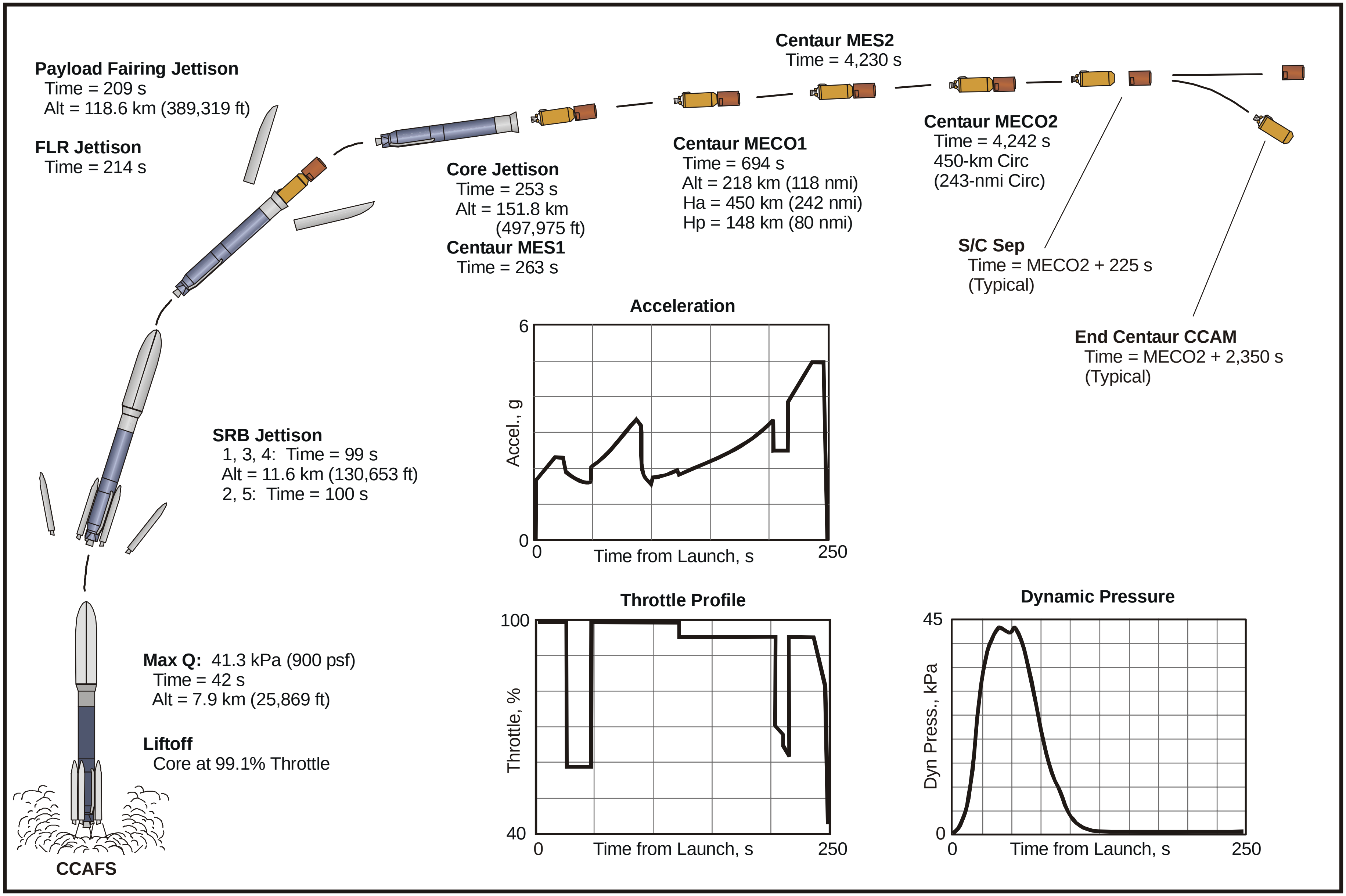


Figure 2.3-5 Typical Atlas V 552 LEO Ascent Profile

linear-tangent-steering phase is implemented through the ADDJUST-designed steering programs, which enhance launch availability by controlling wind-induced flight loads. Atlas flight continues in this guidance-steered phase until propellant depletion BECO. The boost phase of flight ends with booster separation 11 seconds after BECO when the separation charge attached to the forward ring of the ISA is fired. Twelve retrorockets are then fired to push the spent Atlas booster stage away from the Centaur upper stage.

The RD-180 throttle profile design is based on launch vehicle constraints; the throttle profile is optimized on a mission specific basis to obtain the nominal prelaunch trajectory. Liftoff throttle setting is limited to reduce the liftoff, acoustics on the vehicle, and spacecraft. At 120 feet acoustics have decreased sufficiently to allow the engines to throttle up. The transonic throttle setting will be optimized and uplinked to mitigate the wind-induced variation in dynamic pressure. Based on the desire to maintain a robust design, inflight Mach is computed by the flight software and used as criteria for transonic throttle-down and throttle-up. This allows thrust dispersion effects to be sensed for real-time adjustment of these throttle events to control the variation in dynamic pressure. The RD-180 booster engine operates at almost full throttle until vehicle acceleration reaches a certain limit. The throttle setting is then ramped down at a predetermined rate. Guidance actively controls the throttle setting to maintain a constant acceleration until 10 seconds before booster engine cut-off.

Table 2.3-2 Typical Atlas V Mission Launch Vehicle Sequence Data

Atlas V Events/ Time from Liftoff	551 GTO	401 GTO	552 LEO
Guidance Go-Inertial	-11	-11	-11
Liftoff	0	0	0
Strap-On SRM Jettison (SRM 1, 3, 4)	99	-	99
Strap-On SRM Jettison (SRM 2, 5)	100	-	100
Payload Fairing Jettison (PFJ)	212	259	209
Core Engine Cutoff	252	236	248
Core Jettison	257	241	253
Centaur Main Engine Start (MES1)	267	251	263
Centaur Main Engine Cutoff (MECO1)	801	937	694
Start Turn to MES2 Attitude	907	1,162	3,830
Centaur Main Engine Start (MES2)	1,307	1,502	4,230
Centaur Main Engine Cutoff (MECO2)	1,669	1,720	4,242
Start Alignment to Separation Attitude	1,679	1,730	4,252
Separate Spacecraft (SEP)	1,894	1,945	4,467
Start Turn to CCAM Attitude	1,994	2,045	4,567
Centaur End of Mission	4,019	4,070	6,592

2.3.1.3 Atlas V 400—The Atlas V 400 consists of a CCB combined with zero to three strap-on solid rocket boosters (SRB). The booster phase begins with ignition of the RD-180 engine system. The vehicle is held down during start and a portion of throttle up. A vehicle health check is made before achieving full throttle. After passing the health check the hold-down bolts are blown, and throttle up is completed. After a short vertical rise away from the pad, at an altitude of 244 m (800 ft), the 401 and 402 vehicles begin a roll and pitch maneuver to the prescribed ascent profile and direction. The 400 vehicles with SRBs start open-loop pitchover at 76 m (250 ft).

At approximately 0.35 mach the vehicle begins transition to a nominal zero-alpha and zero-beta angle-of-attack phase to minimize aerodynamic loads. This phase is implemented through the launch-day wind-steering system, which enhances launch availability by reducing wind-induced flight loads.

For Atlas V 401 or 402 configurations, after reaching 24,380 m (80,000 ft) until approximately 36,576 m (120,000 ft), an alpha-bias angle-of-attack steering technique may be used to reduce steering losses while maintaining aerodynamic loading within acceptable limits. The booster phase steering profile through the end of alpha-biased steering is implemented through our launch-day wind-steering system, which enhances launch availability by reducing wind-induced flight loads.

At the end of alpha-biased steering, closed-loop guidance steering is enabled. For all 400 vehicles with SRBs the zero-alpha/zero-beta attitude is maintained until 5 seconds after the SRB jettison event when closed-loop guidance steering is enabled. The strap-on SRB jettison sequence is initiated after burnout. At 99 seconds into flight all SRBs are jettisoned.

Near the end of the CCB phase, the RD-180 engine is continuously throttled so that a specific axial acceleration level is not exceeded. This g-level is a function of payload weight and does not exceed 5.5g.

The RD-180 cutoff sequence is initiated when a propellant low-level sensor system indicates that the booster is about to deplete all available propellants. At this time, the engine is throttled down to minimum power level and shut down. All Atlas V 400 configurations retain the payload fairing through the booster phase of flight.

2.3.1.4 Atlas V 500—The Atlas V 500 consists of a CCB combined with zero to five strap-on solid rocket boosters (SRB). The RD-180 and SRB ignition sequence is the same as the Atlas V 400. After a short vertical rise away from the pad, at an altitude of 244 m (800 ft), the 501 and 502 vehicles begin a roll and pitch maneuver to the prescribed ascent profile and direction. The 500 vehicles with SRBs start open-loop pitchover at 76 m (250 ft). At approximately 0.35 mach the vehicle begins transition to a nominal zero-alpha and zero-beta angle-of-attack phase to minimize aerodynamic loads. This phase is implemented through the launch-day wind-steering system, which enhances launch availability by reducing wind-induced flight loads.

For Atlas V 501 or 502 configurations, an alpha-bias angle-of-attack steering technique may be used after reaching 24,380 m (80,000 ft) until approximately 33,528 m (110,000 ft) to reduce steering losses and maintain aerodynamic loading within acceptable limits. At the end of alpha-biased steering, closed-loop guidance steering is enabled. For all 500 vehicles with SRBs the zero-alpha/zero-beta attitude is maintained until 5 seconds after the last SRB jettison event when closed-loop guidance steering is enabled. The strap-on SRB jettison sequence is initiated after burnout. At 99 seconds into flight, the first three SRBs are jettisoned. The last SRBs are then jettisoned 100 seconds into flight.

For Atlas V 500 missions, the payload fairing is jettisoned during the CCB phase. Before payload fairing jettison, the RD-180 engine is throttled down to maintain acceleration at 2.5 g. Typically, the PLF is jettisoned when the 3-sigma free molecular heat flux falls below 1,135 W/m^2 (360 Btu/ft^2-hr). For sensitive spacecraft, payload-fairing jettison can be delayed later into the flight with some performance loss. After payload fairing jettison, the RD-180 is throttled back up.

Near the end of the CCB phase, the RD-180 is continuously throttled so that a specific axial acceleration level is not exceeded. This g-level is a function of payload weight and does not exceed 5.5g.

The engine cutoff sequence is initiated when a propellant low-level sensor system indicates that the core booster is about to deplete all available propellants. At this time, the RD-180 is throttled down to minimum power level and the engine is shut down.

2.3.2 Centaur Phase

2.3.2.1 Atlas IIAS—Centaur main engine start (MES or MES1) occurs 16.5 seconds after the Atlas stage is jettisoned. For direct ascent missions, the Centaur main burn injects the spacecraft into the targeted orbit and then performs a series of preseparation maneuvers. With parking orbit ascent missions, the Centaur first burn (typically the longer of the two) injects the spacecraft into an elliptic performance-optimal parking orbit. After first-burn main engine cutoff (MECO1), the Centaur and spacecraft enter a coast period. During the coast period (about 14 minutes for a typical geosynchronous transfer mission), the Centaur normally aligns its longitudinal axis along the velocity vector. Because typical parking orbit coasts are of short duration, most spacecraft do not require special pointing or roll maneuvers. Should a spacecraft require attitude maneuvers during coast phases, Centaur can accommodate all roll axis alignment requirements and provide roll rates up to 1.5 ±0.3°/s in either direction during non-thrusting periods. Greater roll rates can be evaluated on a mission-peculiar basis. Before the second Centaur burn main engine start (MES2), the vehicle is aligned to the ignition attitude and the engine start sequence is initiated.

At a guidance-calculated start time, Centaur main engines are reignited and the vehicle is guided to the desired orbit. After reaching the target, main engines are shut down (MECO2), and Centaur begins its alignment to the spacecraft separation attitude. Centaur can align to any attitude for separation. Preseparation spinups to 5.0 ±0.5 rpm about the roll axis can be accommodated. In addition, a pitch/yaw plane transverse spin mode can also be used.

After Centaur/spacecraft separation, Centaur conducts its collision and contamination avoidance maneuver (CCAM) to prevent recontact and minimize contamination of the spacecraft.

2.3.2.2 Atlas IIIA/IIIB—The baseline Atlas IIIA Centaur uses a single-engine Centaur configuration. The Atlas IIIB Centaur can use either a dual-engine (DEC) or single-engine (SEC) configuration. After BECO and Atlas/Centaur separation, the Centaur stage ignites its main engines (MES1). The payload fairing (PLF) is then jettisoned while the 3-sigma free-molecular-heat flux is below 1,135 W/m^2 (360 Btu/ft^2-hr), but not before 8 seconds after MES1. The Centaur first burn continues, which is the longer of the two Centaur firings, and injects the vehicle into an elliptic, performance-optimal parking orbit. During the first Centaur burn, the instantaneous impact point (IIP) trace progresses across the Atlantic Ocean and is over the equatorial region of the African continent for less than 1 second.

After Centaur first-burn main-engine cutoff (MECO1), the Centaur and spacecraft enter a coast period. At a guidance-calculated start time near the Equator, the Centaur main engine is reignited (MES2). The vehicle is then steered by guidance into GTO. The Centaur mission on Atlas IIIA/IIIB from this point on is very similar to a Centaur mission on Atlas IIAS.

2.3.2.3 Atlas V 400 and 500—The Atlas V Centaur can use either a dual engine configuration (DEC) or single engine (SEC) configuration. Centaur main engine start (MES) occurs approximately 10 seconds after the CCB stage is jettisoned. For typical Atlas V 400 missions, the payload fairing is jettisoned 8 seconds after MES1, by which time the 3-sigma free molecular heat flux has typically fallen below 1,135 W/m2 (360 Btu/ft^2-hr). For sensitive spacecraft, payload-fairing jettison can be delayed later into the flight with some performance margin loss.

For direct ascent missions, a single, long-duration main engine burn injects the spacecraft into the targeted orbit after which Centaur performs a series of preseparation maneuvers. With parking orbit ascent missions, the first Centaur burn (typically the longer of the two) injects the spacecraft into an elliptical performance-optimal parking orbit. After first-burn main engine cutoff (MECO1), the Centaur and spacecraft enter a coast period (about 10 minutes for an SEC geosynchronous transfer mission) during which the Centaur aligns its longitudinal axis along the velocity vector. Should a spacecraft require attitude maneuvers during longer coast phases, Centaur can accommodate roll axis alignment requirements and provide commanded roll rates from 0.5 to 1.5°/s in either direction during nonthrusting periods. Accommodation of larger roll rates can be evaluated on a mission-unique basis. Before the second Centaur burn main engine start (MES2) the vehicle is aligned to the ignition attitude and, at a guidance-calculated start time, the Centaur main engine is reignited and the vehicle is guided to the desired orbit.

After reaching the target orbit, the main engine is shut down (MECO2) and Centaur begins its alignment to the spacecraft separation attitude. Centaur can align to any attitude for separation.

Preseparation spinups of up to 5.0 ±0.5 rpm about the roll axis can be accommodated. In addition, a pitch/yaw plane transverse spin mode can be used.

The Atlas V 500 series vehicles have the capability to perform three Centaur burns and long coast periods. For geosynchronous missions the Centaur is equipped with an extended mission kit (EMK), which includes additional battery power, additional helium pressurant gas, a full compliment of hydrazine maneuvering propellant and radiation shielding over the hydrogen tank to reduce boiloff. After MECO2 the Centaur enters a long 5-hour coast to geosynchronous altitude during which it performs numerous thermal conditioning maneuvers. Again, at a guidance-calculated time, the Centaur aligns to the burn attitude, the main engine is reignited (MES3) and the vehicle is guided to the desired orbit

After Centaur/spacecraft separation, Centaur conducts a collision and contamination avoidance maneuver (CCAM) to prevent recontact and minimize contamination of the spacecraft. A blowdown of remaining Centaur propellants follows.

2.3.3 Injection Accuracy and Separation Control

Atlas' combination of precision guidance hardware with flexible guidance software provides accurate payload injection conditions for a wide variety of missions. In response to changing mission requirements, minimal data are required to specify targeted end conditions to provide rapid preflight retargeting. These functional capabilities have been demonstrated on many low-Earth orbit (LEO), GTO, lunar, and interplanetary missions.

Injection accuracies for a variety of GTO and LEO missions are displayed in Table 2.3.3-1 and are typical of 3-sigma accuracies following final upper-stage burn. On all missions to GTO, Atlas has met all mission injection accuracy requirements.

Past lunar and interplanetary mission accuracy requirements and achievements are in Table 2.3.3-2. With the exception of the SOHO mission, these accuracies resulted from use of the older inertial measurement group (IMG) flight computer and would be further improved with use of our current inertial navigation unit (INU) guidance system. On some planetary missions, the guidance requirement included orientation of a spacecraft-imbedded solid rocket kick stage to achieve proper final planetary intercept conditions. The major error source on these missions was the uncertainty of solid rocket kick stage impulse.

2.3.3.1 Attitude Orientation and Stabilization—During coast phases, the guidance, navigation, and control (GN&C) system can orient the spacecraft to any desired attitude. The guidance system can reference an attitude vector to a fixed inertial frame or a rotating orthogonal frame defined by the instantaneous position and velocity vector. The reaction control system (RCS) autopilot incorporates three-axis-stabilized attitude control for attitude hold and maneuvering. In addition to a precision attitude control mode for spacecraft preseparation stabilization, Centaur can provide a stabilized spin rate to the spacecraft about any desired axis, including arbitrary combinations of vehicle primary axes. The Centaur system can accommodate longitudinal spin rates up to 5.0 ±0.5 rpm, subject to some limitation due to space vehicle mass property misalignments. Spin rates about transverse axes have been demonstrated up to 7°/s. A detailed analysis for each Centaur/spacecraft combination will determine the maximum achievable spin rate. Furthermore, known rates imparted to the spacecraft by the separation system (e.g., due to spacecraft center of gravity offsets) can be compensated for by imparting an equal and opposite rate before the separation event. This pre-compensation feature can improve the final spacecraft pointing, nutation, and transverse rates, subject to spin rate requirements and attitude control constraints.

Table 2.3.3-1 Typical Injection Accuracies at Spacecraft Separation

Atlas IIAS/IIIA										
Orbit at Centaur Spacecraft Separation				**± 3-sigma Errors**						
Mission	**Apogee, km (nmi**	**Perigee, km (nmi)**	**Inclin-ation, °**	**Semi-major Axis, km (nmi)**	**Apogee, km (nmi)**	**Perigee, km (nmi)**	**Inclin-ation, °**	**Eccentricity**	**Argument of Perigee, °**	**RAAN, °**
GTO	35,941 (19,407)	167 (90)	27.0	N/A	117 (63.2)	2.4 (1.3)	0.02	N/A	0.23	0.26
GTO	35,949 (19,411)	167 (90)	22.1	N/A	109 (58.9)	2.2 (1.2)	0.02	N/A	0.19	0.21
Super-synchron-ous	123,500 (66,685)	167 (90)	27.5	625 (337)	1,250 (675)	2.8 (1.5)	0.01	N/A	0.17	0.26
Inter-mediate Circular Orbit	10,350 (5,589)	10,350 (5,589)	45.0	42.7 (23.0)	N/A	N/A	0.07	0.002	N/A	0.08
Elliptical Transfer	10,350 (5,589)	167 (90)	45.0	N/A	40.0 (21.6)	2.40 (1.3)	0.07	N/A	0.07	0.08
LEO (Circular)	1,111 (600)	1,111 (600)	63.4	19.4 (10.5)	N/A	N/A	0.15	0.004	N/A	0.11
Legend: N/A = Not Applicable or Available										

Atlas IIIB/Atlas V										
Orbit at Centaur Spacecraft Separation				**± 3-sigma Errors**						
Mission	**Apogee, km (nmi)**	**Perigee, km (nmi)**	**Inclin-ation, °**	**Semi-major Axis, km (nmi)**	**Apogee, km (nmi)**	**Perigee, km (nmi)**	**Inclin-ation, °**	**Eccentricity**	**Argument of Perigee, °**	**RAAN, °**
GTO (Coast ≤ 800 s)	35,897 (19,383)	195 (105)	25.6	N/A	168 (91)	4.6 (2.5)	0.025	N/A	0.2	TBS
GTO (Coast ≥ 800 s) Atlas V 400 Series	35,765 (19,312)	4,316 (2,330)	21.7	N/A	238 (129)	12.0 (6.5)	0.025	N/A	0.37	0.39
GTO (Coast ≥ 800 s) Atlas V 500 Series	TBS	TBS	TBS	TBS	TBS	TBS	TBS	TBS	TBS	TBS
Super-synchron-ous	77,268 (41,722)	294.5 (159)	26.4	N/A	586 (316)	3.6 (1.9)	0.02	N/A	0.32	—
Legend: N/A = Not Applicable or Available										

The extensive capabilities of the GN&C system allow the upper stage to satisfy a variety of spacecraft orbital requirements, including thermal control maneuvers, sun-angle pointing constraints, and telemetry transmission maneuvers.

2.3.3.2 Separation Pointing Accuracies—Pointing accuracy just before spacecraft separation is a function of guidance system hardware, guidance software, and autopilot attitude hold capabilities. In the nonspinning precision pointing mode, the system can maintain attitude errors less than 0.7°, and attitude rates less than 0.2, 0.2, and 0.25°/s about the pitch, yaw, and roll axes, respectively (before separation) (Table 2.3.3.2-1). Although the attitude and rates of a nonspinning spacecraft after separation (after loss

Table 2.3.3-2 Lunar and Interplanetary Mission Accuracy

Mission	Figure of Merit (FOM), m/s — Mission Requirement	Figure of Merit (FOM), m/s — Guidance System 1σ
Surveyor	< 50	7.0
Mariner Mars	< 13.5	3.5
Mariner Venus Mercury	< 13.5	2.4
Pioneer 10	< 39	39*
Pioneer 11	< 36	36*
Viking 1	< 15	3.6
Viking 2	< 15	3.5
Voyager 1	< 21	16.2*
Voyager 2	< 21	17.6*
Pioneer Venus 1	< 7.5	2.3
Pioneer Venus 2	< 12.0	3.2
SOHO	< 10.3	6.9
Note: * Major Error Source-Solid Kickstage Motor		

Table 2.3.3.2-1 Summary of Guidance and Control Capabilities

Coast Phase Attitude Control	
• Roll Axis Pointing, °, Half Angle	> 1.6
• Passive Thermal Control Commanded Rate, °/s (Clockwise or Counterclockwise)	
– Minimum	0.5
– Maximum	1.5
Centaur Separation Parameters at Separation Command (with No Spin Requirement)	
• Pitch, Yaw, Roll Axis Pointing, °, Half Angle	> 0.7
• Body Axis Rates, °/s	
– Pitch	±0.2
– Yaw	±0.2
– Roll	±0.25
Spacecraft Separation Parameters at Separation Command (with Transverse Spin Requirement)	
• Transverse Rotation Rate, °/s	> 7.0
Spacecraft Separation Parameters Following Separation (with Nonspinning or Slowspinning Requirement)	
• Pitch, Yaw & Roll Axis Pointing (per Axis), °	> 0.7
• Body Axis Rates, °/s	
– Pitch	±0.6
– Yaw	±0.6
– Roll	±0.5
Spacecraft Separation Parameters Following Separation (with Longitudinal Spin Requirement)	
• Nutation, °, Half Angle	> 5.0
• Momentum Pointing, °, Half Angle	> 3.0
• Spin Rate, °/s	> 30.0 ±3.0
• Note: Capabilities Are Subject to Spacecraft Mass Properties Limitations	

of contact between the Centaur and the spacecraft) are highly dependent on mass properties of the spacecraft, attitude typically can be maintained within 0.7° per axis, body axis rates are typically less than 0.6°/s in the pitch or yaw axis and 0.5°/s in the roll axis. The angular momentum of the spacecraft after separation is often a concern. Total spacecraft angular momentum is typically less than 15 N-m-s. Separation conditions for a particular spacecraft are assessed during the mission-peculiar separation analysis.

Centaur can also use a transverse spin separation mode in which an end-over-end “rotation” is initiated before separating the payload. A rotation rate of up to 7°/s is possible about the pitch or yaw axis for typical spacecraft.

Any other axis of spin can be achieved before separation. For example, the spin axis can be aligned with the principal axes of the spacecraft (as provided by the customer). The magnitude of the spin rate achievable for an arbitrary axis will depend upon spacecraft mass properties, and must be determined on a mission peculiar basis.

For a mission requiring preseparation spinup, conditions just before spacecraft separation combine with any tipoff effects induced by the separation system and any spacecraft principal axis misalignments to produce postseparation momentum pointing and nutation errors. Here, nutation is defined as the angle between the actual space vehicle geometric spin axis and the spacecraft momentum vector. Although dependent on actual spacecraft mass properties (including uncertainties) and the spin rate, momentum pointing and maximum nutation errors following separation are typically less than 3.0 and 5.0°, respectively.

The Atlas V Centaur III flight control systems have been generically designed to accommodate payloads that fall within the range of payload mass properties, correlated with specific vehicle maneuvers, as identified in Table 2.3.3.2-2. The payload mass properties identified in this table include

Table 2.3.3.2-2 Design Range of Payload Mass Properties

Atlas V Config.	Prepayload Separation Maneuver	Payload Mass, kg (lbm)	Forward cg Location, * mm (in.)	Lateral cg Offset, mm (in.)	Moments of Inertia, kg-m^2 (slug-ft^2)	Products of Inertia, kg-m^2 (slug-ft^2)
CIIIS	5-rpm Longitudinal Axis Spin	910–5,670 (2,000–12,500)	1,016–4,572 (40–180) **	±12 (±0.5)	Ixx= 410–5,420 (300–4,000) Iyy= 390–9,490 (285–7,000) Izz= 39–9,490 (285–7,000)	Ixy= ± 68 (± 50) Ixz= ± 68 (± 50) Iyz= ± 340 (± 250)
CIIID	5-rpm Longitudinal Axis Spin	4,080–5,670 (9,000–12,500)	1,778–4,572 (70–180) **	±12 (±0.5)	Ixx= 2,580–5,420 (1,900–4,000) Iyy= 2,030–9,490 (1,500–7,000) Izz= 2,030–9,490 (1,500–7,000)	Ixy= ± 68 (± 50) Ixz= ± 68 (± 50) Iyz= ± 340 (± 250)
All	7-deg/s Transverse Axis Spin	1,810–5,670 (4,000–12,500)	1,016–3,302 (40–130)	±76 (±3)	Ixx= 1,080–5,420 (800–4,000) Iyy= 1,360–9,490 (1,000–7,000) Izz= 1,360–9,490 (1,000–7,000)	Ixy= ± 135 (± 100) Ixz= ± 135 (± 100) Iyz= ± 135 (± 100)
400	3-Axis Stabilized Attitude Hold	910–9,070 (2,000–20,000)	1,016–4,572 (40–180)	±127 (±5)	Ixx= 410–10,850 (300–8,000) Iyy= 390–27,100 (285–20,000) Izz= 390–27,100 (285–20,000)	Ixy= ± 910 (± 670) Ixz= ± 910 (± 670) Iyz= ± 910 (± 670)
500	3-Axis Stabilized Attitude Hold	1,360–19,050 (3,000–42,000)	1,016–5,715 (40–225)	±127 (±5)	Ixx= 680–40,700 (500–30,000) Iyy= 1,020–258,000 (750–190,000) Izz= 1,020–258,000 (750–190,000)	Ixy= ± 2,700 (± 2,000) Ixz= ± 2,700 (± 2,000) Iyz= ± 2,700 (± 2,000)

* Longitudinal cg Location Is Tabulated here as Inches Forward of the Forward Ring of the Centaur Forward Adapter.

** Longitudinal cg Position Is Constrained as a Function of Payload Mass. Payload Longitudinal cg Position Is Restricted to 100 in. above the Forward Ring of the Centaur Forward Adapter for a 5,800-lbm Payload. Linear Interpolation from this Point to Extreme Points Is Used.

By Definition, the x-axis Is the Centaur Longitudinal Axis with Positive Direction Measure Forward. The y-axis Is the Centaur Pitch Axis, and the z-axis Is the Centaur Yaw Axis.

the spacecraft(s), the payload adapter, payload separation system, and the associated 3-sigma uncertainties. Payloads, which fall outside of these generic design ranges, may be accommodated on a mission peculiar basis.

2.3.3.3 Separation System—The relative velocity between the spacecraft and the Centaur is a function of the mass properties of the separated spacecraft and the separation mechanism. Our separation systems are designed to preclude recontact between the spacecraft and Centaur and provide adequate separation for collision and contamination avoidance. Typically, the separation system achieves at least 0.9 ft/s relative separation velocities.

2.4 PERFORMANCE GROUND RULES

Atlas performance ground rules for various missions with launch from CCAFS in Florida or VAFB in California are described in this section.

2.4.1 Payload Systems Weight Definition

Performance capabilities quoted throughout this document are presented in terms of payload systems weight (PSW). Payload systems weight is defined as the total mass delivered to the target orbit, including the separated spacecraft, the spacecraft-to-launch vehicle adapter, and all other hardware required on the launch vehicle to support the payload (e.g., payload flight termination system, harnessing). Table 2.4.1-1 provides masses for our standard payload adapters (Ref Sect. 4.1.2 and Appendix E for payload adapter details). Data are also provided estimating performance effects of various mission-peculiar hardware requirements. As a note, performance effects shown are approximate. The launch vehicle trajectory, spacecraft mass, and mission target orbit can affect the performance contributions of each mission peculiar item.

2.4.2 Payload Fairings

2.4.2.1 Atlas IIAS—Atlas IIAS performance shown in this document is based on use of the 4.2-m (14-ft) diameter LPF. For spacecraft that require greater volume than the standard LPF, a 0.9-m (3-ft) stretch to our large fairing has been developed (EPF). Performance for IIAS GTO missions will degrade approximately 45 kg (100 lb) with its use. Additional fairing information is in Section 4.1.

2.4.2.2 Atlas IIIA/IIIB—Atlas IIIA performance is based on use of the 4.2-m (14-ft) diameter, 3-ft stretched EPF. Atlas IIIB performance is based on use of the 4.2-m (14-ft) diameter LPF.

2.4.2.3 Atlas V 400—Atlas V 400 performance is based on use of the 4.2-m (14-ft) diameter, 3-ft stretched EPF. GTO performance with the 4.2-m (14-ft) diameter LPF is approximately 35 kg more with the 401 configuration and 44 kg more with the 431 configuration. The XEPF (EPF + 3 ft.) has approximately the same payload partial.

2.4.2.4 Atlas V 500—Atlas V 500 configuration performance is based on use of the 5-m short PLF. For spacecraft that require greater volume, the 5-m medium PLF is available. For Atlas V 500 configurations that use the 5-m medium PLF, the typical range of performance degradation is 59 kg (131 lb) for the Atlas V 511 and 521, to 67 kg (147 lb) for the Atlas V 501 GTO missions. Additional fairing information can be found in Section 4.1.

2.4.3 Launch Vehicle Performance Confidence Levels

Atlas missions are targeted to meet the requirements of each user. Historically, Atlas and most U.S.-launched missions have been designed with a 3-sigma performance confidence level (99.87% probability of hitting target orbit elements). With the flexibility of Atlas/Centaur hardware and flight software, performance confidence levels can be set based on each mission's requirements. The minimum residual shutdown (MRS) performance option, discussed later in this section, takes full advantage of this concept. All Earth escape-related performance, IIAS and III LEO and VAFB data in this document are based on the 3-sigma confidence level.

Table 2.4.1-1 Performance Effects of Spacecraft-Required Hardware

Performance Effect of Payload Adapter Masses, kg (lb)			
Type A 44 (97)	Type C-15 32.6 (72)	Type C-22 45.3 (100)	Type D 54 (119)
Type B1194VS 91 (200)	Type B1 53 (116)	Type C-13 29.5 (65)	Type E 98 (215)
Type D1666VS 91 (200)	Type F1663 91 (200)		
Performance Effect of Other Spacecraft-Required Hardware, kg (lb)			
Standard Package 8 (18)	PLF Acoustic Panels 11 (25) LPF	PLF Thermal Shield 4 (9) LPF	Environment Verification Pkg (Telepak, Instruments) 9 (20)
Standard package on Centaur consists of flight termination system (FTS) and airborne harness; payload fairing standard package consists of two standard access doors, reradiating antenna, and customer logo.			
Other Hardware			
• Centaur Hardware Affects Performance at 1-kg (2.2-lb) Mass to 1-kg (2.2-lb) Performance Ratio • Payload Fairing Hardware Affects Performance at ~ 9-kg (19.8-lb) Mass to 1-kg (2.2-lb) Performance Ratio			

For Atlas V Lockheed Martin has baselined a 99% confidence-level performance reserve. Because many of today's communications satellites can benefit from reduced launch vehicle confidence levels (and associated nominal performance increases), MRS data are also discussed. Lockheed Martin will respond to any desired performance confidence-level requirement needed by the user.

2.4.4 Centaur Short-Burn Capability

For LEO mission applications, Lockheed Martin has evaluated launch vehicle requirements for short-duration Centaur second burns. With missions requiring short-duration second burns (10-30 seconds), propellant residuals will be biased to ensure proper engine propellant inlet conditions at MES. Centaur main engine burns as short as 10 seconds are possible. All performance data shown using short-duration burns include performance effects of propellant-level control at MES2.

2.4.5 Centaur Long-Coast Capability

The standard Centaur incorporates items supporting park orbit coast times up to 65 minutes. These items include 150-amp-hr main vehicle battery, shielding on the Centaur aft bulkhead, and three helium bottles. A Centaur extended-mission kit has been developed to support long-duration Centaur parking orbit coasts. Coasts of up to 3-hours in duration are doable and are constrained by helium pressurant and hydrazine reaction control system (RCS) propellant expenditure. The long-coast kit consists of a larger vehicle battery, shielding on the Centaur aft bulkhead, additional helium capacity, and an additional hydrazine bottle. Longer duration missions can be supported by the 500 series with the addition of a fourth helium bottle, an additional second hydrazine load, substitution of two 250-amp-hr main vehicle batteries for the 150 amp-hr battery, and an LH_2 tank sidewall radiation shield. Thus equipped, the 500 series can support a full three-burn GSO mission.

Performance estimates using long parking or transfer orbit coasts include the effect of an extended-mission kit. See Section 8.1.1 for additional details.

2.4.6 Heavy-Payload Lift Capability

Centaur equipment module and payload adapters have been optimized for geosynchronous transfer missions. To manage the larger payload masses (typically greater than 4,080 kg [9,000 lb]), two heavy-payload interfaces have been identified. Figure 2.4.6-1 illustrates these interfaces. The strengthened equipment module is in production. In both cases the user must account for the mass of a spacecraft-to-launch vehicle adapter. In addition, the stated performance penalty must be accounted for if the truss adapter is exercised. See Section 8.2.1 for additional details.

2.5 GEOSYNCHRONOUS LAUNCH MISSION TRAJECTORY AND PERFORMANCE OPTIONS

Through Centaur's flexible flight software, a number of trajectory designs are possible. Depending on mission requirements, total satellite mass, dry mass-to-propellant mass ratio, and type of satellite propulsion (liquid or solid) system, one of the following trajectory design options will prove optimal:

1) Geosynchronous transfer (and reduced inclination transfers),
2) Supersynchronous transfer,
3) Subsynchronous transfer and perigee velocity augmentation.

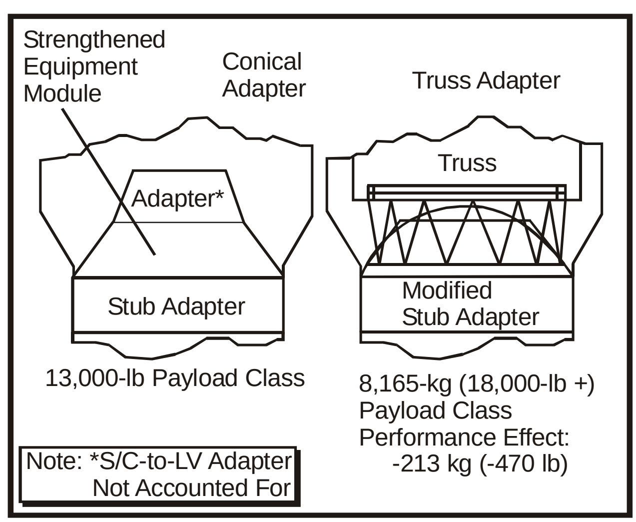


Figure 2.4.6-1 Heavy Payload Interfaces

2.5.1 Geosynchronous Transfer

The GTO mission is the standard mission design for communications satellite launches. Figure 2.5.1-1 illustrates the orbital mission profile involved. The transfer orbit inclination achieved depends on launch vehicle capability, satellite launch mass, and performance characteristics of both systems. Based on the performance of the Atlas family and enhanced capabilities of today's liquid apogee engine (LAE) subsystems, Lockheed Martin is finding that a 27° inclination is optimal for maximizing satellite beginning-of-life mass given an optimally sized satellite propulsion system. The 300-plus-second specific impulses of current LAEs have resulted in a shift in optimum inclination from 26.5 to 27°. With satellites weighing less than the GTO capability of the launch vehicle, excess performance can be used to further reduce inclination, raise perigee or both.

Although the GTO design is intuitively the standard launch option, single-payload manifesting allows the option of alternate designs that can extend geostationary satellite lifetimes. Supersynchronous transfers, subsynchronous transfers, and other mission enhancement options can enhance lifetime with satellites that use common sources of liquid propellant for orbit insertion and on-orbit stationkeeping.

2.5.2 Supersynchronous Transfer

The supersynchronous trajectory design offers an increase in beginning-of-life propellants by minimizing the delta-velocity required of the satellite for orbit insertion. This option is available if the launch vehicle capability to standard GTO is greater than the payload system weight (PSW). Initially excess capability is used to increase apogee altitude. If the apogee altitude capability exceeds the satellite maximum allowable altitude, excess launch vehicle performance is used to lower orbit inclination. Optimally, when inclination reaches about 20 degrees the perigee altitude starts to increase rapidly. The delta-velocity required of the spacecraft to reach GSO continues to decline. At supersynchronous altitudes, the decreased inertial velocity allows the satellite to make orbit plane changes more efficiently. The satellite makes the plane change and raises perigee to geosynchronous altitude in one or more apogee burns. It then coasts to perigee and circularizes into final geostationary orbit (GSO). The total delta-velocity in this supersynchronous transfer design is less than would be required to inject from an equivalent performance reduced inclination geosynchronous transfer, resulting

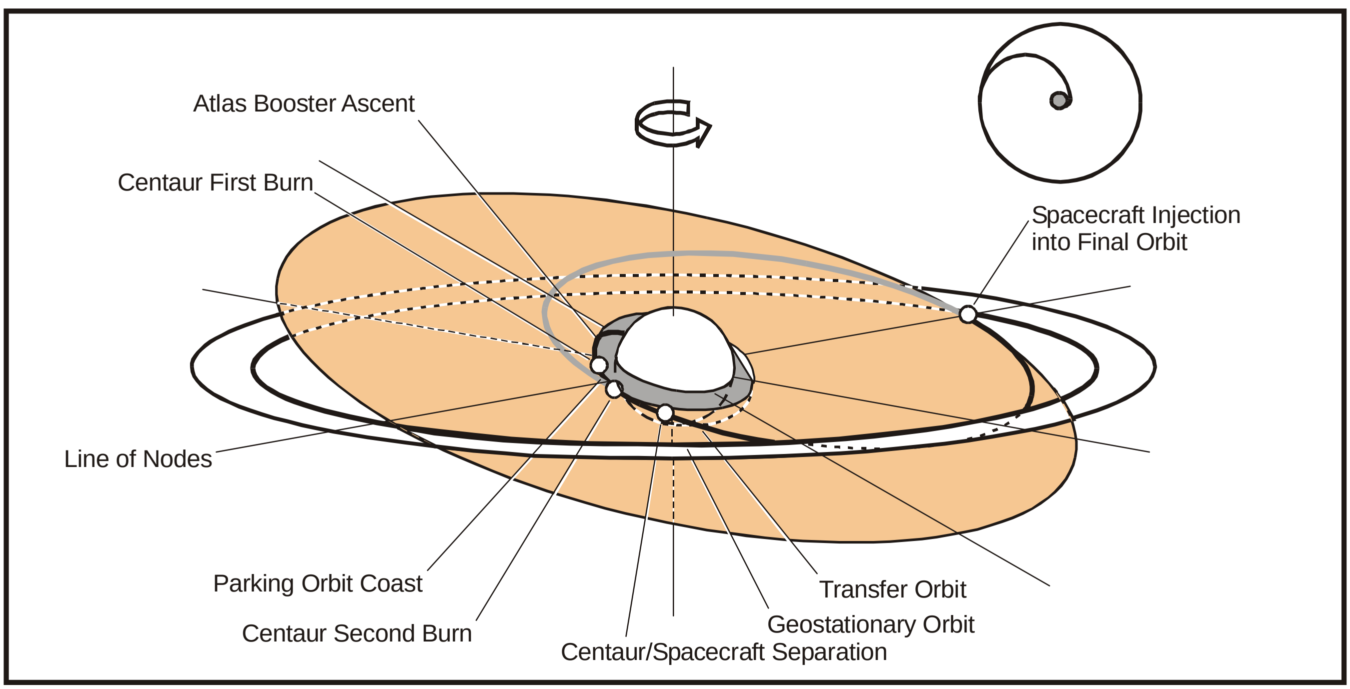


Figure 2.5.1-1 The Geosynchronous Transfer Orbit Mission Trajectory Profile

in more satellite propellants available for on-orbit operations. Figure 2.5.2-1 illustrates the supersynchronous trajectory mission profile. Table 2.5.2-1 quantifies potential mission gains with the supersynchronous mission for a 1,850-kg (4,078-lb) satellite. Supersynchronous transfer trajectories have been flown on 25 missions starting with the Atlas II/EUTELSAT II (AC-102) mission launched in December 1991.

2.5.3 Subsynchronous Transfer and Perigee Velocity Augmentation

The perigee velocity augmentation (PVA) trajectory design, compared with the standard GTO design, can provide increased propellant mass at beginning-of-life on GSO. This is beneficial when propellant tank capacity is large with respect to the dry mass. Atlas delivers the satellite to a subsynchronous intermediate transfer orbit (apogee less than geosynchronous) with an inclination of approximately 27° because the satellite mass exceeds GTO launch capability. The separated satellite coasts to subsequent transfer orbit perigee(s), where the satellite supplies the required delta-velocity for insertion into geosynchronous transfer. At apogee, using one or more burns, the satellite lowers inclination and circularizes into GSO. As illustrated in Table 2.5.3-1, mass at beginning-of-life is enhanced. The orbit profile is shown in Figure 2.5.3-1. Several Atlas flights, including six UHF/EHF missions, have successfully used the subsynchronous transfer option.

2.6 MISSION OPTIMIZATION AND ENHANCEMENT

Atlas trajectory designs are developed using an integrated trajectory simulation executive and a state-of-the-art optimization algorithm (sequential quadratic programming [SQP]). This optimization capability shapes the trajectory profile from liftoff through spacecraft injection into targeted orbit.

The SQP technique uses up to 50 independent design variables chosen to maximize a performance index and satisfy specified constraints. Typical control variables include boost-phase initial pitch and roll (launch azimuth) maneuvers, Atlas sustainer steering, and Centaur steering for all burns. In addition, spacecraft pitch and yaw attitudes and ignition times can be included as part of the total optimization process. The optimization program is formulated with up to 40 equality and inequality constraints on variables, such as dynamic pressure, tracker elevation angle, and range safety.

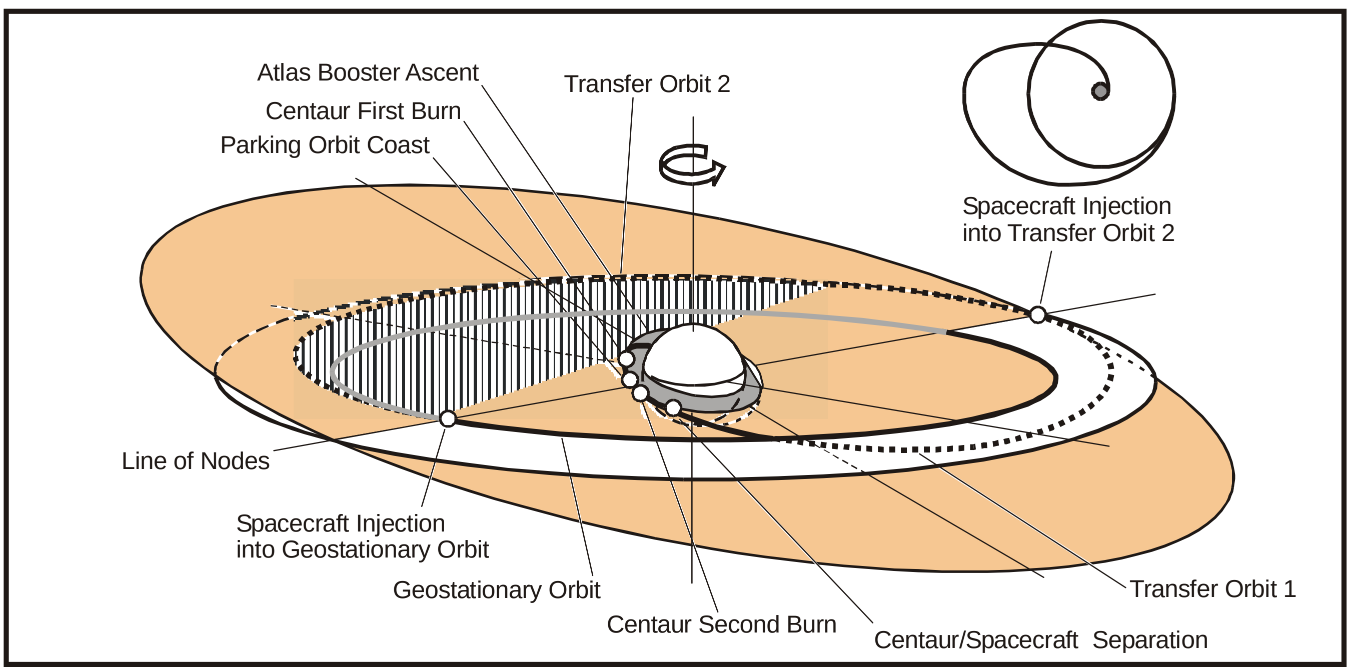


Figure 2.5.2-1 The Supersynchronous Transfer Orbit Mission Trajectory Profile

Table 2.5.2-1 Mission Benefits of Supersynchronous Transfer

Parameter	GTO	Super-synch
S/C Mass, kg	1,850	1,850
Transfer Orbit Parameters • Perigee Altitude, km • Apogee Altitude, km • Orbit Inclination, ° • Argument of Perigee,°	 391 35,786 19.6 180	 183 50,000 21.2 180
Final Orbit	GSO	GSO
S/C ΔV Required for GSO Insertion, m/s	1,643	1,607
Estimated Mission Lifetime, years	6.7	7.2
S/C Gains Propellant for Additional 0.5 year of Lifetime		

Table 2.5.3-1 Mission Benefits of Subsynchronous Transfer

Parameter	GTO	Sub-synch
S/C Mass, kg	3,654	3,654
S/C Offload To Meet GTO Launch Capability, kg	-160	0
Transfer Orbit Parameters • Perigee Altitude, km • Apogee Altitude, km • Orbit Inclination, ° • Argument of Perigee, °	 167 35,786 27.0 180	 167 30,000 27.0 180
Final Orbit	GSO	GSO
S/C ΔV Required for GSO Insertion, m/s	1,806	1,913
S/C Mass at Beginning of Life, kg	1,916	1,933
Estimated Mission Lifetime, years	12.1	12.6
S/C Gains Propellant for Additional 0.5 year of Lifetime		

With Lockheed Martin's experience with launch of interplanetary and scientific spacecraft, an additional trajectory analysis tool is available to assist in the mission design/mission optimization process. The N-BODY trajectory simulation program is used, with our optimization capability, in developing launch vehicle missions requiring precision inertial targeting in which perturbation effects of other celestial bodies are required to be considered.

These trajectory analysis tools, and our extensive guidance and targeting capabilities, enable Atlas mission optimization to include spacecraft characteristics and programmatic goals. The most widely used mission enhancement options are:

1) Inflight retargeting (IFR),
2) Minimum residual shutdown (MRS),
3) IFR/MRS,
4) Explicit right ascension of ascending node (RAAN) control.

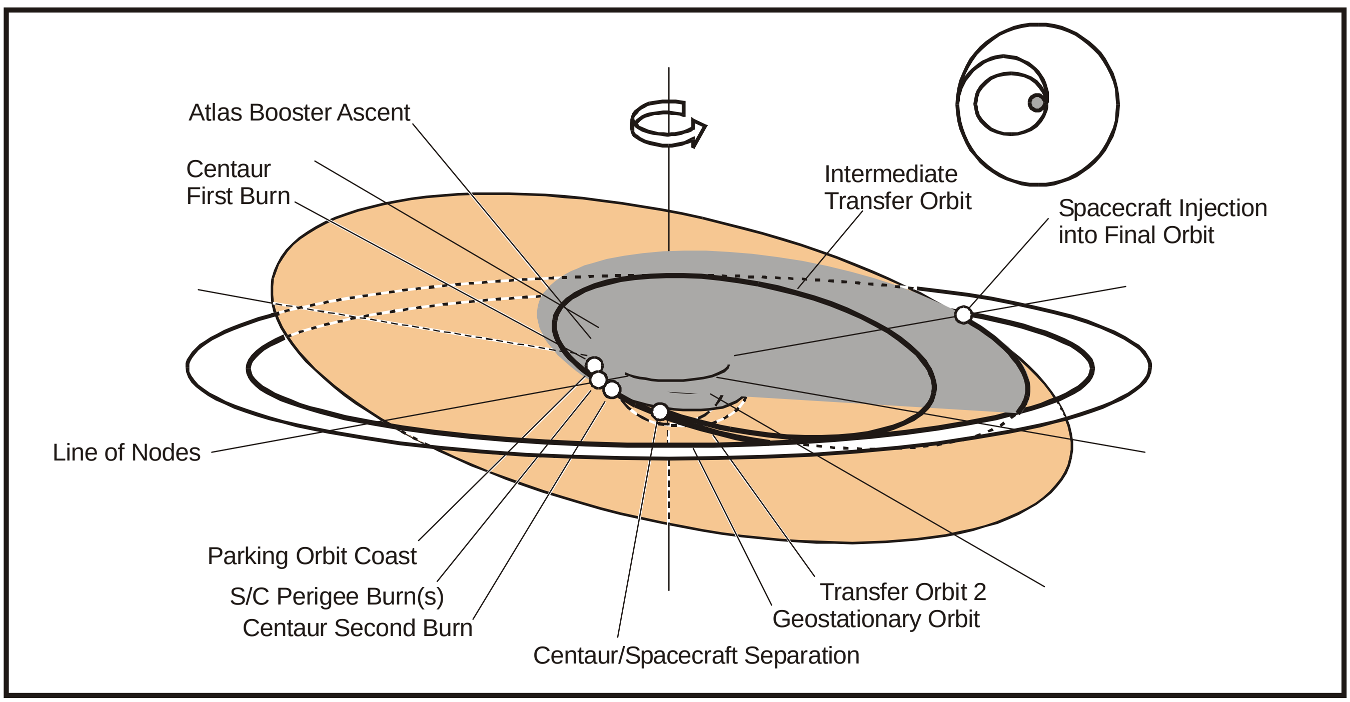


Figure 2.5.3-1 The Subsynchronous Transfer Orbit Mission Trajectory Profile

2.6.1 Inflight Retargeting (IFR)

The software capability of the Centaur upper stage makes it possible to evaluate Atlas performance in flight and then target for an optimal injection condition that is a function of the actual performance of the booster stage. Centaur can be retargeted to a variable transfer orbit inclination, apogee, perigee, argument of perigee, or any combination of the above. IFR can provide a dual performance benefit. First, the nominal launch vehicle flight performance reserve (FPR) is reduced when the FPR contribution due to Atlas dispersions is eliminated. Second, whether Atlas performance is high, nominal, or low, the retargeting logic is calibrated to devote all remaining propellant margin to benefit the mission. With IFR, any desired level of confidence of a guidance shutdown can be implemented. IFR has been successfully flown on 15 missions starting with the Atlas II/EUTELSAT II mission.

2.6.2 Minimum Residual Shutdown (MRS)

Centaur propellants may be burned to minimum residuals for a significant increase in nominal performance capability. When burning to minimum residuals, FPR propellants are eliminated to nominally gain additional delta-velocity from Centaur.

It is practical for Centaur to burn all its propellants when the satellite has a liquid propulsion system that is capable of correcting for variations in launch vehicle performance. This option is particularly attractive and appropriate when the trajectory design includes a supersynchronous or subsynchronous (PVA) transfer orbit. MRS is not an option for satellites using solid propellant (fixed impulse) OISs, because FPR propellants are required to ensure that the Centaur injection conditions will match the capability of the fixed impulse stage.

When Centaur burns all propellants to minimum residuals, the liquid propellant satellite corrects for the effects of launch vehicle dispersions. These dispersions primarily affect apogee altitude. Variations in other transfer orbit parameters are minor. The performance variation associated with MRS can also be quantified as an error in injection velocity that can be approximated as a dispersion in transfer orbit perigee velocity. Table 2.6.2-1 documents MRS perigee velocity injection variations for the Atlas IIAS and Atlas IIIA/IIIB. MRS has been successfully executed for 31 missions and has become the typical operations mode for GTO-type missions.

2.6.3 IFR/MRS Combination

This mission design allows the transfer orbit inclination target to vary with Atlas stage performance while apogee altitude varies with Centaur stage performance. As with MRS alone, no Centaur propellants remain to guarantee a specific transfer orbit target so this design takes full advantage of flight performance. IFR/MRS can be used with an apogee cap or unconstrained apogee but the range of apogee altitudes is reduced by using a part of the performance for inclination reduction.

2.6.4 Right Ascension of Ascending Node (RAAN) Control

Some satellite mission objectives may require launch-on-time placement into transfer and/or final orbit. For Earth orbital missions, this requirement typically manifests itself as a RAAN target value or range of values. Centaur's heritage of meeting the inertial orbit placement requirements associated with planetary missions makes it uniquely capable of targeting to an orbit RAAN (or range of RAANs dictated by actual launch time in a launch window) in addition to the typical or target parameters. With GTO missions, some satellite mission operational lifetimes can be enhanced by controlling RAAN of the targeted transfer orbit. A satellite intended to operate in a non-0° geosynchronous final orbit can benefit with proper RAAN placement. A drift

Table 2.6.2-1 Atlas IIAS, IIIA and IIIB Orbit Injection Performance Variations with MRS

Perigee Velocity Dispersions	2.33 sigma	3 sigma
Atlas IIAS	66.5 m/s (218 ft/s)	85.3 m/s (280 ft/s)
Atlas IIIA	59.4 m/s (195 ft/s)	76.5 m/s (251 ft/s)
Atlas IIIB	60.4 m/s (198 ft/s)	77.7 m/s (255 ft/s)

toward a 0° inclination orbit can help reduce the typical north-south stationkeeping budget of the satellite, thereby increasing the amount of time the satellite can remain in an operational orbit.

Control of the node, and therefore RAAN, is obtained by varying the argument of perigee of the transfer orbit and the satellite burn location with respect to the Equator. The difference between the inclination of the transfer orbit and final orbits and the latitude of the satellite burn determines the amount of nodal shift between the transfer orbit and the mission orbit. Control of this shift is used to compensate for off-nominal launch times, keeping the inertial node of the final orbit fixed throughout a long-launch window. Centaur software can be programmed to control the argument of perigee (satellite burn location) as a function of time into the launch window to obtain the desired final orbit inertial node. Atlas IIA/INMARSAT-3 missions used this technique to gain an additional 3 years of mission lifetime.

2.7 ATLAS IIAS PERFORMANCE DATA

For Atlas IIAS, relevant atmospheric ascent parameters are shown in Figure 2.7-1. Detailed performance data are provided in Figures 2.7-2 through 2.7-3 and 2.7-6 through 2.7-11. Data are shown for several types of launch missions and are based on the vehicle's lift capability. Tabular performance data for both the 99% confidence-level GCS and MRS are provided in Tables 2.7-1 and 2.7-2. Performance ground rules are shown on each curve and additional information is provided in the following paragraphs.

2.7.1 Elliptical Transfer Capability

The optimum trajectory profile for achieving elliptical transfer orbits is the parking orbit ascent. Atlas performance capability for 27° inclined orbits is shown in Figures 2.7-2 and 2.7-3. The 27° inclined orbit missions are launched at flight azimuths that have been approved and flown for many missions and are near optimal for geostationary transfer missions. The Centaur second burn is executed near the first descending node of the parking orbit (near the Equator).

For some missions, an ascending node injection into the transfer orbit may offer advantages. The performance degradation and mission constraints associated with this mission type can be analyzed on a mission-peculiar basis.

2.7.2 Reduced Inclination Elliptical Transfer Capability

The inclination effect on payload systems weight capability for a geosynchronous transfer orbit design for inclinations between 30° and 18°. Performance degrades as inclination drops below 28.5° in part because the launch vehicle IIP trace is constrained by Range Safety requirements to remain a minimum range approved distance off the Ivory Coast of Africa. This requirement dictates that yaw steering be implemented in the ascent phase to meet Range Safety requirements while attempting to lower park orbit inclination toward the desired transfer orbit target value. The remaining inclination is completed with yaw steering in the Centaur second burn. Data are shown at the 99% confidence-level GCS and MRS for transfer orbit apogee altitudes of 35,788 km (19,324 nmi) and 100,000 km (53,996 nmi).

2.7.3 Earth-Escape Performance Capability

Earth-escape mission performance is shown in Figures 2.7-6. Centaur's heritage as a high-energy upper stage makes it ideal for launching spacecraft into Earth-escape trajectories. Performance data shown use the parking orbit ascent design and a near-planar ascent to an orbit that contains the outgoing asymptote of the escape hyperbola with a 1-hour coast time between the upper-stage burns. Performance improves as coast time is shortened. The actual coast time necessary to achieve the desired departure asymptote will be determined by specific mission requirements. Our quoted performance assumes that 3-sigma (99.87% confidence level) flight performance reserves are held.

Additional performance data are shown for an optional vehicle configuration that uses the parking orbit ascent design with a customer-supplied third stage, a near-optimum size solid propellant orbit insertion stage (OIS) based on the Thiokol STAR 48V (TEM-711-18) motor. This vehicle configuration is advantageous for missions that require a very high-energy Earth departure, cases in which vehicle staging effects make it more efficient for a third stage to provide an additional energy increment. The reference performance mission was targeted similarly to the no-OIS case with the STAR 48V burn occurring just after Centaur/STAR 48V separation.

2.7.4 Low-Earth Orbit Capability

Atlas can launch payloads into a wide range of LEOs from Cape Canaveral using direct ascent or parking orbit ascent mission profiles. LEO capabilities typically require heavy-payload modifications.

Direct Ascent to Circular Orbit—Figure 2.7-7 shows circular orbit payload systems weight capability to LEO using the one Centaur burn mission profile. The maximum capability is available with planar ascent to a 28.5° inclination orbit. As shown, inclinations from 28.5 up to 55° are possible with the direct ascent. Given known Range Safety constraints, direct ascent performance to inclinations greater than 55° are not possible due to land overflight constraints up the Eastern seaboard of the United States and Canada. Direct ascent performance to reduced inclination orbits (down to ~22°) is also possible, but at the expense of substantial performance due to Range Safety overflight constraints over the Ivory Coast of Africa.

Direct Ascent to Elliptical Orbit—Figure 2.7-7 also shows elliptical orbit performance capability using the direct ascent with perigee altitude at 185 km (100 nmi). Similar Range Safety and orbital mechanics constraints limit inclinations available with a Florida launch.

Parking Orbit Ascent to Circular Orbit—Payload delivery to low-altitude circular orbit can be accomplished by two or more upper-stage burns. The first Centaur burn is used to inject the Centaur and payload into an elliptic parking orbit. A park orbit perigee altitude of 148 km (80 nmi) is assumed for our reference cases. Expected parking orbit coast durations will require use of the Centaur extended mission kit. The second Centaur burn will circularize the spacecraft into the desired orbit altitude.

Circular orbit performance capabilities for altitudes between 400 km (216 nmi) and 2,000 km (~1,000 nmi) are shown for various inclinations in Figure 2.7-7. Data are shown for 28.5, 45, 55, 60 and 63.4° inclinations. With high-inclination orbits (inclinations greater than 55°), Range Safety requirements require that Atlas meet instantaneous impact constraints up the Eastern seaboard of the United States and Canada. Additional inclination is added in the later stages of the Centaur first burn and with the second Centaur burn. As desired orbit inclination increases, performance degradations become more pronounced. High-inclination orbit performance capabilities are more optimally achieved with launch from VAFB (Sect. 2.7.6).

2.7.5 Intermediate Circular Orbits

Performance data are shown in Figure 2.7-8 for altitudes between 5,000 km (2,700 nmi) and 20,000 km (11,000 nmi). Similar ground rules apply to intermediate circular orbit data as to LEO circular orbit data except with respect to heavy payload requirements. The lower performance capabilities associated with the higher energy circular missions should allow use of standard payload interfaces.

2.7.6 VAFB Elliptical Orbit Transfer Capability

Atlas IIAS (only, at this time) can launch payloads into high-inclination elliptical transfer from our West Coast launch site at VAFB. Transfer orbits at 63.4° are of interest because the rotation rate of the line of apsides is zero. These missions use a parking orbit ascent mission profile with the second burn executed near the first antinode (argument of perigee equaling 270°). Figure 2.7-9 shows performance capability for elliptical transfer orbits with apogee altitudes between 5,000 km (2,700 nmi) and 50,000 km (27,000 nmi). Performance is shown for both 99.87% confidence-level GCS and MRS.

2.7.7 VAFB LEO Performance

Figures 2.7-10 provides low-Earth orbit performance data for launches from our West Coast launch site. Orbits with inclinations ranging from 60° through Sun-synchronous are shown. Data are provided for mission types similar in scope to those discussed in Section 2.7.4.

2.7.8 VAFB High-Inclination, High-Eccentricity Orbit Capability

From VAFB, Atlas can insert payloads into orbits with 12-hour or 24-hour periods at an inclination of 63.4°. With the rotation rate of the line of apsides being zero, these orbits repeat their ground trace. These missions use a park orbit ascent mission similar to that described in Section 2.7.6. Figure 2.7-11 shows performance capability for these high-inclination, high-eccentricity orbits with perigee altitudes between 250 km (135 nmi) and 2,500 km (1,350 nmi). Performance capability is shown for 99.87% confidence-level GCS.

2.8 ATLAS IIIA PERFORMANCE DATA

Figure 2.8-1 illustrates the ascent characteristics and Figures 2.8-2 and 2.8-3 and Tables 2.8-1 and 2.8-2 illustrate the performance of the Atlas IIIA vehicle to geotransfer orbits as described in Section 2.7. Figure 2.8-6 shows escape capability; and Figures 2.8-7 and 2.8-8 show LEO and intermediate circular orbit performance capability.

2.9 ATLAS IIIB (DEC) PERFORMANCE DATA

Figure 2.9-1 illustrates the ascent characteristics and Figures 2.9-2 and 2.9-3 and Tables 2.9-1 and 2.9-2 illustrate the performance of the Atlas IIIB DEC vehicle to geotransfer orbits as described in Section 2.7. Figure 2.9-6 shows escape capability; and Figures 2.9-7 and 2.9-8 show LEO and intermediate circular orbit performance capability.

2.10 ATLAS IIIB (SEC) PERFORMANCE DATA

Figure 2.10-1 illustrates the ascent characteristics and Figures 2.10-2 and 2.10-3 and Tables 2.10-1 and 2.10-2 illustrate the performance of the Atlas IIIB SEC vehicle to geotransfer orbits as described in Sections 2.7.1 and 2.7.2 only.

2.11 ATLAS V 401 AND 402 PERFORMANCE DATA

Figures 2.11-1a and 2.11-1b illustrate the ascent characteristics of the 401. Figures 2.11-2 and 2.11-3 and Tables 2.11-1 and 2.11-2 illustrate the performance of the Atlas V 401 vehicle to geotransfer orbits as described in Sections 2.7.1 and 2.7.2. Figure 2.11-4 depicts the 401’s minimum delta-velocity to geosynchronous orbit with various apogee caps. Figure 2.11-6 shows escape capability without the Star 48V solid motor for the 401. Figure 2.11-7 shows Atlas V 402 LEO performance.

2.12 ATLAS V 411 PERFORMANCE DATA

Figures 2.12-1a and 2.12-1b illustrate the ascent characteristics of the 411. Figures 2.12-2 and 2.12-3 and Tables 2.12-1 and 2.12-2 illustrate the performance of the Atlas V 411 vehicle to geotransfer orbits as described in Sections 2.7.1 and 2.7.2. Figure 2.12-4 depicts the minimum delta-velocity to geosynchronous orbit with various apogee caps. Figure 2.12-6 shows escape capability without the Star 48V solid motor.

2.13 ATLAS V 421 PERFORMANCE DATA

Figures 2.13-1a and 2.13-1b illustrate the ascent characteristics of the 421. Figures 2.13-2 and 2.13-3 and Tables 2.13-1 and 2.13-2 illustrate the performance of the Atlas V 421 vehicle to geotransfer orbits as described in Sections 2.7.1 and 2.7.2. Figure 2.13-4 depicts the minimum delta-velocity to geosynchronous orbit with various apogee caps. Figure 2.13-6 shows escape capability without the Star 48V solid motor.

2.14 ATLAS V 431 PERFORMANCE DATA

Figures 2.14-1a and 2.14-1b illustrate the ascent characteristics of the 431. Figures 2.14-2 and 2.14-3 and Tables 2.14-1 and 2.14-2 illustrate the performance of the Atlas V 431 vehicle to geotransfer orbits as described in Sections 2.7.1 and 2.7.2. Figure 2.14-4 depicts the minimum delta-velocity to geosynchronous orbit with various apogee caps. Figure 2.14-6 shows escape capability without the Star 48V solid motor.

2.15–2.20 ATLAS V 501-551 PERFORMANCE DATA

Figures 2.15-1a through 2.20-1b illustrate the ascent characteristics of the 501 through 551 respectively. Figures 2.15-2a through 2.15-3b and Tables 2.15-1, 2.15-2a and 2.15-2b illustrate the performance of the Atlas V 501 through 551 vehicles to geotransfer orbits as described in Sections 2.7.1 and 2.7.2. Figure 2.15-5 depicts the 521-551 minimum delta-velocity to geosynchronous orbit using three burns. Data is not available for the 501 and 511 configurations due to lack of adequate helium pressurant for the third burn. Figures 2.15-6 through 2.20-6 show Earth escape (C3) performance for the 501 through 551 vehicles with and without the Star 48V orbit insertion stage (OIS). Figures 2.18-7 and 2.18-8 show LEO and intermediate circular orbit performance for the 531 configuration.

2.21–2.26 ATLAS V 532 AND 552 PERFORMANCE DATA

Figures 2.24-7, 2.24-8, 2.26-7 and 2.26-8 illustrate the performance of the Atlas V 532 and 552 vehicles to LEO and intermediate circular orbits as described in Sections 2.7.4 and 2.7.5.

The following sections contain the figures and tables described herein and are arranged by Mission Category. We hope this is more convenient for the user.

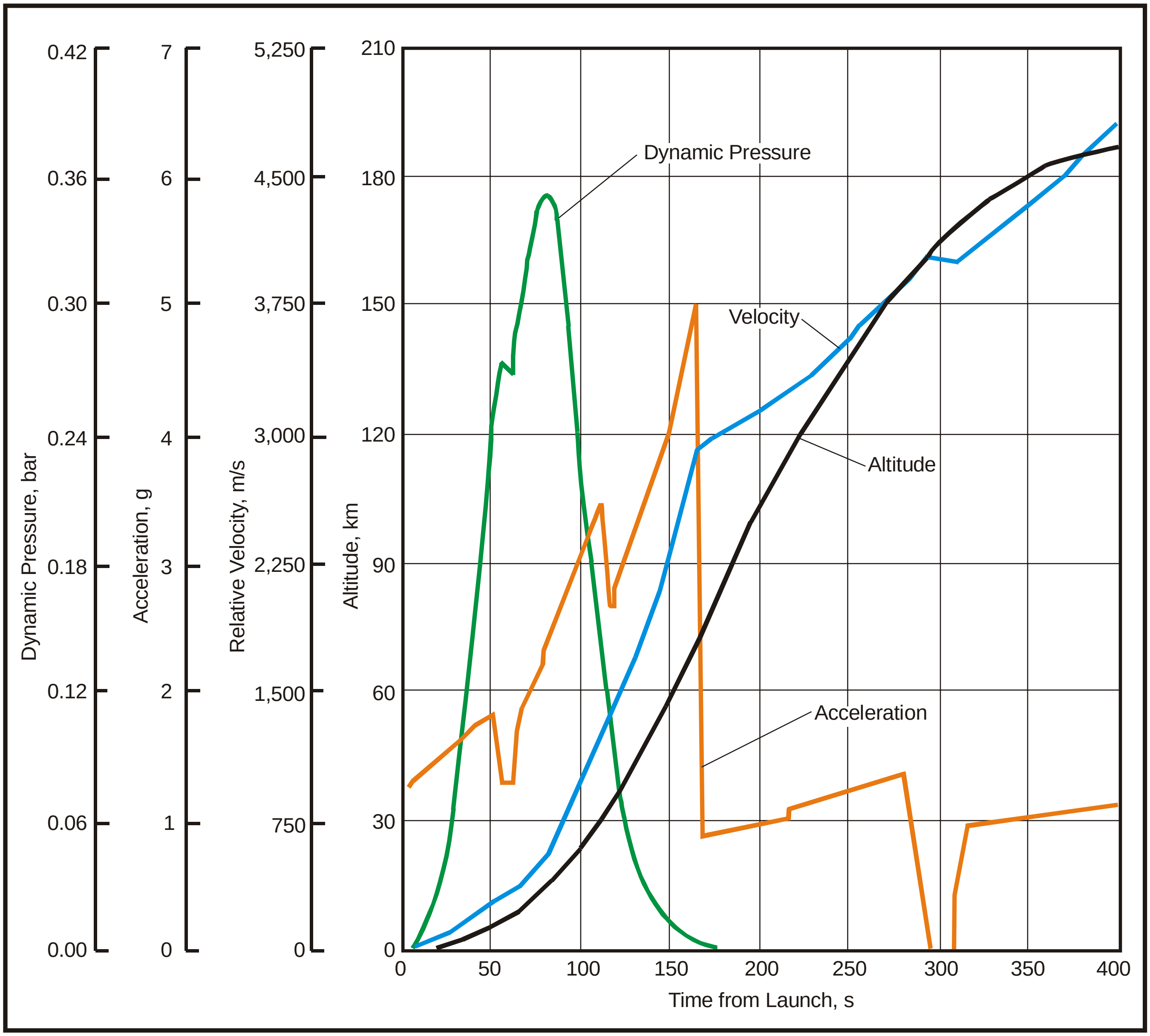


Figure 2.7-1 Atlas IIAS Nominal Ascent Data

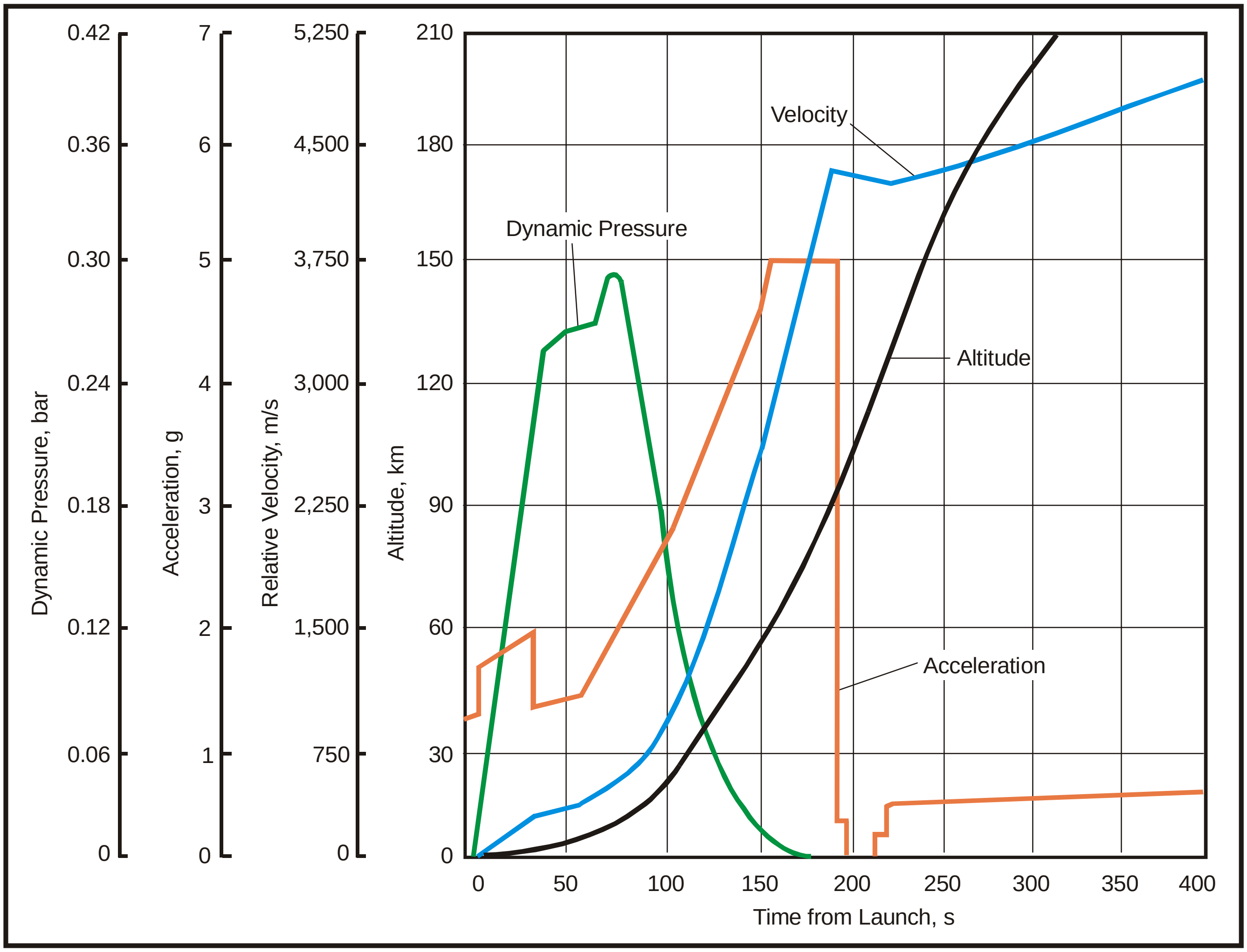


Figure 2.8-1 Atlas IIIA Nominal Ascent Data

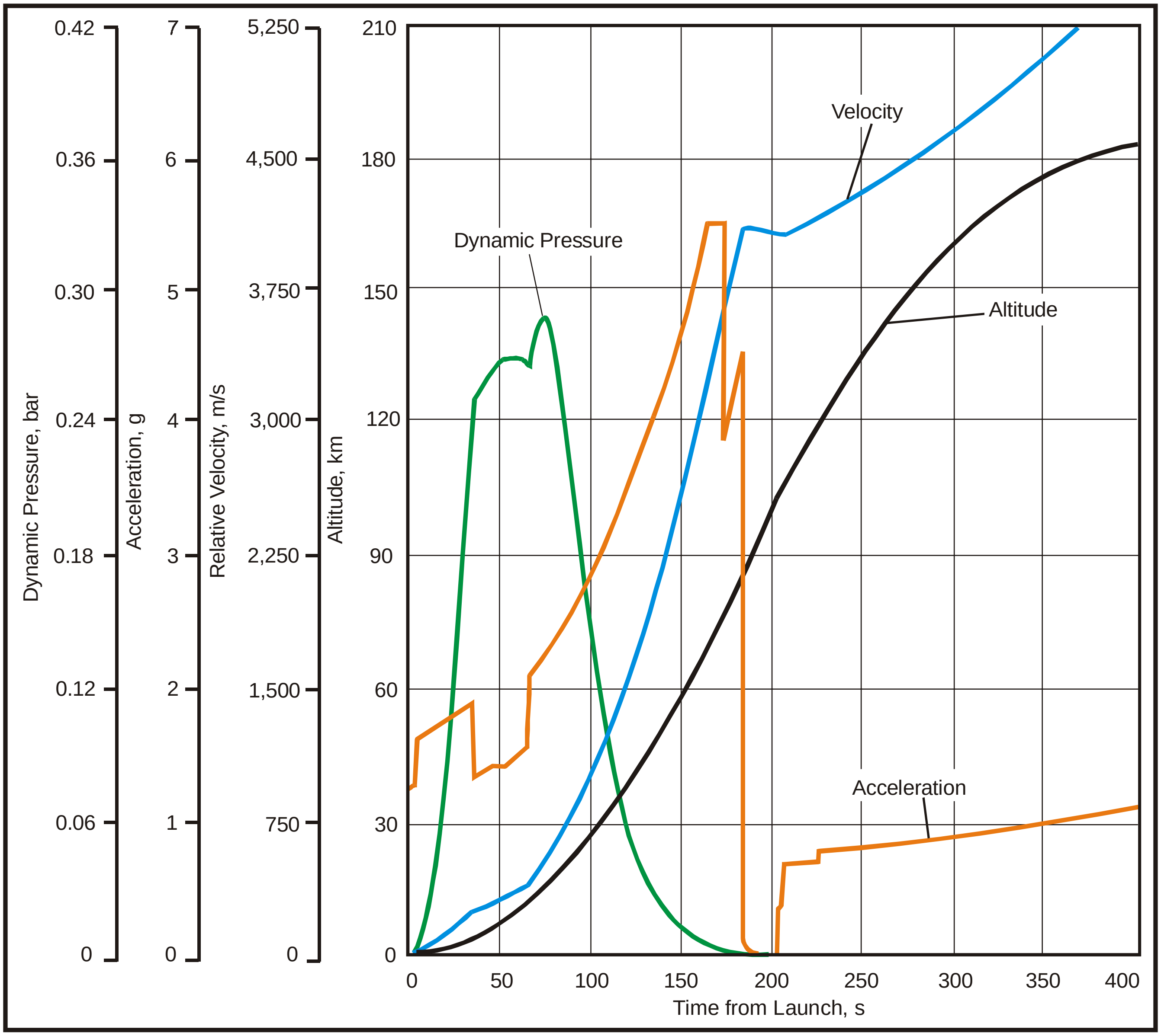


Figure 2.9-1 Atlas IIIB (DEC) Nominal Ascent Data

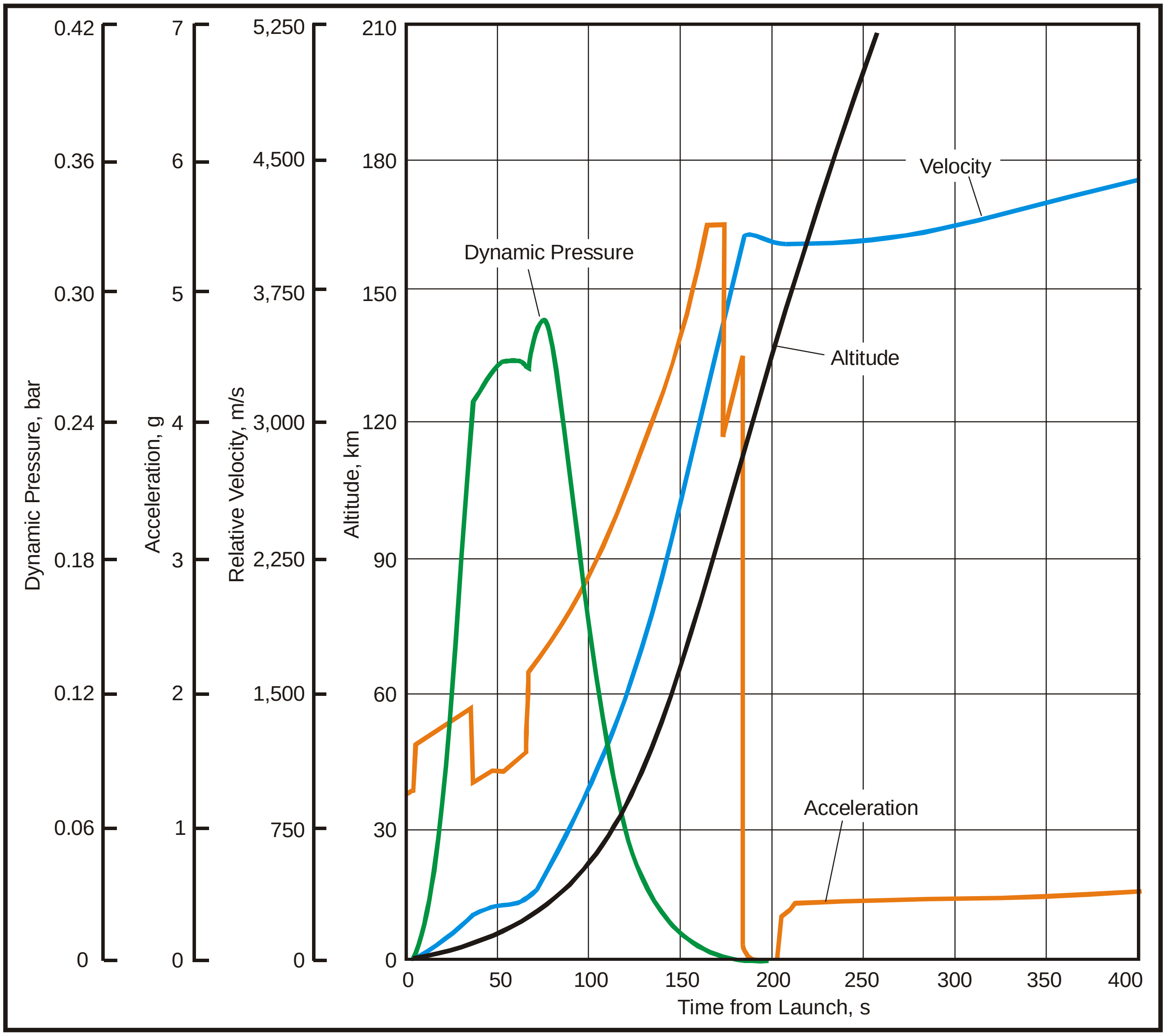


Figure 2.10-1 Atlas IIIB (SEC) Nominal Ascent Data

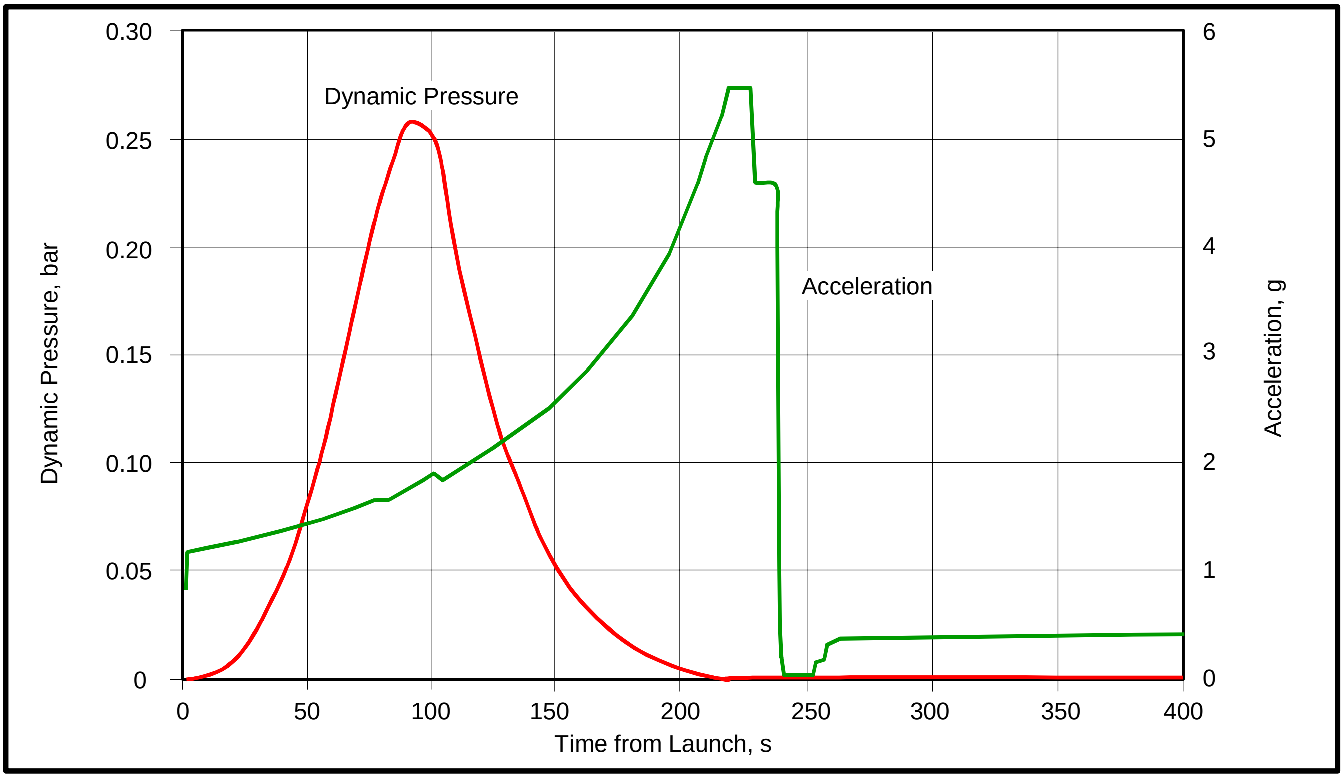


Figure 2.11-1a Atlas V 401 Nominal Ascent Data

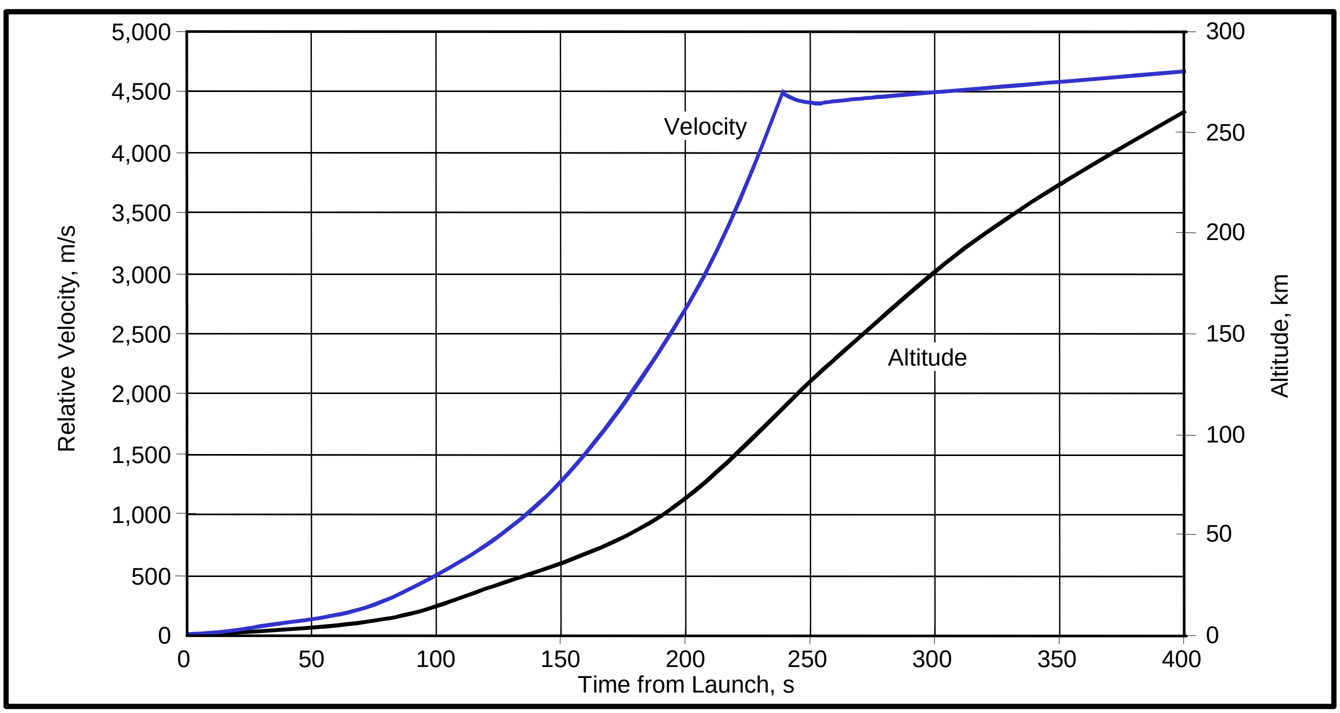


Figure 2.11-1b Atlas V 401 Nominal Ascent Data

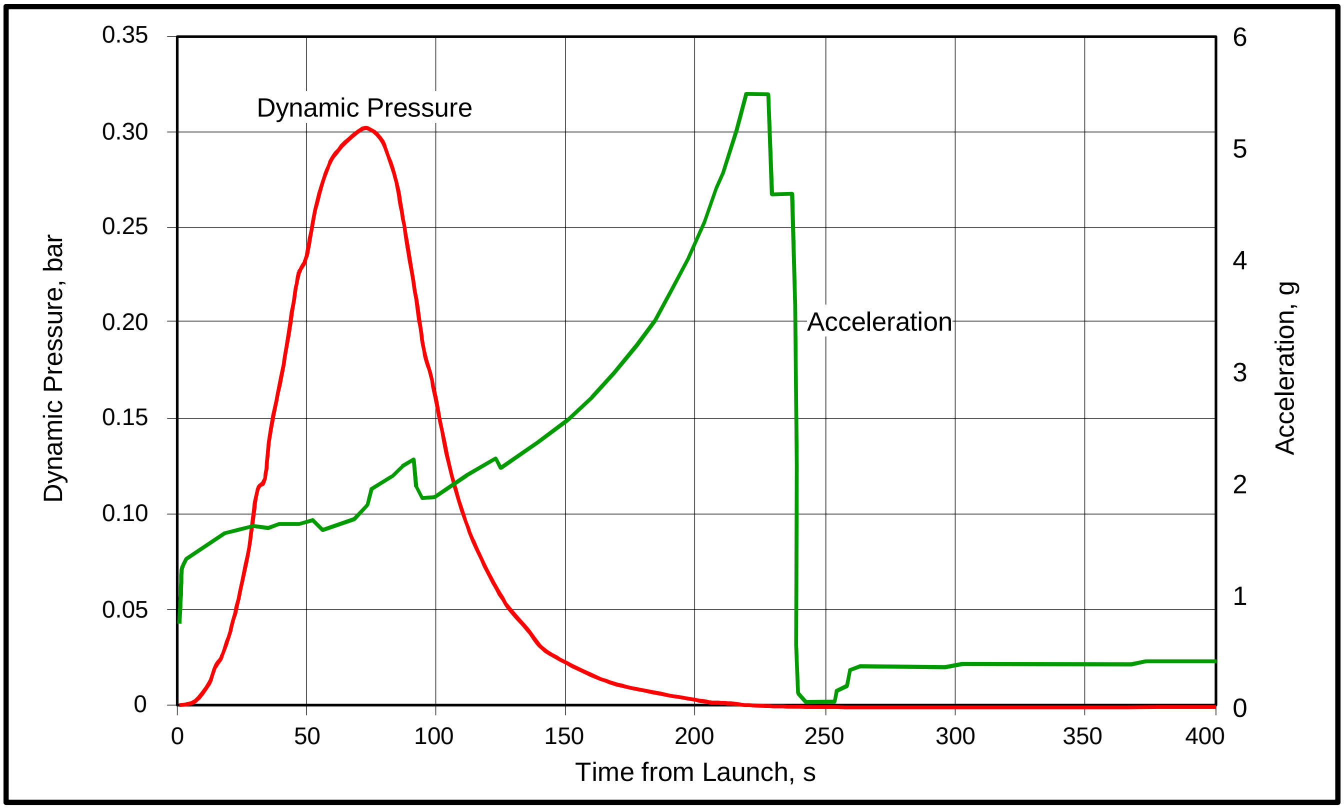


Figure 2.12-1a Atlas V 411 Nominal Ascent Data

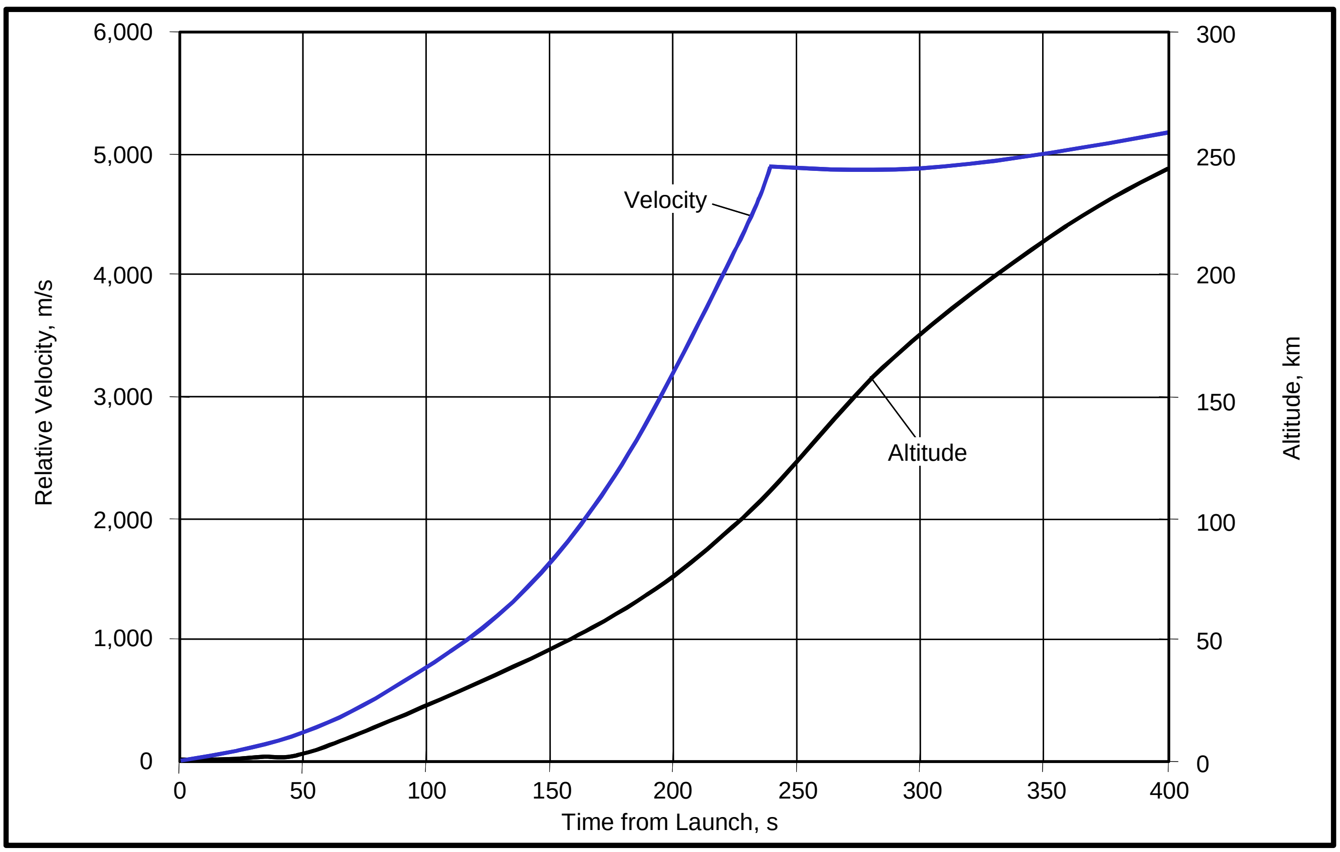


Figure 2.12-1b Atlas V 411 Nominal Ascent Data

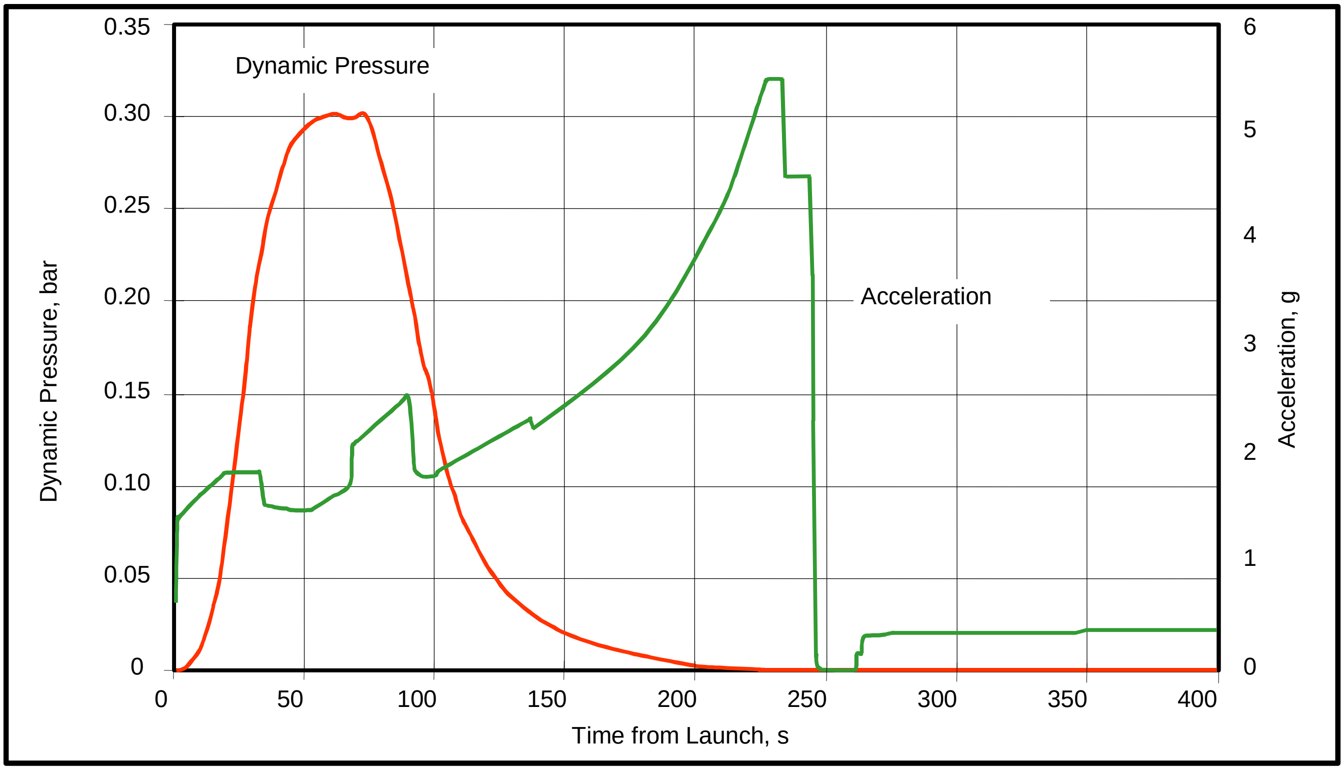


Figure 2.13-1a Atlas V 421 Nominal Ascent Data

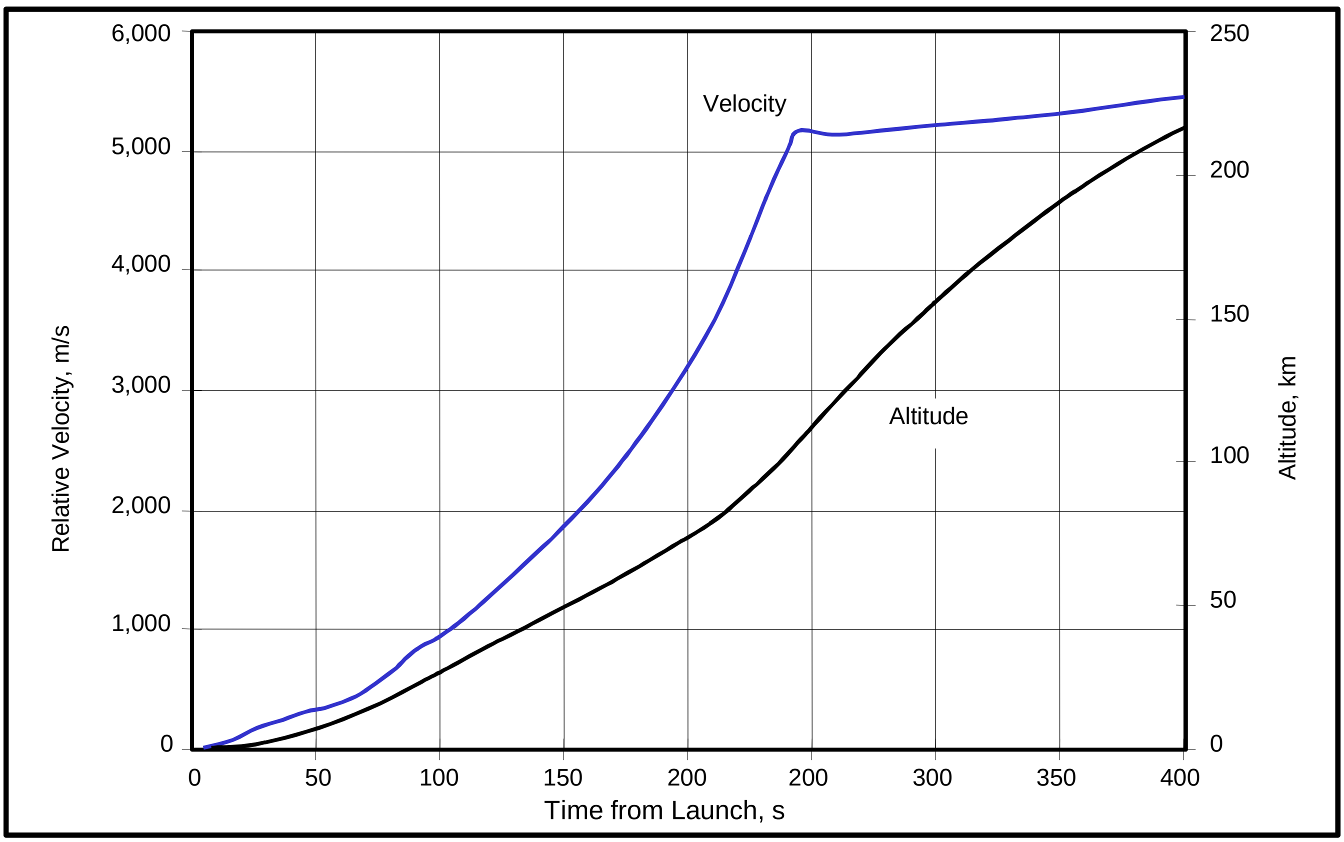


Figure 2.13-1b Atlas V 421 Nominal Ascent Data

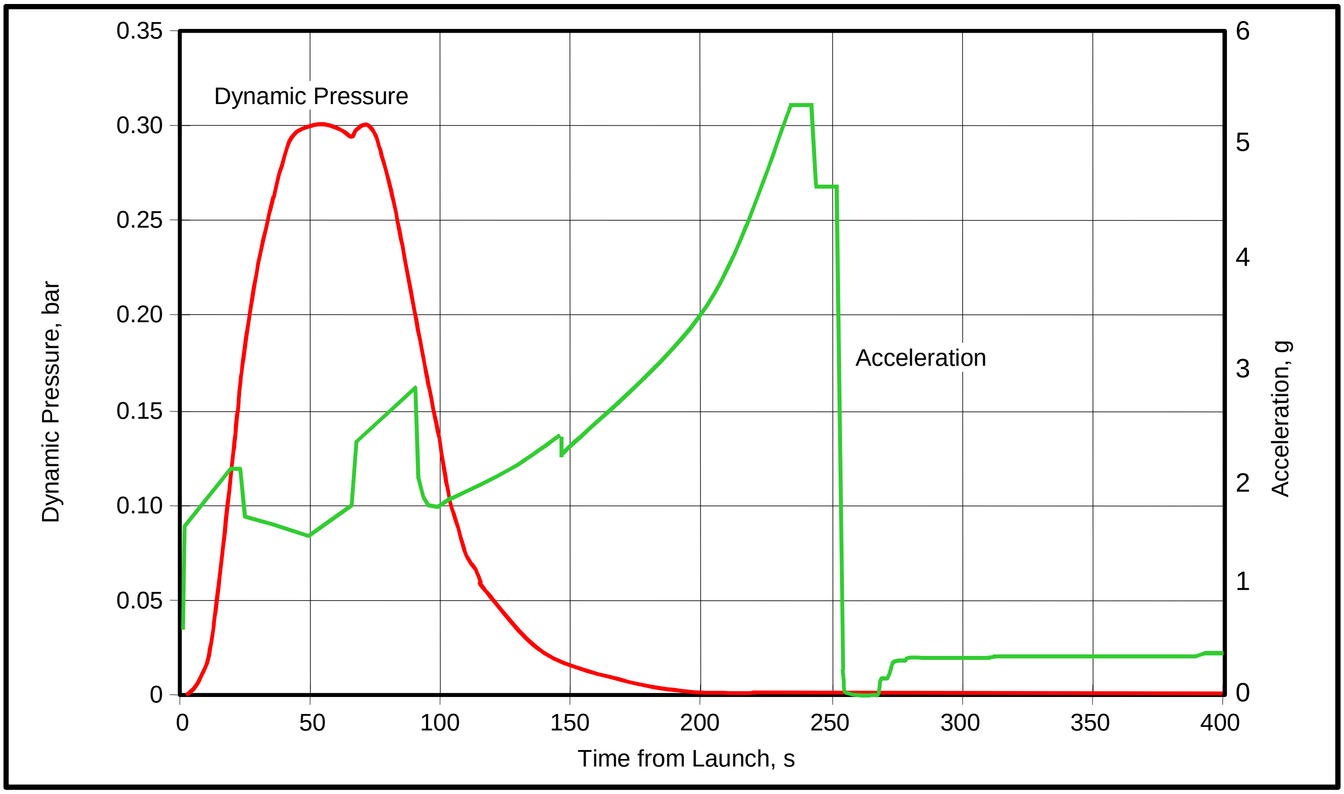


Figure 2.14-1a Atlas V 431 Nominal Ascent Data

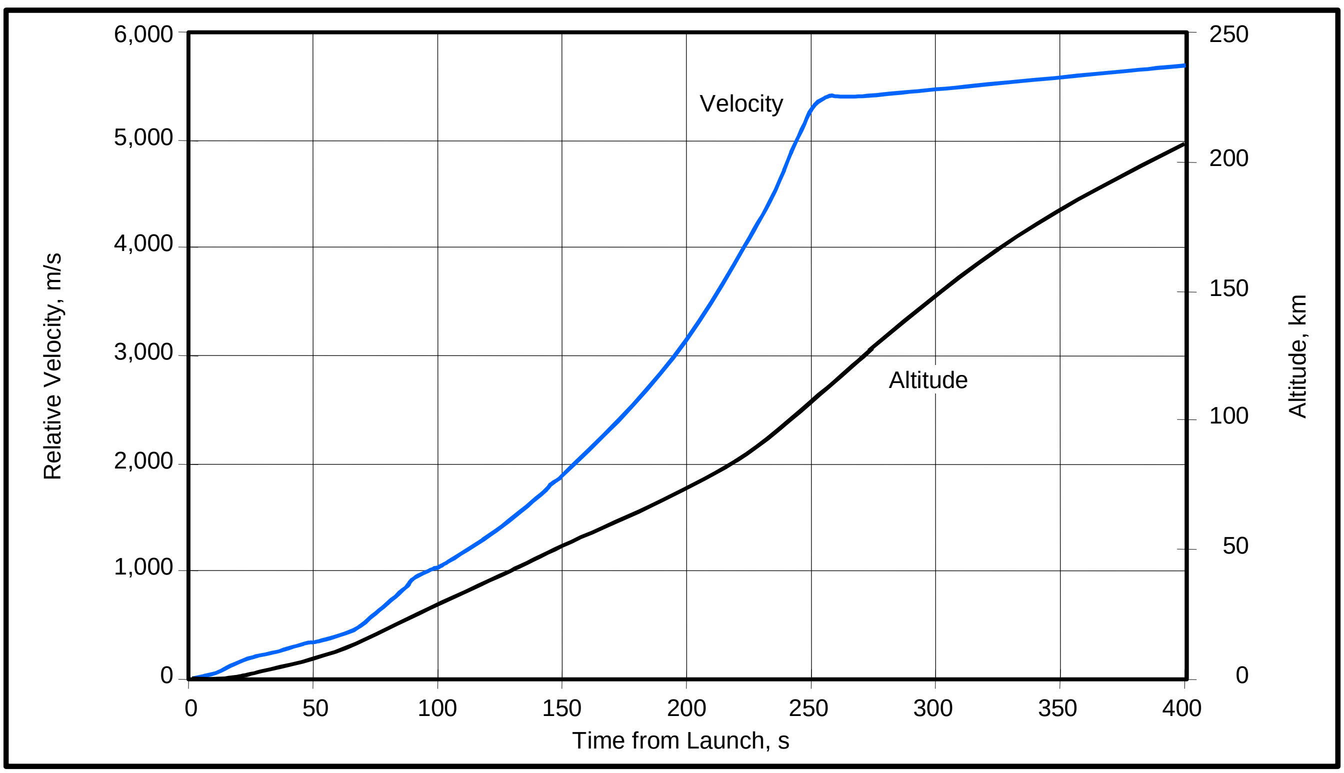


Figure 2.14-1b Atlas V 431 Nominal Ascent Data

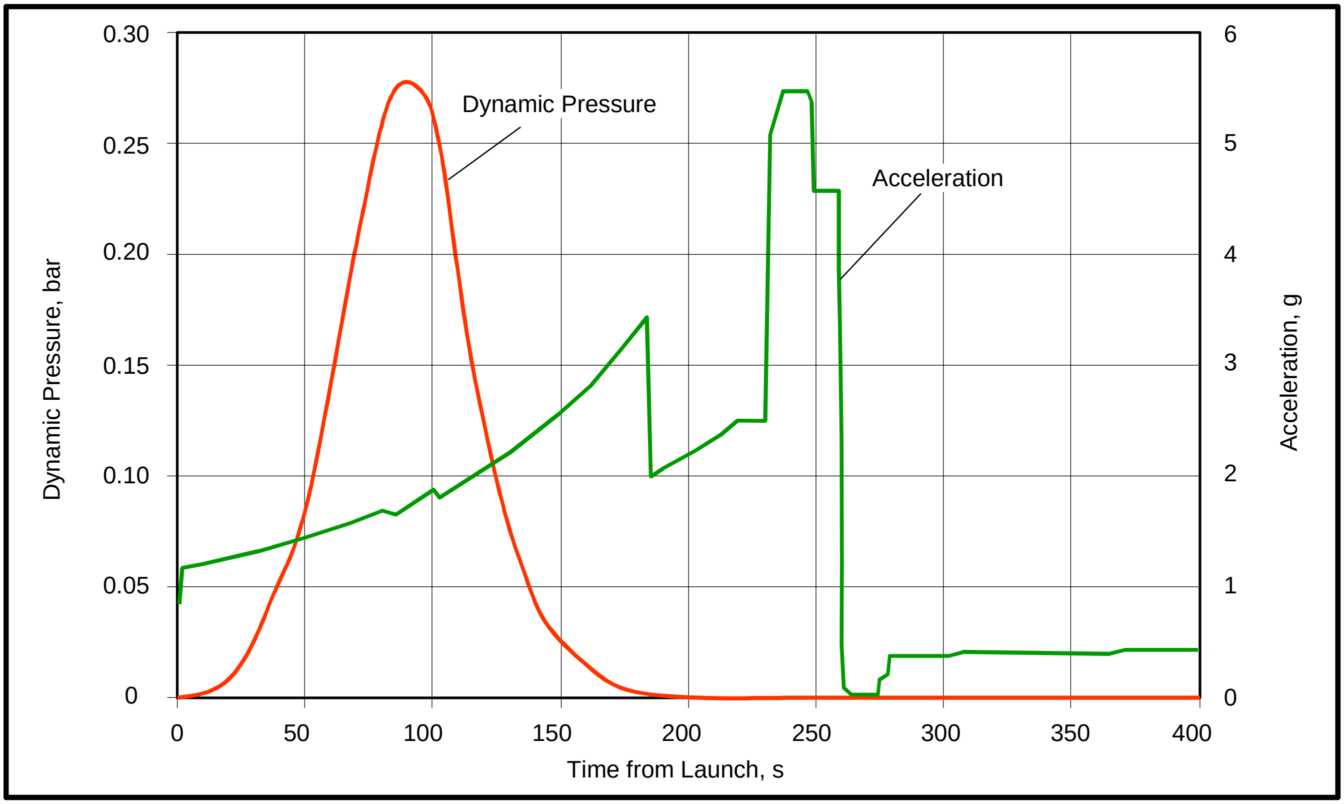


Figure 2.15-1a Atlas V 501 Nominal Ascent Data

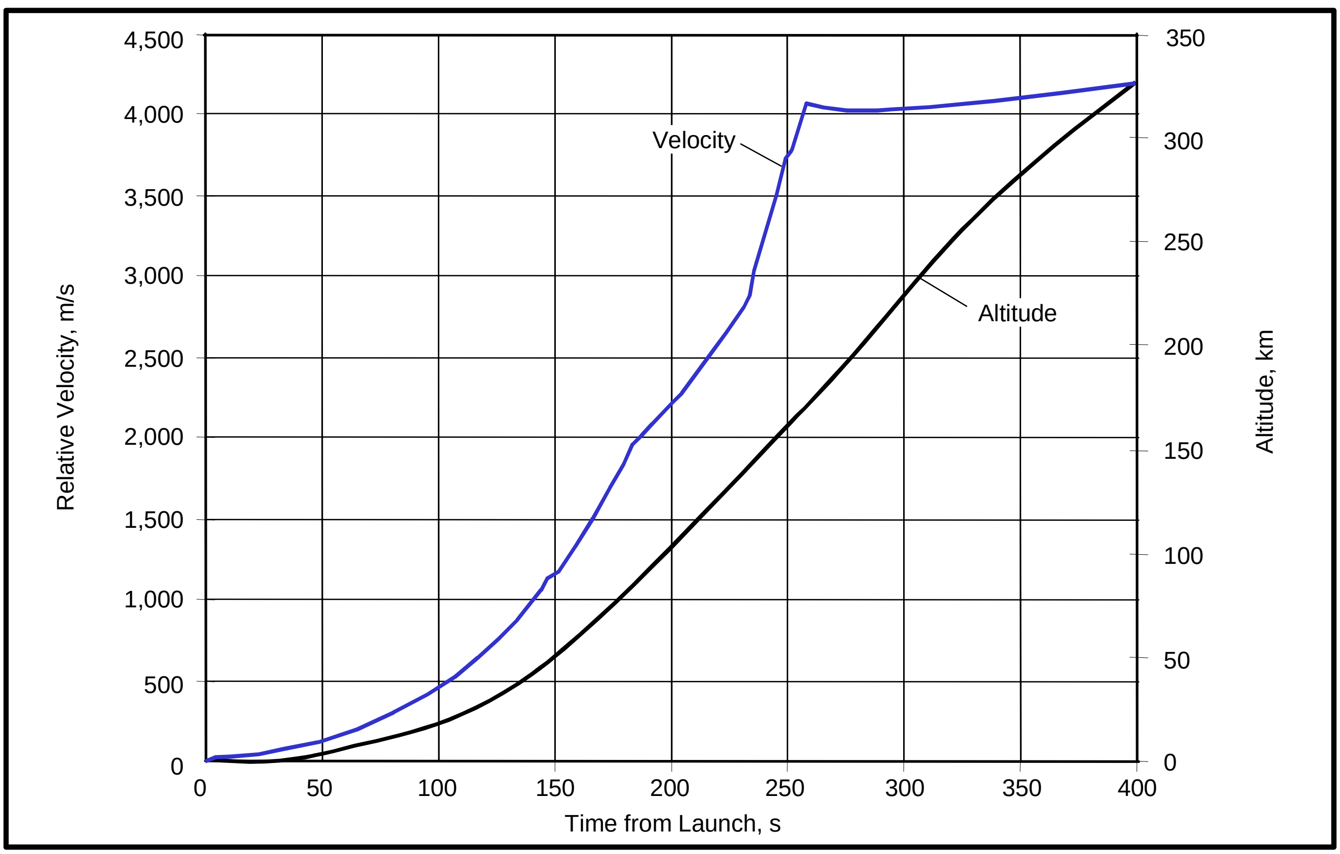


Figure 2.15-1b Atlas V 501 Nominal Ascent Data

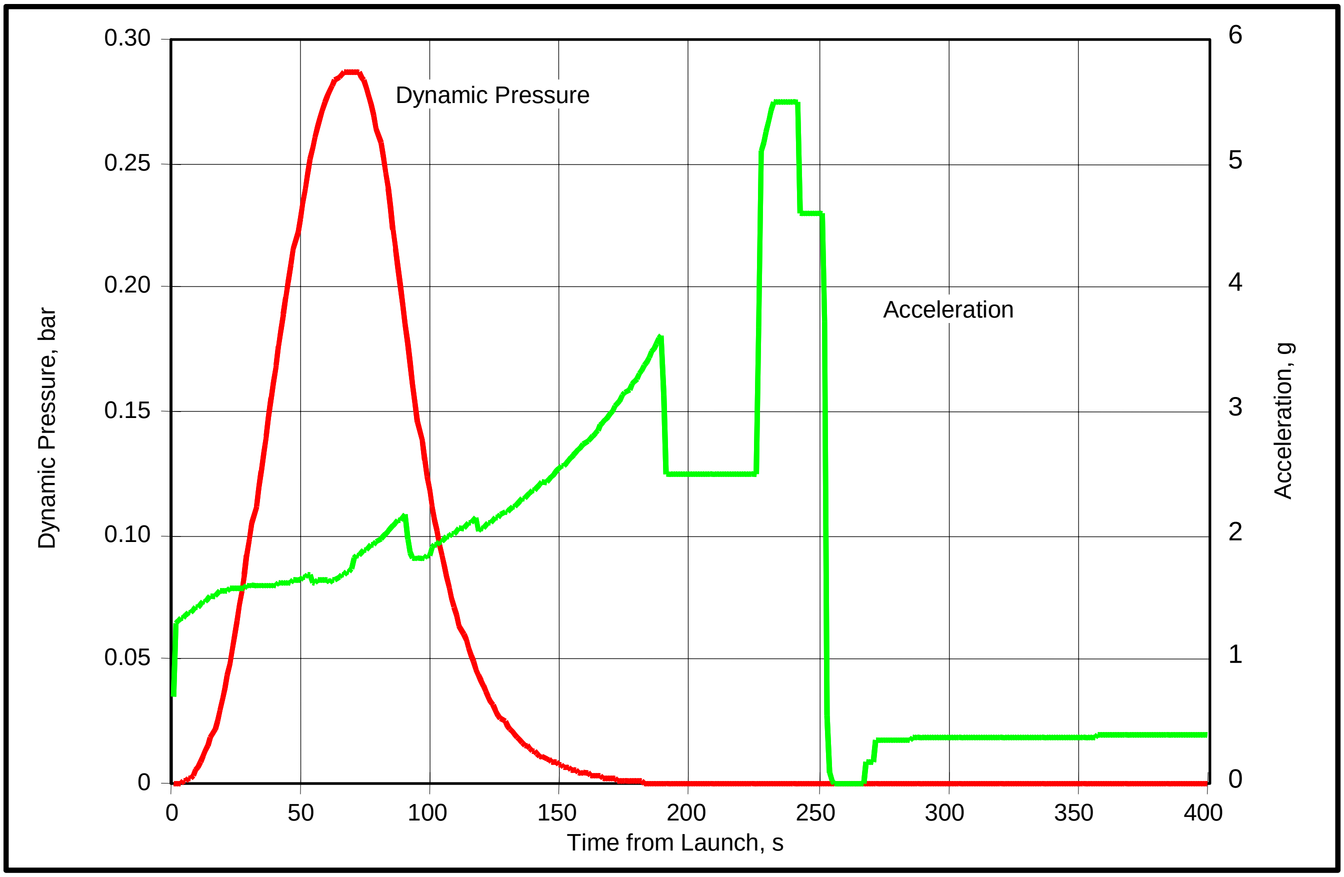


Figure 2.16-1a Atlas V 511 Nominal Ascent Data

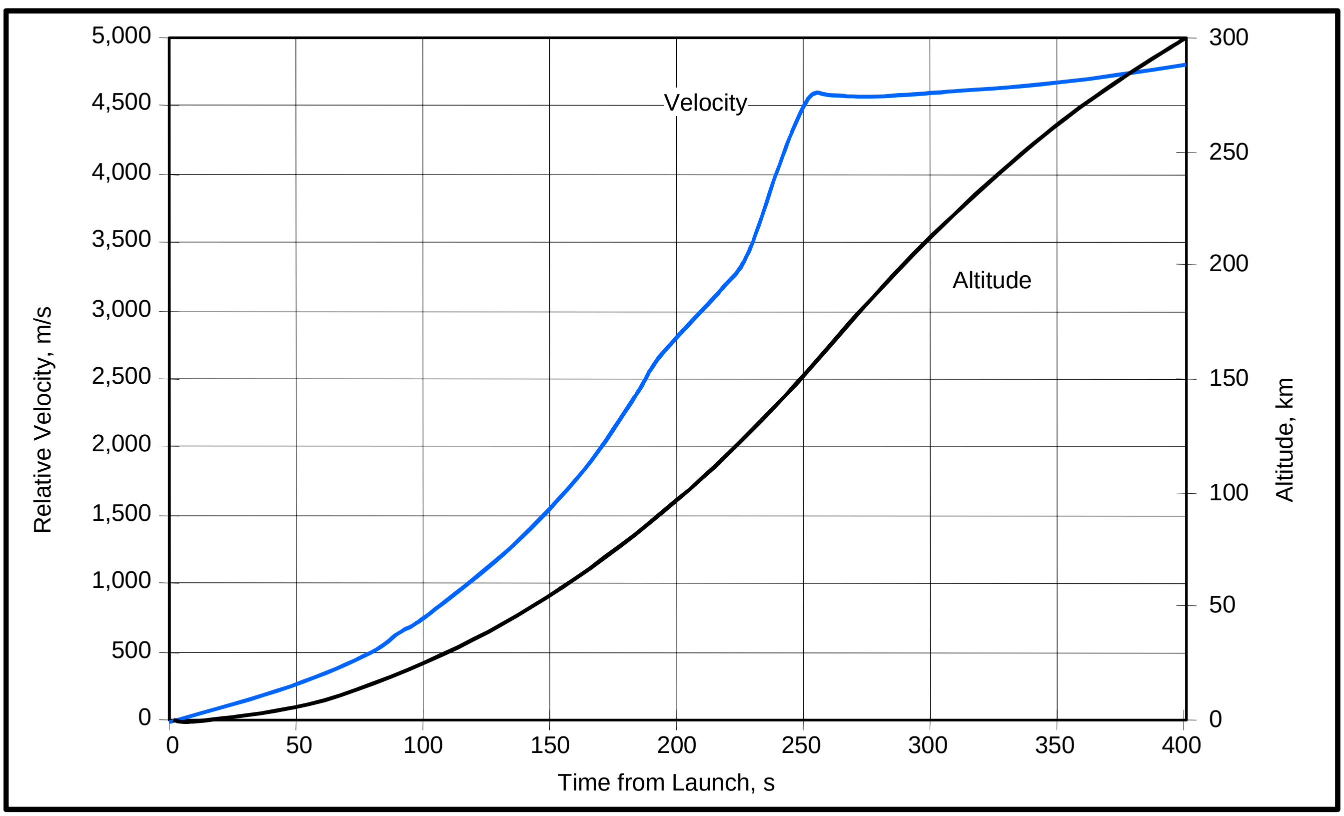


Figure 2.16-1b Atlas V 511 Nominal Ascent Data

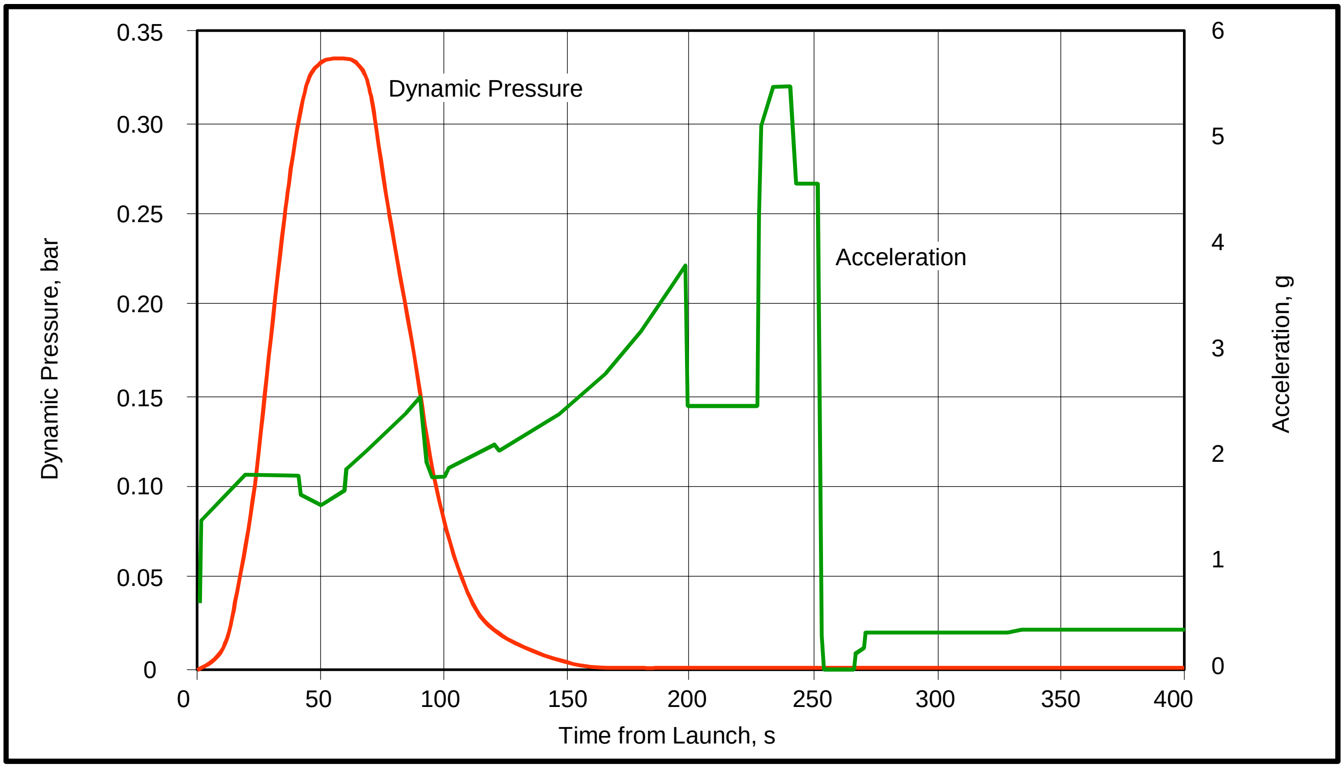


Figure 2.17-1a Atlas V 521 Nominal Ascent Data

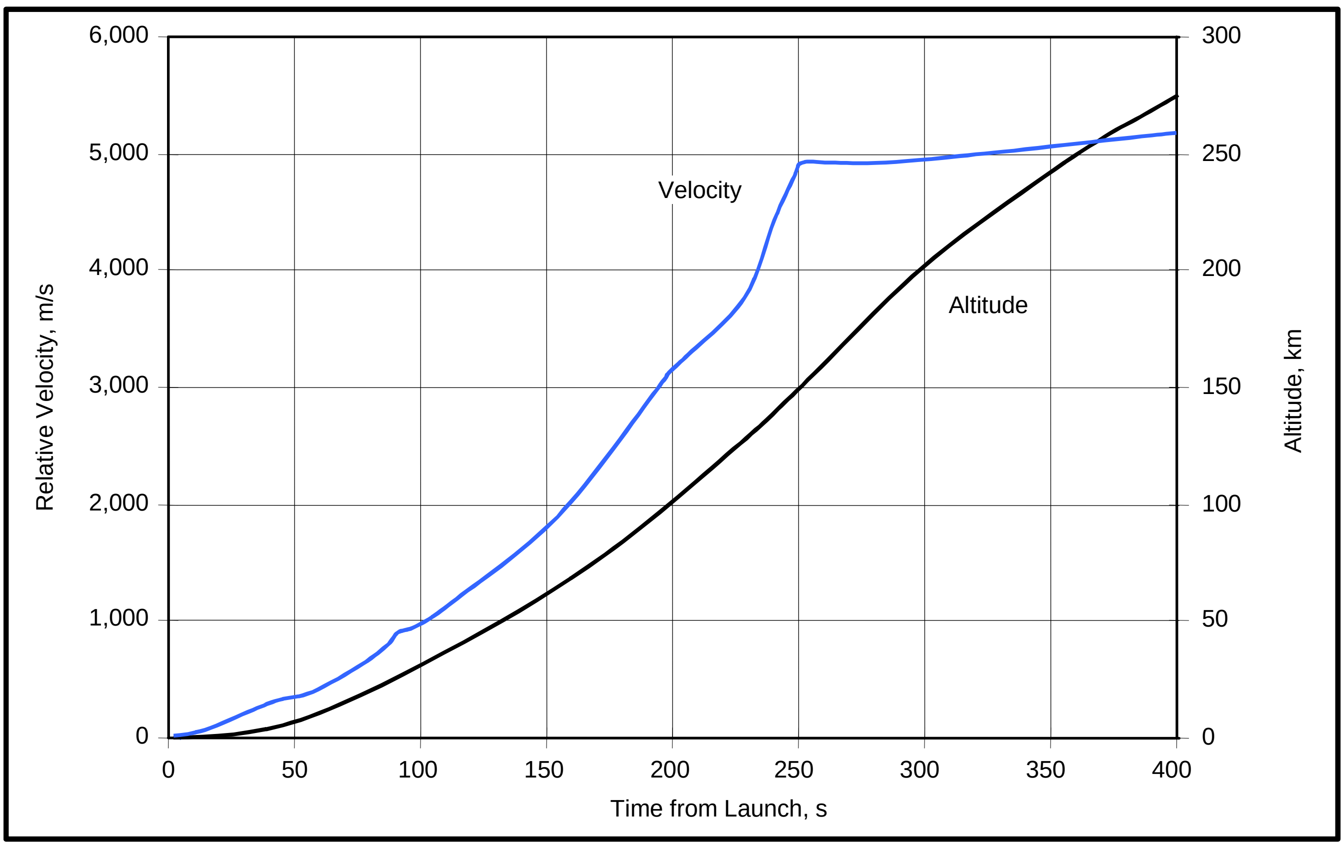


Figure 2.17-1b Atlas V 521 Nominal Ascent Data

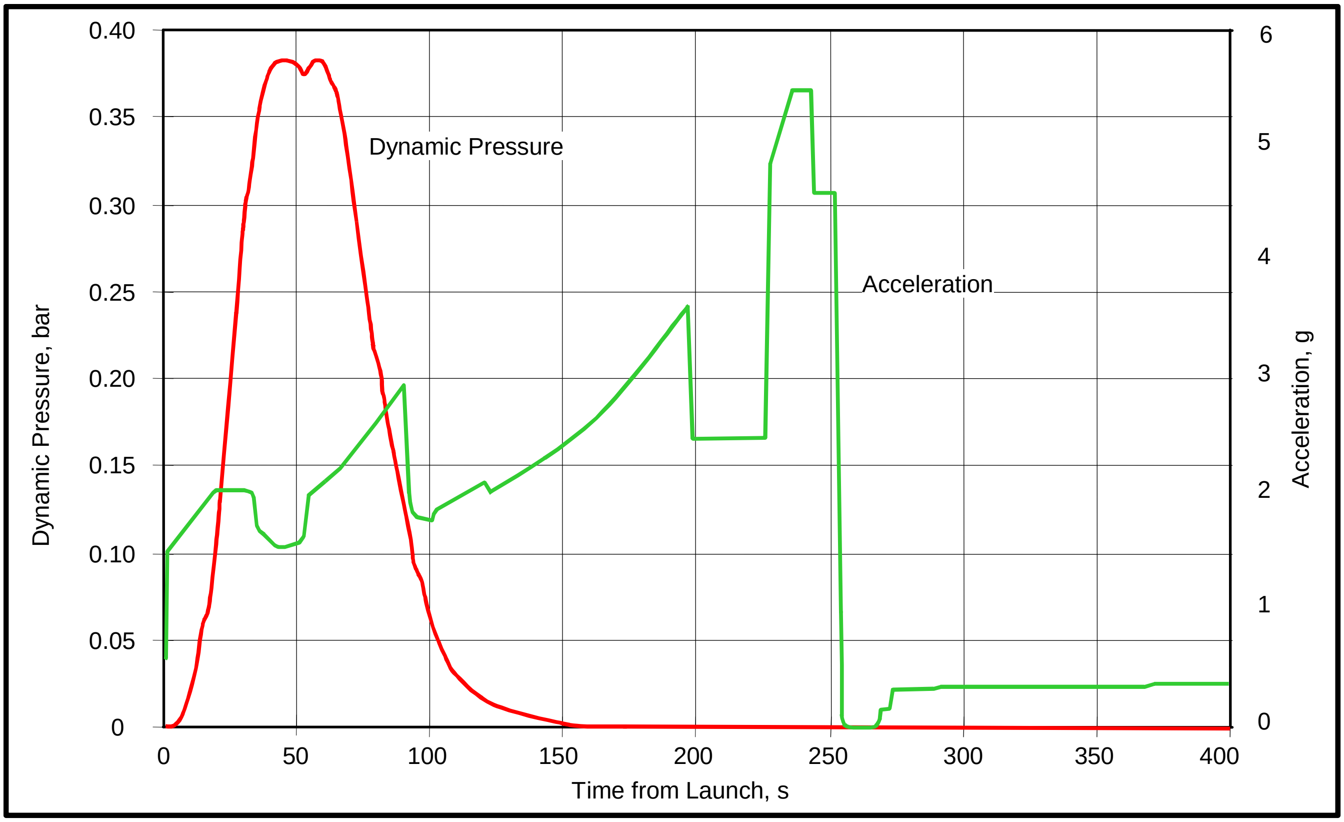


Figure 2.18-1a Atlas V 531 Nominal Ascent Data

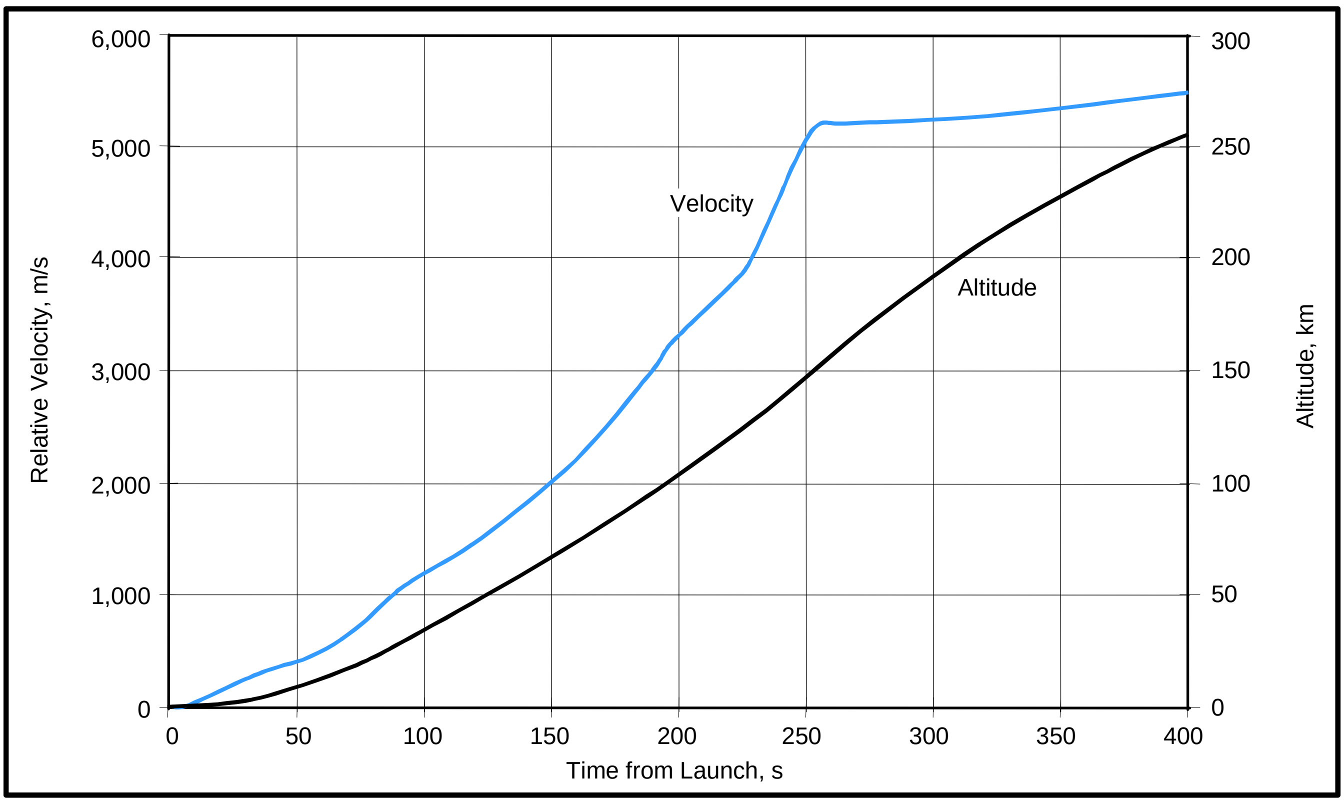


Figure 2.18-1b Atlas V 531 Nominal Ascent Data

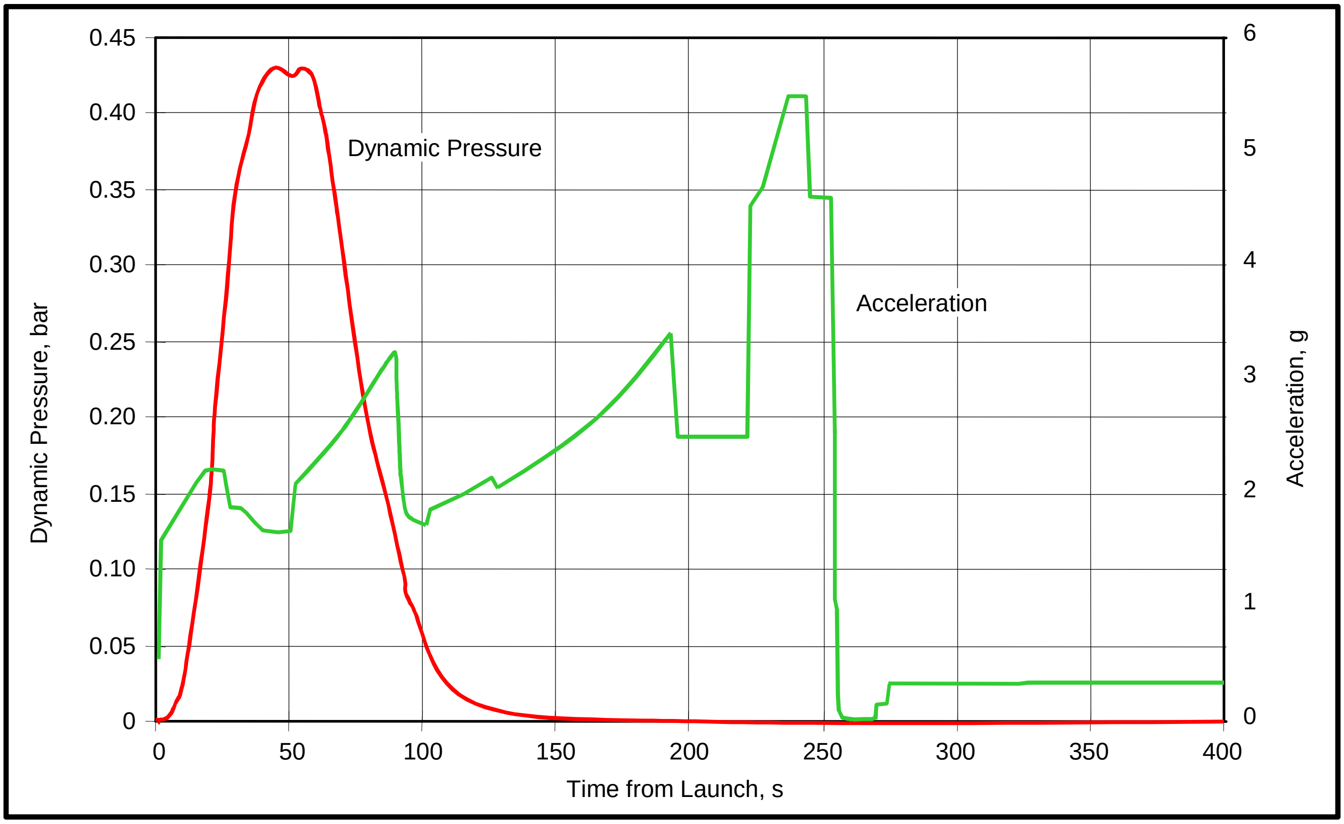


Figure 2.19-1a Atlas V 541 Nominal Ascent Data

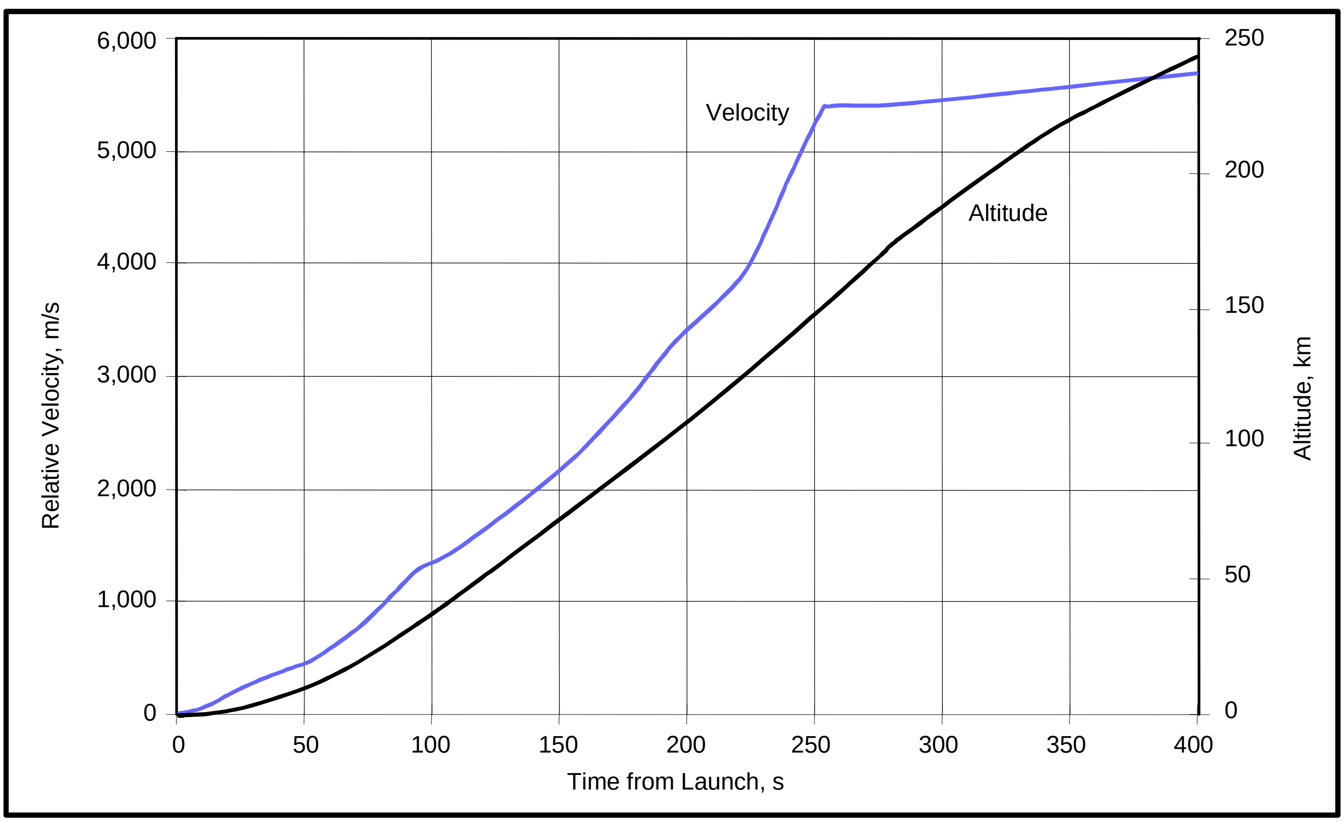


Figure 2.19-1b Atlas V 541 Nominal Ascent Data

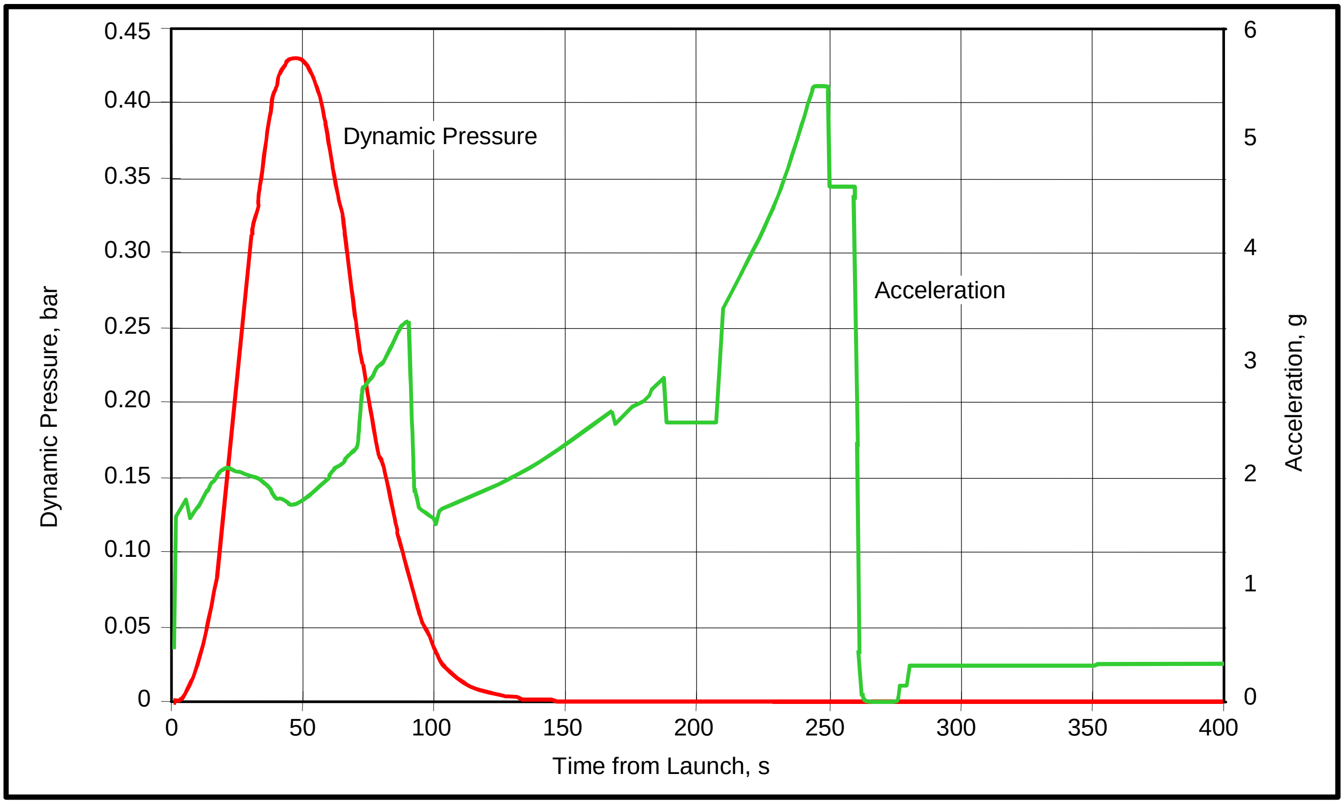


Figure 2.20-1a Atlas V 551 Nominal Ascent Data

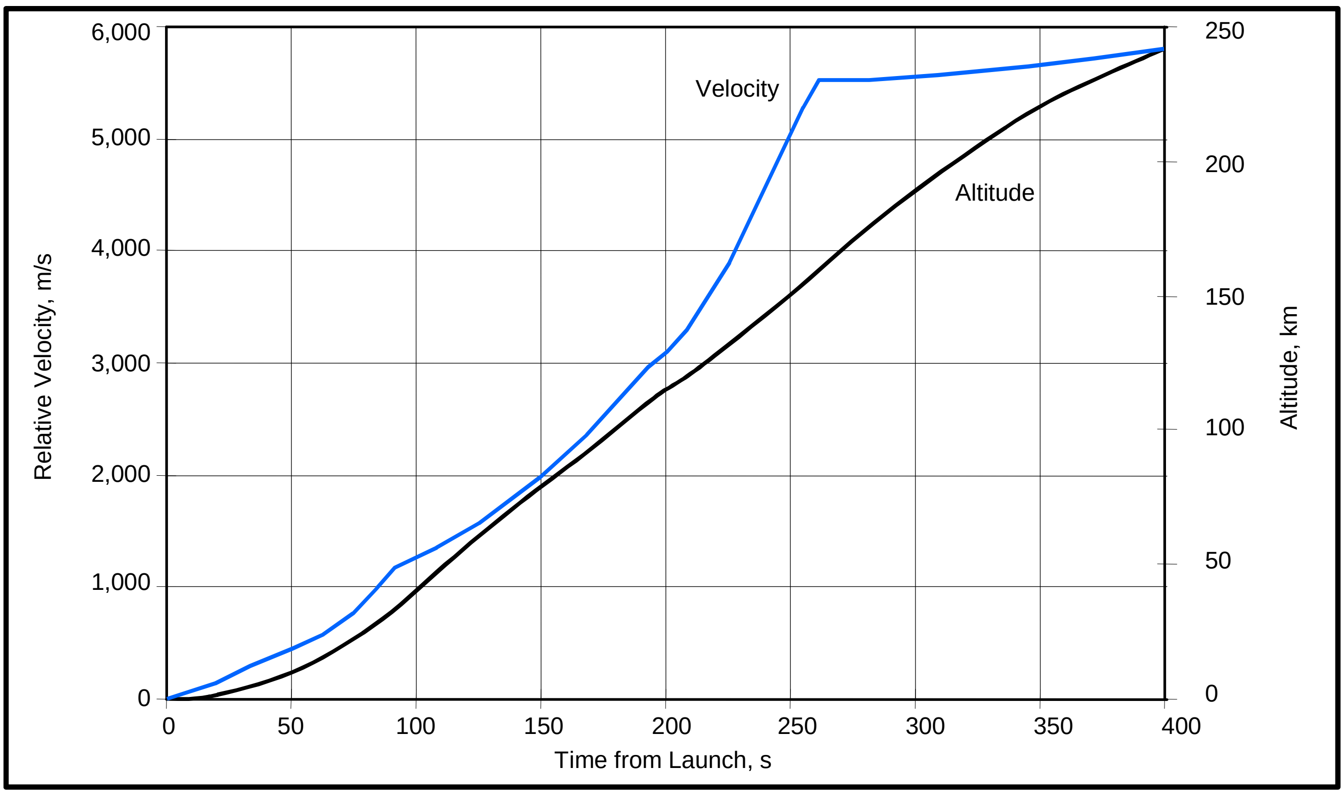


Figure 2.20-1b Atlas V 551 Nominal Ascent Data

Table 2.7-1 Atlas IIAS Geotransfer Orbit Performance—PSW vs Apogee Altitude

Apogee Altitude		Payload Systems Weight, kg (lb) Atlas IIAS			
km	(nmi)	MRS		GCS	
150,000	(80,993.5)	3,055	(6,735)	2,987	(6,586)
140,000	(75,593.9)	3,074	(6,777)	3,006	(6,627)
130,000	(70,194.4)	3,095	(6,824)	3,027	(6,673)
120,000	(64,794.8)	3,120	(6,879)	3,052	(6,728)
110,000	(59,395.2)	3,150	(6,944)	3,081	(6,792)
100,000	(53,995.7)	3,184	(7,020)	3,114	(6,867)
95,000	(51,295.9)	3,204	(7,065)	3,134	(6,910)
90,000	(48,596.1)	3,226	(7,113)	3,156	(6,958)
85,000	(45,896.3)	3,251	(7,167)	3,180	(7,011)
80,000	(43,196.5)	3,278	(7,228)	3,207	(7,071)
75,000	(40,496.8)	3,308	(7,294)	3,237	(7,136)
70,000	(37,797.0)	3,343	(7,371)	3,271	(7,212)
65,000	(35,097.2)	3,383	(7,458)	3,310	(7,297)
60,000	(32,397.4)	3,428	(7,559)	3,355	(7,396)
55,000	(29,697.6)	3,481	(7,674)	3,406	(7,510)
52,500	(28,347.7)	3,511	(7,740)	3,436	(7,575)
50,000	(26,997.8)	3,543	(7,811)	3,467	(7,644)
47,500	(25,647.9)	3,578	(7,889)	3,502	(7,721)
45,000	(24,298.0)	3,617	(7,974)	3,540	(7,805)
42,500	(22,948.2)	3,660	(8,069)	3,582	(7,898)
40,000	(21,598.3)	3,707	(8,173)	3,629	(7,000)
37,500	(20,248.4)	3,759	(8,288)	3,680	(8,114)
35,786	**(19,322.9)**	**3,799**	**(8,376)**	**3,719**	**(8,200)**
35,000	(18,898.5)	3,818	(8,418)	3,738	(8,241)
32,500	(17,548.6)	3,885	(8,565)	3,803	(8,386)
30,000	(16,198.7)	3,960	(8,731)	3,877	(8,549)
27,500	(14,848.8)	4,046	(8,921)	3,962	(8,736)
25,000	(13,498.9)	4,147	(9,142)	4,061	(8,954)
22,500	(12,149.0)	4,263	(9,400)	4,176	(9,207)
20,000	(10,799.1)	4,402	(9,706)	4,313	(9,508)
17,500	(9,449.2)	4,570	(10,075)	4,477	(9,872)
15,000	(8,099.4)	4,775	(10,528)	4,680	(10,317)
12,500	(6,749.5)	5,034	(11,100)	4,934	(10,879)
11,000	(5,939.5)	5,224	(11,517)	5,121	(11,290)
10,000	(5,399.6)	5,369	(11,837)	5,263	(11,605)
9,000	(4,859.6)	5,532	(12,197)	5,424	(11,959)
8,000	(4,319.7)	5,717	(12,605)	5,606	(12,360)
7,000	(3,779.7)	5,928	(13,070)	5,813	(12,817)
6,000	(3,239.7)	6,172	(13,607)	6,053	(13,346)
5,000	(2,699.8)	6,455	(14,232)	6,332	(13,961)

Note: Large (4.2-m) Diameter Payload Fairing Jettison at 3-sigma qv<1,135 W/m^2 (360 Btu/ft^2-hr);
Parking Orbit Perigee Altitude = 148.2 km (80 nmi);
Transfer Orbit Perigee Altitude = 166.7 (90 nmi);
Transfer Orbit Inclination = 27.0°;
Argument of Perigee = 180°

Table 2.8-1 Atlas IIIA Geotransfer Orbit Performance—PSW vs Apogee Altitude

Apogee Altitude		Payload Systems Weight, kg (lb) Atlas IIIA			
km	(nmi)	MRS		GCS	
150,000	(80,993.5)	3,335	(7,352)	3,272	(7,214)
145,000	(78,293.7)	3,344	(7,373)	3,281	(7,234)
140,000	(75,594.0)	3,355	(7,396)	3,292	(7,257)
135,000	(72,894.2)	3,366	(7,420)	3,302	(7,280)
130,000	(70,194.4)	3,377	(7,446)	3,314	(7,306)
125,000	(67,494.6)	3,390	(7,474)	3,326	(7,333)
120,000	(64,794.8)	3,403	(7,503)	3,339	(7,362)
115,000	(62,095.0)	3,418	(7,536)	3,354	(7,394)
110,000	(59,395.2)	3,434	(7,571)	3,370	(7,429)
105,000	(56,695.5)	3,451	(7,609)	3,387	(7,467)
100,000	(53,995.7)	3,470	(7,651)	3,406	(7,508)
95,000	(51,295.9)	3,490	(7,694)	3,426	(7,553)
90,000	(48,596.1)	3,514	(7,748)	3,449	(7,603)
85,000	(45,896.3)	3,540	(7,805)	3,474	(7,659)
80,000	(43,196.5)	3,568	(7,867)	3,502	(7,721)
75,000	(40,496.8)	3,601	(7,938)	3,533	(7,790)
70,000	(37,797.0)	3,637	(8,018)	3,569	(7,869)
65,000	(35,097.2)	3,678	(8,109)	3,610	(7,958)
60,000	(32,397.4)	3,725	(8,213)	3,656	(8,061)
55,000	(29,697.6)	3,780	(8,334)	3,711	(8,181)
50,000	(26,997.8)	3,845	(8,477)	3,774	(8,321)
45,000	(24,298.1)	3,922	(8,647)	3,851	(8,489)
40,000	(21,598.3)	4,016	(8,854)	3,943	(8,692)
35,786	**(19,322.9)**	**4,112**	**(9,065)**	**4,037**	**(8,900)**
35,000	(18,898.5)	4,132	(9,109)	4,057	(8,944)
30,000	(16,198.7)	4,279	(9,434)	4,202	(9,264)
25,000	(13,498.9)	4,472	(9,860)	4,392	(9,683)
20,000	(10,799.1)	4,737	(10,444)	4,653	(10,258)
15,000	(8,099.4)	5,122	(11,291)	5,031	(11,092)
10,000	(5,399.6)	5,727	(12,626)	5,628	(12,407)
5,000	(2,699.8)	6,809	(15,011)	6,696	(14,762)

Note: Extended Payload Fairing Jettison at 3-sigma qv<1,135 W/m^2 (360 Btu/ft^2-hr);
Parking Orbit Perigee Altitude $\geq$ 148.2 km (80 nmi);
Transfer Orbit Perigee Altitude $\geq$ 166.7 (90 nmi);
Orbit Inclination = 27.0°;
Argument of Perigee = 180°

Table 2.9-1 Atlas IIIB (DEC) Geotransfer Orbit Performance—PSW vs Apogee Altitude

Apogee Altitude		Payload Systems Weight, kg (lb) Atlas IIIB (DEC)			
km	(nmi)	MRS		GCS	
150,000	(80,993.5)	3,697	(8,150)	3,603	(7,944)
145,000	(78,293.7)	3,708	(8,174)	3,614	(7,968)
140,000	(75,594.0)	3,720	(8,201)	3,626	(7,994)
135,000	(72,894.2)	3,732	(8,229)	3,639	(8,022)
130,000	(70,194.4)	3,746	(8,259)	3,652	(8,051)
125,000	(67,494.6)	3,761	(8,291)	3,667	(8,083)
120,000	(64,794.8)	3,777	(8,326)	3,682	(8,118)
115,000	(62,095.0)	3,794	(8,364)	3,699	(8.155)
110,000	(59,395.2)	3,813	(8,405)	3,717	(8,195)
105,000	(56,695.5)	3,833	(8,450)	3,737	(8,239)
100,000	(53,995.7)	3,855	(8,499)	3,759	(8,288)
95,000	(51,295.9)	3,880	(8,553)	3,783	(8,340)
90,000	(48,596.1)	3,906	(8,612)	3,810	(8,399)
85,000	(45,896.3)	3,936	(8,678)	3,839	(8,464)
80,000	(43,196.5)	3,970	(8,752)	3,872	(8,536)
75,000	(40,496.8)	4,007	(8,834)	3,909	(8,617)
70,000	(37,797.0)	4,050	(8,928)	3,950	(8,709)
65,000	(35,097.2)	4,098	(9,034)	3,998	(8,814)
60,000	(32,397.4)	4,154	(9,157)	4,053	(8,934)
55,000	(29,697.6)	4,218	(9,299)	4,116	(9,074)
50,000	(26,997.8)	4,294	(9,467)	4,191	(9,239)
45,000	(24,298.1)	4,385	(9,668)	4,280	(9,436)
40,000	(21,598.3)	4,496	(9,911)	4,389	(9,675)
35,786	**(19,322.9)**	**4,609**	**(10,161)**	**4,500**	**(9,920)**
35,000	(18,898.5)	4,633	(10,213)	4,523	(9,972)
30,000	(16,198.7)	4,807	(10,597)	4,695	(10,350)
25,000	(13,498.9)	5,037	(11,104)	4,921	(10,848)
20,000	(10,799.1)	5,352	(11,800)	5,231	(11,533)
15,000	(8,099.4)	5,814	(12,817)	5,685	(12,532)
10,000	(5,399.6)	6,550	(14,440)	6,408	(14,127)
5,000	(2,699.8)	7,898	(17,413)	7,734	(17,051)

Note: Large Payload Fairing Jettison at 3-sigma qv<1,135 W/m^2 (360 Btu/ft^2-hr);
Parking Orbit Perigee Altitude ≥ 148.2 km (80 nmi);
Transfer Orbit Perigee Altitude ≥ 166.7 (90 nmi);
Orbit Inclination = 27.0°;
Argument of Perigee = 180°

Table 2.10-1 Atlas IIIB (SEC) Geotransfer Orbit Performance—PSW vs Apogee Altitude

Apogee Altitude		Payload System Weight, kg (lb) Atlas IIB (SEC)			
km	(nmi)	MRS		GCS	
150,000	(80,993.5)	3,430	(7,561)	3,365	(7,419)
145,000	(78,293.7)	3,439	(7,582)	3,374	(7,439)
140,000	(75,594.0)	3,449	(7,604)	3,384	(7,461)
135,000	(72,894.2)	3,460	(7,628)	3,395	(7,485)
130,000	(70,194.4)	3,472	(7,654)	3,406	(7,510)
125,000	(67,494.6)	3,484	(7,681)	3,419	(5,538)
120,000	(64,794.8)	3,498	(7,711)	3,432	(7,567)
115,000	(62,095.0)	3,512	(7,743)	3,447	(7,599)
110,000	(59,395.2)	3,528	(7,778)	3,462	(7,633)
105,000	(56,695.5)	3,545	(7,816)	3,480	(7,671)
100,000	(53,995.7)	3,564	(7,858)	3,498	(7,712)
95,000	(51,295.9)	3,585	(7,904)	3,519	(7,757)
90,000	(48,596.1)	3,608	(7,954)	3,541	(7,807)
85,000	(45,896.3)	3,633	(8,010)	3,566	(7,861)
80,000	(43,196.5)	3,661	(8,072)	3,594	(7,923)
75,000	(40,496.8)	3,693	(8,142)	3,625	(7,992)
70,000	(37,797.0)	3,729	(8,220)	3,660	(8,069)
65,000	(35,097.2)	3,769	(8,310)	3,700	(8,158)
60,000	(32,397.4)	3,816	(8,413)	3,746	(8,259)
55,000	(29,697.6)	3,870	(8,532)	3,800	(8,377)
50,000	(26,997.8)	3,933	(8,671)	3,862	(8,514)
45,000	(24,298.1)	4,009	(8,838)	3,937	(8,679)
40,000	(21,598.3)	4,100	(9,039)	4,027	(8,877)
35,786	**(19,322.9)**	**4,193**	**(9,244)**	**4,119**	**(9,080)**
35,000	(18,898.5)	4,213	(9,287)	4,138	(9,122)
30,000	(16,198.7)	4,354	(9,600)	4,278	(9,432)
25,000	(13,498.9)	4,540	(10,009)	4,461	(9,835)
20,000	(10,799.1)	4,791	(10,562)	4,709	(10,382)
15,000	(8,099.4)	5,151	(11,355)	5,065	(11,166)
10,000	(5,399.6)	5,705	(12,578)	5,614	(12,376)
5,000	(2,699.8)	6,659	(14,681)	6,561	(14,465)

Note: Large Payload Fairing Jettison at 3-sigma qv<1,135 W/m2^2 (360 Btu/ft^2-hr);
Parking Orbit Perigee Altitude = 148.2 km (80 nmi);
Transfer Orbit Perigee Altitude = 166.7 km (90 nmi);
Orbit Inclination = 27.0°;
Argument of Perigee = 180°

Table 2.11-1 Atlas V 401 Geotransfer Orbit Performance—PSW vs Apogee Altitude

Apogee Altitude		Payload Systems Weight, kg (lb) Atlas V 401			
km	(nmi)	MRS		2.33-sigma GCS	
4,630	(2,500)	8,206	(18,090)	7,946	(17,519)
5,556	(3,000)	7,928	(17,479)	7,677	(16,924)
6,482	(3,500)	7,680	(16,932)	7,435	(16,392)
7,408	(4,000)	7,456	(16,438)	7,217	(15,912)
8,334	(4,500)	7,257	(15,999)	7,023	(15,484)
9,260	(5,000)	7,079	(15,606)	6,850	(15,102)
10,186	(5,500)	6,916	(15,247)	6,691	(14,752)
11,112	(6,000)	6,770	(14,924)	6,549	(14,438)
12,038	(6,500)	6,634	(14,625)	6,417	(14,147)
13,427	(7,250)	6,455	(14,231)	6,243	(13,763)
14,816	(8,000)	6,296	(13,881)	6,088	(13,422)
16,205	(8,750)	6,156	(13,573)	5,952	(13,123)
17,594	(9,500)	6,033	(13,301)	5,833	(12,859)
18,983	(10,250)	5,919	(13,050)	5,722	(12,614)
20,372	(11,000)	5,818	(12,827)	5,623	(12,397)
23,150	(12,500)	5,642	(12,437)	5,451	(12,018)
25,928	(14,000)	5,495	(12,115)	5,309	(11,704)
28,706	(15,500)	5,372	(11,843)	5,189	(11,439)
31,484	(17,000)	5,264	(11,606)	5,084	(11,209)
34,262	(18,500)	5,173	(11,404)	4,995	(11,012)
35,786	**(19,323)**	**5,127**	**(11,302)**	**4,950**	**(10,913)**
37,966	(20,500)	5,066	(11,168)	4,891	(10,783)
40,744	(22,000)	4,996	(11,015)	4,823	(10,633)
43,522	(23,500)	4,937	(10,884)	4,766	(10,507)
46,300	(25,000)	4,882	(10,762)	4,712	(10,388)
49,078	(26,500)	4,832	(10,653)	4,663	(10,281)
51,856	(28,000)	4,787	(10,555)	4,620	(10,186)
54,634	(29,500)	4,744	(10,458)	4,578	(10,092)
57,412	(31,000)	4,708	(10,379)	4,542	(10,014)
62,968	(34,000)	4,643	(10,236)	4,480	(9,876)
68,524	(37,000)	4,586	(10,111)	4,424	(9,754)
74,080	(40,000)	4,537	(10,003)	4,377	(9,649)
79,636	(43,000)	4,496	(9,911)	4,336	(9,559)
87,044	(47,000)	4,447	(9,803)	4,288	(9,454)
92,600	(50,000)	4,414	(9,730)	4,256	(9,383)
98,156	(53,000)	4,386	(9,669)	4,229	(9,324)
109,268	(59,000)	4,337	(9,562)	4,182	(9,219)
120,380	(65,000)	4,297	(9,474)	4,143	(9,134)
131,492	(71,000)	4,262	(9,395)	4,108	(9,057)
142,604	(77,000)	4,234	(9,334)	4,081	(8,997)
153,716	(83,000)	4,208	(9,276)	4,056	(8,941)

Note: EPF Jettison at 3-sigma qv ≤1,135 W/m^2 (360 Btu/ft^2-hr)
Parking Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Orbit Inclination = 27.0°
Argument of Perigee = 180°

Table 2.12-1 Atlas V 411 Geotransfer Orbit Performance—PSW vs Apogee Altitude

Apogee Altitude		Payload Systems Weight, kg (lb) Atlas V 421			
km	(nmi)	MRS		2.33-sigma GCS	
150,000	(80,994)	5,050	(11,133)	4,877	(10,752)
145,000	(78,294)	5,061	(11,159)	4,888	(10,777)
140,000	(75,594)	5,074	(11,187)	4,900	(10,803)
135,000	(72,894)	5,087	(11,216)	4,913	(10,832)
130,000	(70,194)	5,102	(11,248)	4,928	(10,864)
125,000	(67,495)	5,119	(11,286)	4,944	(10,901)
120,000	(64,795)	5,138	(11,327)	4,963	(10,941)
115,000	(62,095)	5,158	(11,373)	4,983	(10,987)
110,000	(59,395)	5,182	(11,424)	5,006	(11,036)
105,000	(56,695)	5,207	(11,479)	5,031	(11,091)
100,000	(53,996)	5,234	(11,540)	5,057	(11,150)
95,000	(51,296)	5,263	(11,603)	5,086	(11,213)
90,000	(48,596)	5,296	(11,675)	5,118	(11,282)
85,000	(45,896)	5,331	(11,753)	5,152	(11,358)
80,000	(43,197)	5,370	(11,838)	5,190	(11,442)
75,000	(40,497)	5,413	(11,934)	5,232	(11,535)
70,000	(37,797)	5,463	(12,043)	5,281	(11,642)
65,000	(35,097)	5,519	(12,168)	5,336	(11,765)
60,000	(32,397)	5,586	(12,315)	5,402	(11,909)
55,000	(29,698)	5,664	(12,487)	5,479	(12,080)
50,000	(26,998)	5,758	(12,693)	5,572	(12,283)
45,000	(24,298)	5,870	(12,940)	5,683	(12,528)
40,000	(21,598)	6,004	(13,237)	5,816	(12,823)
35,786	**(19,323)**	**6,139**	**(13,534)**	**5,950**	**(13,118)**
35,000	(18,898)	6,166	(13,595)	5,977	(13,178)
30,000	(16,199)	6,362	(14,027)	6,171	(13,606)

Note: EPF Jettison at 3-sigma qv ≤ 1,135 W/m^2 (360 Btu/ft^2-hr)
Parking Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 (100 nmi)
Orbit Inclination = 27°
Argument of Perigee = 180°

Table 2.13-1 Atlas V 421 Geosynchronous Transfer Orbit Performance—PSW vs Apogee Altitude

Apogee Altitude		Payload Systems Weight, kg (lb) Atlas V 421			
km	(nmi)	MRS		2.33-sigma GCS	
150,000	(80,994)	5,788	(12,760)	5,590	(12,324)
145,000	(78,294)	5,800	(12,786)	5,602	(12,350)
140,000	(75,594)	5,815	(12,820)	5,617	(12,384)
135,000	(72,894)	5,833	(12,859)	5,635	(12,423)
130,000	(70,194)	5,853	(12,905)	5,654	(12,466)
125,000	(67,495)	5,874	(12,951)	5,675	(12,512)
120,000	(64,795)	5,897	(13,000)	5,698	(12,561)
115,000	(62,095)	5,920	(13,052)	5,721	(12,614)
110,000	(59,395)	5,947	(13,111)	5,747	(12,670)
105,000	(56,695)	5,975	(13,172)	5,775	(12,731)
100,000	(53,996)	6,006	(13,241)	5,805	(12,798)
95,000	(51,296)	6,039	(13,314)	5,838	(12,871)
90,000	(48,596)	6,077	(13,397)	5,875	(12,952)
85,000	(45,896)	6,119	(13,489)	5,916	(13,041)
80,000	(43,197)	6,164	(13,589)	5,961	(13,141)
75,000	(40,497)	6,215	(13,703)	6,011	(13,253)
70,000	(37,797)	6,274	(13,831)	6,069	(13,379)
65,000	(35,097)	6,340	(13,978)	6,134	(13,524)
60,000	(32,397)	6,417	(14,147)	6,210	(13,690)
55,000	(29,698)	6,506	(14,343)	6,298	(13,885)
50,000	(26,998)	6,611	(14,576)	6,402	(14,115)
45,000	(24,298)	6,737	(14,853)	6,527	(14,390)
40,000	(21,598)	6,889	(15,188)	6,678	(14,723)
35,786	**(19,323)**	**7,042**	**(15,525)**	**6,830**	**(15,058)**
35,000	(18,898)	7,074	(15,595)	6,862	(15,127)
30,000	(16,199)	7,300	(16,094)	7,086	(15,623)

Note: EPF Jettison at 3-sigma qv ≤ 1,135 W/m^2 (360 Btu/ft^2-hr)
Parking Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 (100 nmi)
Orbit Inclination = 27°
Argument of Perigee = 180°

Table 2.14-1 Atlas V 431 Geosynchronous Transfer Orbit Performance—PSW vs Apogee Altitude

Apogee Altitude		Payload System Weight, kg (lb) Atlas V 431			
km	(nmi)	MRS		2.33-sigma GCS	
150,000	(80,994)	6,468	(14,259)	6,248	(13,774)
145,000	(78,294)	6,487	(14,302)	6,267	(13,817)
140,000	(75,594)	6,509	(14,349)	6,288	(13,862)
135,000	(72,894)	6,530	(14,397)	6,309	(13,910)
130,000	(70,194)	6,554	(14,450)	6,332	(13,960)
125,000	(67,495)	6,579	(14,503)	6,357	(14,014)
120,000	(64,795)	6,605	(14,563)	6,382	(14,071)
115,000	(62,095)	6,633	(14,623)	6,410	(14,132)
110,000	(59,395)	6,663	(14,690)	6,439	(14,197)
105,000	(56,695)	6,696	(14,762)	6,471	(14,266)
100,000	(53,996)	6,730	(14,838)	6,504	(14,339)
95,000	(51,296)	6,767	(14,920)	6,540	(14,419)
90,000	(48,596)	6,808	(15,009)	6,580	(14,506)
85,000	(45,896)	6,852	(15,107)	6,623	(14,602)
80,000	(43,197)	6,902	(15,216)	6,672	(14,709)
75,000	(40,497)	6,958	(15,339)	6,727	(14,830)
70,000	(37,797)	7,022	(15,480)	6,790	(14,969)
65,000	(35,097)	7,096	(15,643)	6,863	(15,130)
60,000	(32,397)	7,182	(15,834)	6,948	(15,318)
55,000	(29,698)	7,283	(16,057)	7,048	(15,539)
50,000	(26,998)	7,403	(16,321)	7,167	(15,801)
45,000	(24,298)	7,545	(16,634)	7,308	(16,112)
40,000	(21,598)	7,713	(17,005)	7,475	(16,481)
35,786	**(19,323)**	**7,879**	**(17,371)**	**7,640**	**(16,844)**
35,000	(18,898)	7,913	(17,445)	7,674	(16,918)
30,000	(16,199)	8,150	(17,967)	7,909	(17,436)

Note: EPF Jettison at 3-sigma qv ≤ 1,135 W/m^2 (360 Btu/ft^2-hr)
Parking Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Orbit Inclination = 27°
Argument of Perigee = 180°

Table 2.15-1 Atlas V 500 (SEC) Series Geotransfer Orbit Performance—PSW vs Apogee Altitude

Apogee Altitude		Payload Systems Weight, kg (lb)											
		Atlas V 501				Atlas V 511				Atlas V 521			
km	(nmi)	MRS		2.33-sigma GCS		MRS		2.33-sigma GCS		MRS		2.33-sigma GCS	
5,000	(2,700)	6,669	(14,703)	6,448	(14,216)	8,975	(19,786)	8,649	(19,067)	10,753	(23,706)	10,398	(22,924)
10,000	(5,400)	5,705	(12,578)	5,512	(12,152)	7,627	(16,815)	7,335	(16,172)	9,076	(20,009)	8,770	(19,334)
15,000	(8,099)	5,128	(11,305)	4,951	(10,915)	6,843	(15,086)	6,571	(14,486)	8,110	(17,879)	7,832	(17,266)
20,000	(10,799)	4,747	(10,466)	4,582	(10,101)	6,340	(13,976)	6,075	(13,392)	7,511	(16,559)	7,250	(15,984)
25,000	(13,499)	4,480	(9,877)	4,322	(9,528)	5,985	(13,196)	5,750	(12,677)	7,093	(15,638)	6,844	(15,089)
30,000	(16,199)	4,288	(9,452)	4,135	(9,116)	5,732	(12,637)	5,489	(12,101)	6,789	(14,967)	6,549	(14,439)
35,786	**(19,323)**	**4,118**	**(9,078)**	**3,970**	**(8,752)**	**5,484**	**(12,091)**	**5,270**	**(11,618)**	**6,517**	**(14,367)**	**6,285**	**(13,856)**
40,000	(21,598)	4,020	(8,862)	3,874	(8,542)	5,379	(11,858)	5,146	(11,345)	6,342	(13,982)	6,115	(13,482)
45,000	(24,298)	3,925	(8,652)	3,782	(8,339)	5,257	(11,589)	5,029	(11,087)	6,204	(13,677)	5,981	(13,185)
50,000	(26,998)	3,846	(8,478)	3,705	(8,169)	5,157	(11,370)	4,928	(10,864)	6,095	(13,437)	5,875	(12,952)
55,000	(29,698)	3,779	(8,332)	3,641	(8,027)	5,072	(11,181)	4,844	(10,679)	6,006	(13,241)	5,789	(12,762)
60,000	(32,397)	3,723	(8,208)	3,586	(7,906)	4,996	(11,015)	4,775	(10,528)	5,931	(13,076)	5,716	(12,602)
65,000	(35,097)	3,674	(8,100)	3,539	(7,802)	4,934	(10,878)	4,714	(10,393)	5,865	(12,931)	5,652	(12,461)
70,000	(37,797)	3,632	(8,007)	3,498	(7,711)	4,882	(10,763)	4,663	(10,280)	5,807	(12,801)	5,595	(12,335)
77,000	(41,577)	3,581	(7,894)	3,448	(7,602)	4,815	(10,615)	4,599	(10,140)	5,704	(12,576)	5,496	(12,116)
Apogee Altitude		Atlas V 531				Atlas V 541				Atlas V 551			
km	(nmi)	MRS		2.33-sigma GCS		MRS		2.33-sigma GCS		MRS		2.33-sigma GCS	
5,000	(2,700)	12,175	(26,841)	11,782	(25,976)	13,517	(29,799)	13,087	(28,853)	14,651	(32,299)	14,180	(31,262)
10,000	(5,400)	10,302	(22,712)	9,964	(21,968)	11,423	(25,184)	11,055	(24,373)	12,397	(27,330)	11,993	(26,441)
15,000	(8,099)	9,232	(20,354)	8,926	(19,679)	10,231	(22,556)	9,898	(21,822)	11,106	(24,484)	10,741	(23,680)
20,000	(10,799)	8,553	(18,855)	8,266	(18,224)	9,478	(20,894)	9,167	(20,209)	10,289	(22,684)	9,949	(21,934)
25,000	(13,499)	8,086	(17,826)	7,813	(17,225)	8,958	(19,748)	8,662	(19,096)	9,730	(21,450)	9,406	(20,736)
30,000	(16,199)	7,751	(17,087)	7,488	(16,508)	8,582	(18,919)	8,297	(18,292)	9,323	(20,553)	9,011	(19,866)
35,786	**(19,323)**	**7,454**	**(16,433)**	**7,200**	**(15,873)**	**8,255**	**(18,200)**	**7,980**	**(17,593)**	**8,971**	**(19,778)**	**8,670**	**(19,114)**
40,000	(21,598)	7,285	(16,061)	7,036	(15,512)	8,070	(17,791)	7,800	(17,196)	8,770	(19,334)	8,474	(18,683)
45,000	(24,298)	7,120	(15,697)	6,876	(15,158)	7,897	(17,410)	7,632	(16,827)	8,574	(18,901)	8,284	(18,264)
50,000	(26,998)	6,985	(15,398)	6,744	(14,869)	7,747	(17,078)	7,486	(16,504)	8,414	(18,549)	8,129	(17,922)
55,000	(29,698)	6,871	(15,147)	6,634	(14,625)	7,622	(16,803)	7,365	(16,237)	8,279	(18,251)	7,998	(17,633)
60,000	(32,397)	6,773	(14,933)	6,539	(14,417)	7,515	(16,568)	7,262	(16,009)	8,164	(17,999)	7,887	(17,389)
65,000	(35,097)	6,690	(14,749)	6,459	(14,239)	7,423	(16,365)	7,172	(15,812)	8,066	(17,782)	7,792	(17,178)
70,000	(37,797)	6,617	(14,589)	6,388	(14,083)	7,343	(16,189)	7,095	(15,641)	7,981	(17,595)	7,709	(16,996)
77,000	(41,577)	6,531	(14,398)	6,304	(13,898)	7,249	(15,980)	7,003	(15,439)	7,878	(17,368)	7,609	(16,776)

Note: 5-m Short PLF Jettison at 3-sigma $qV \le 1{,}135\ W/m^2$ ($360\ Btu/ft^2$-hr)
Parking Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 167 km (90 nmi)
Orbit Inclination = 27.0°
Argument of Perigee = 180°

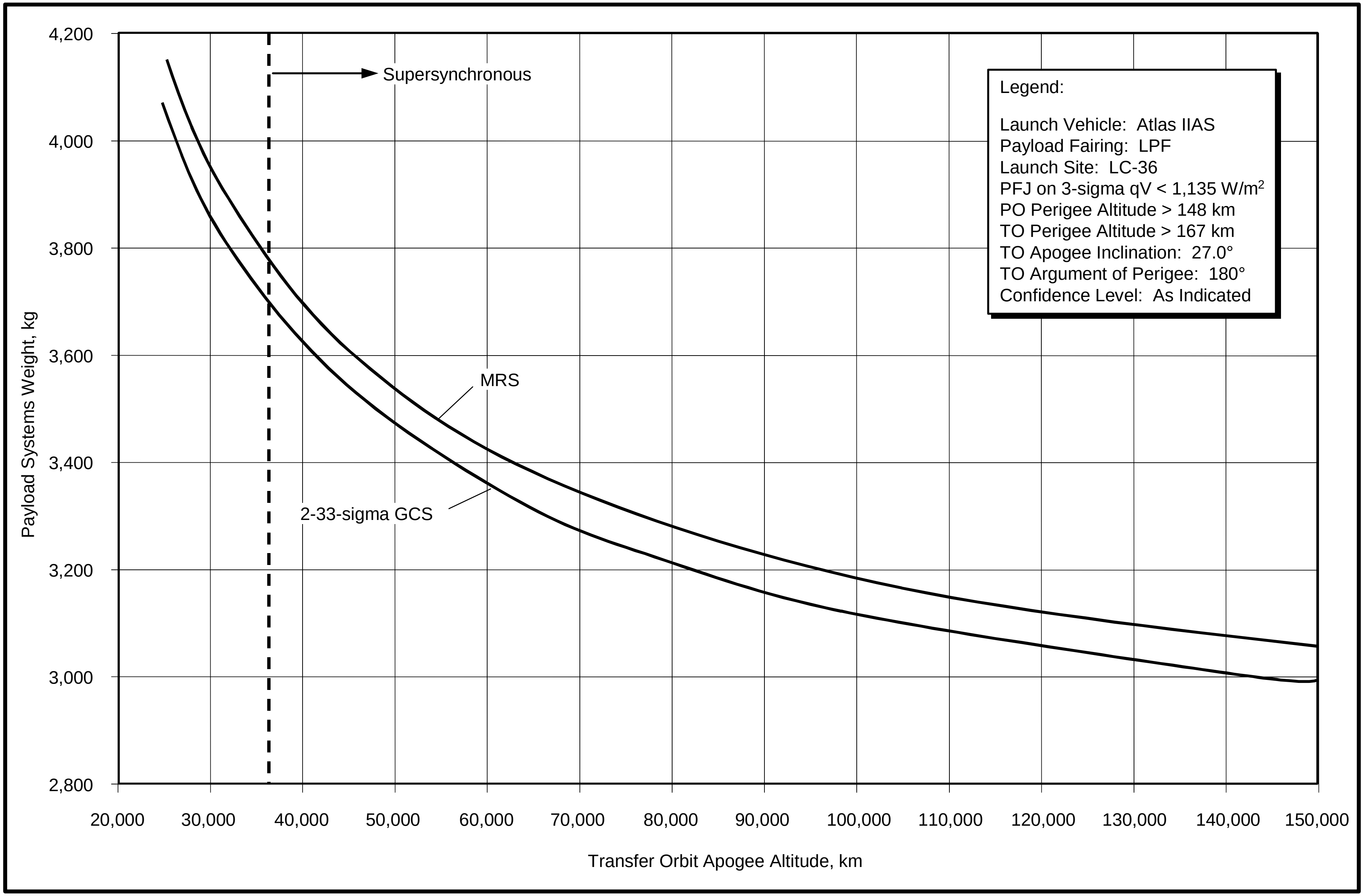


Figure 2.7-2 Atlas IIAS Performance to Geotransfer Orbit

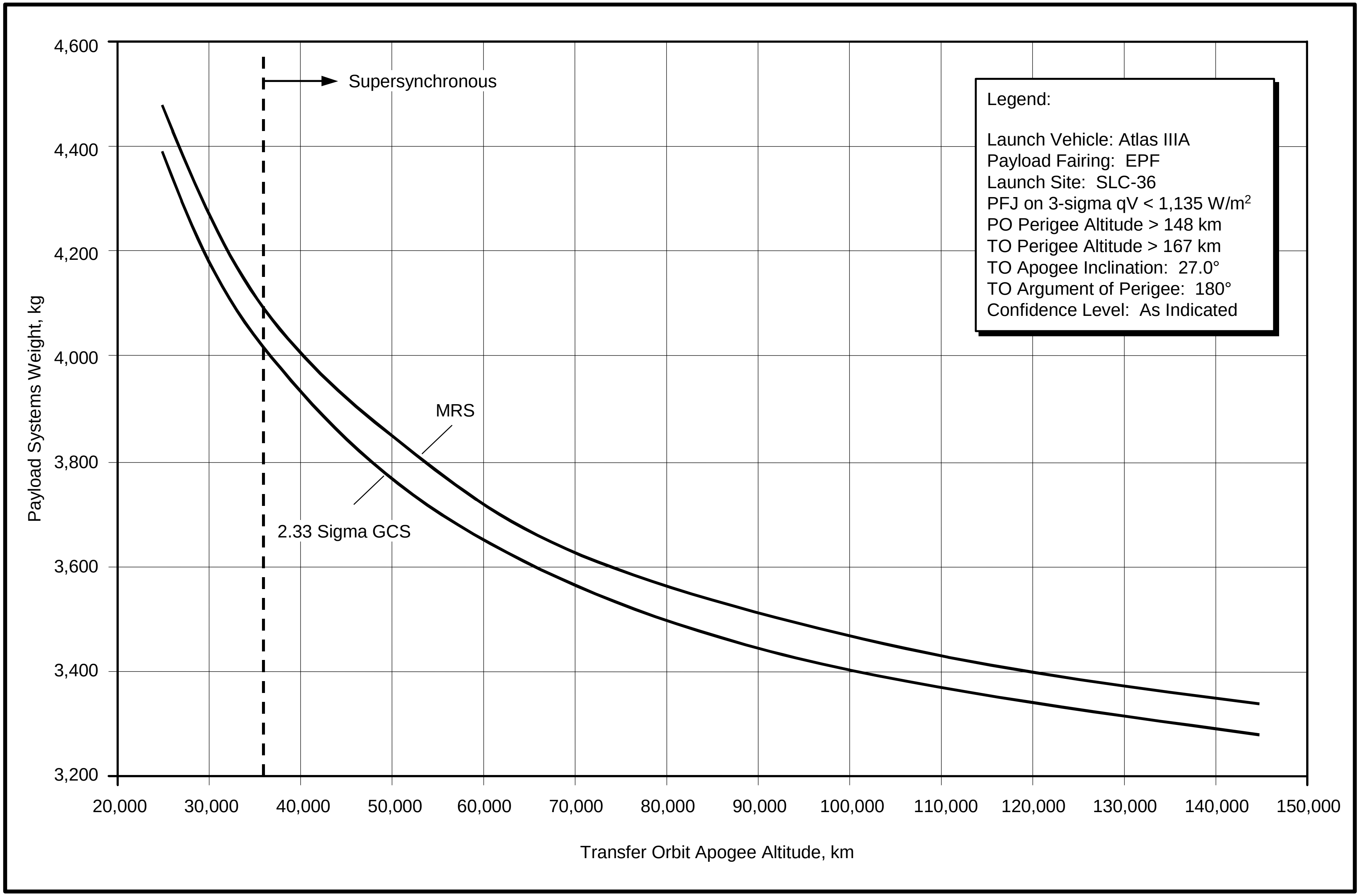


Figure 2.8-2 Atlas IIIA Performance to Geotransfer Orbit

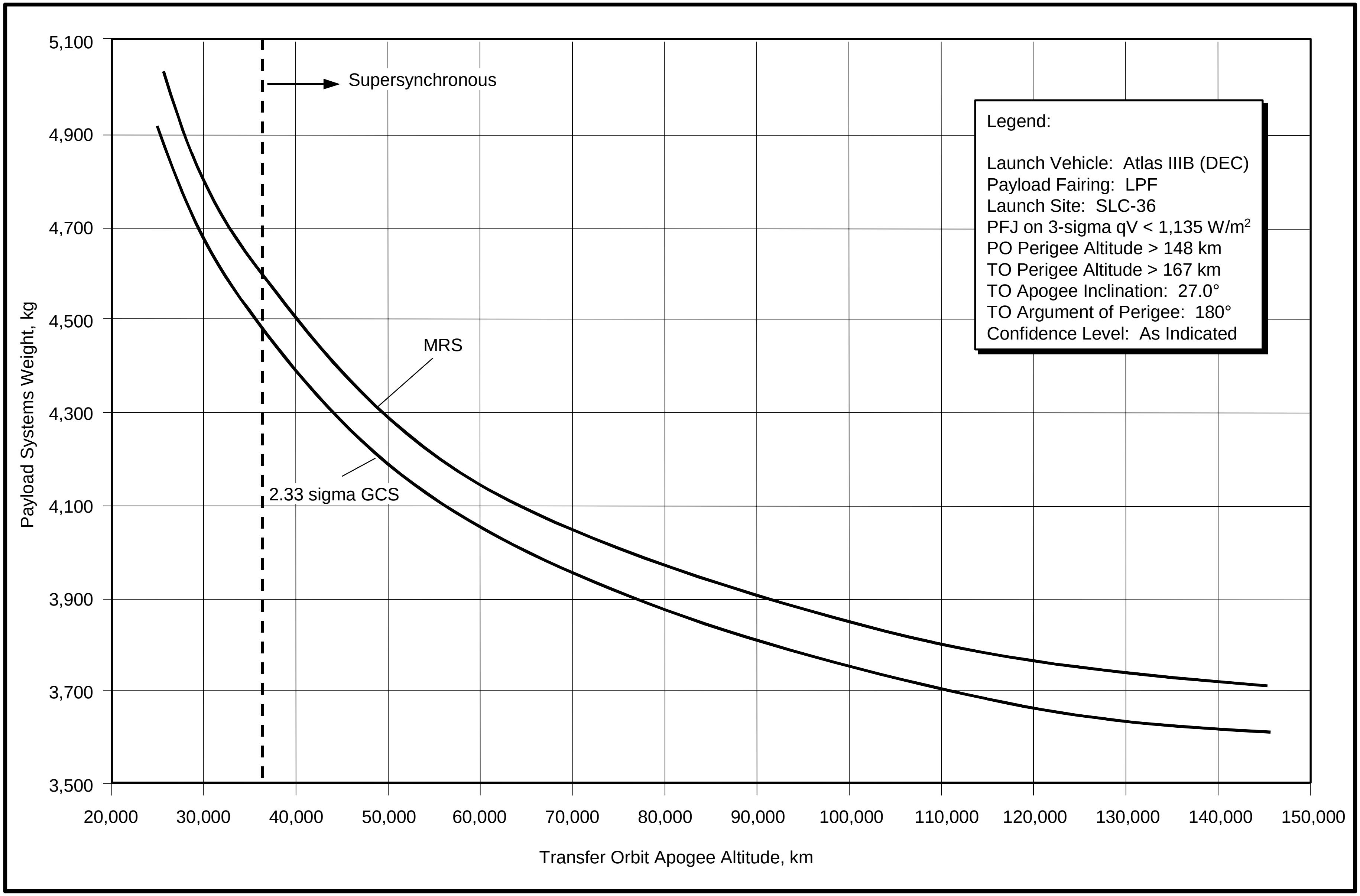


Figure 2.9-2 Atlas IIIB (DEC) Performance to Geotransfer Orbit

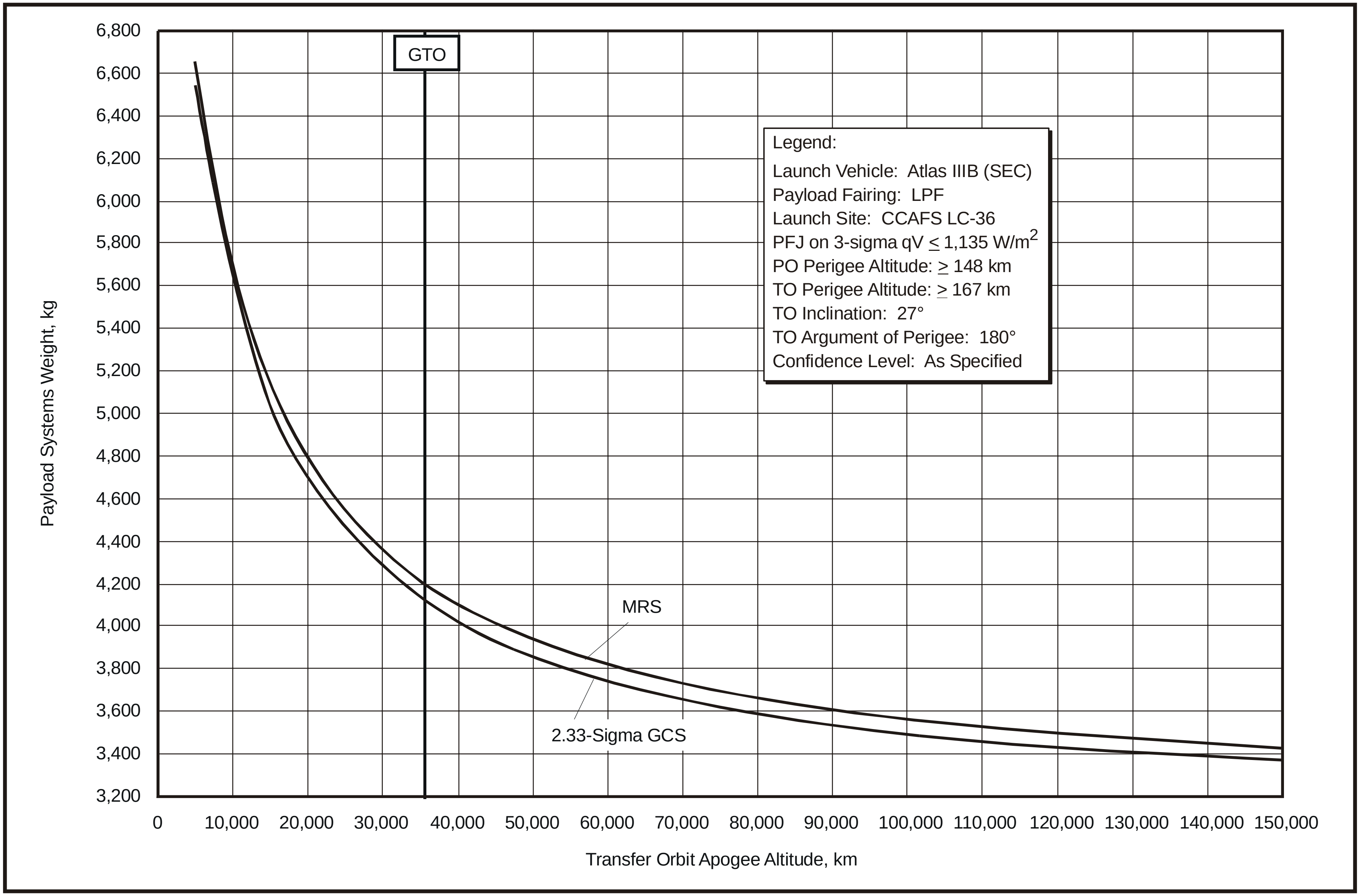


Figure 2.10-2 Atlas IIIB (SEC) CCAFS Performance to Geotransfer Orbit

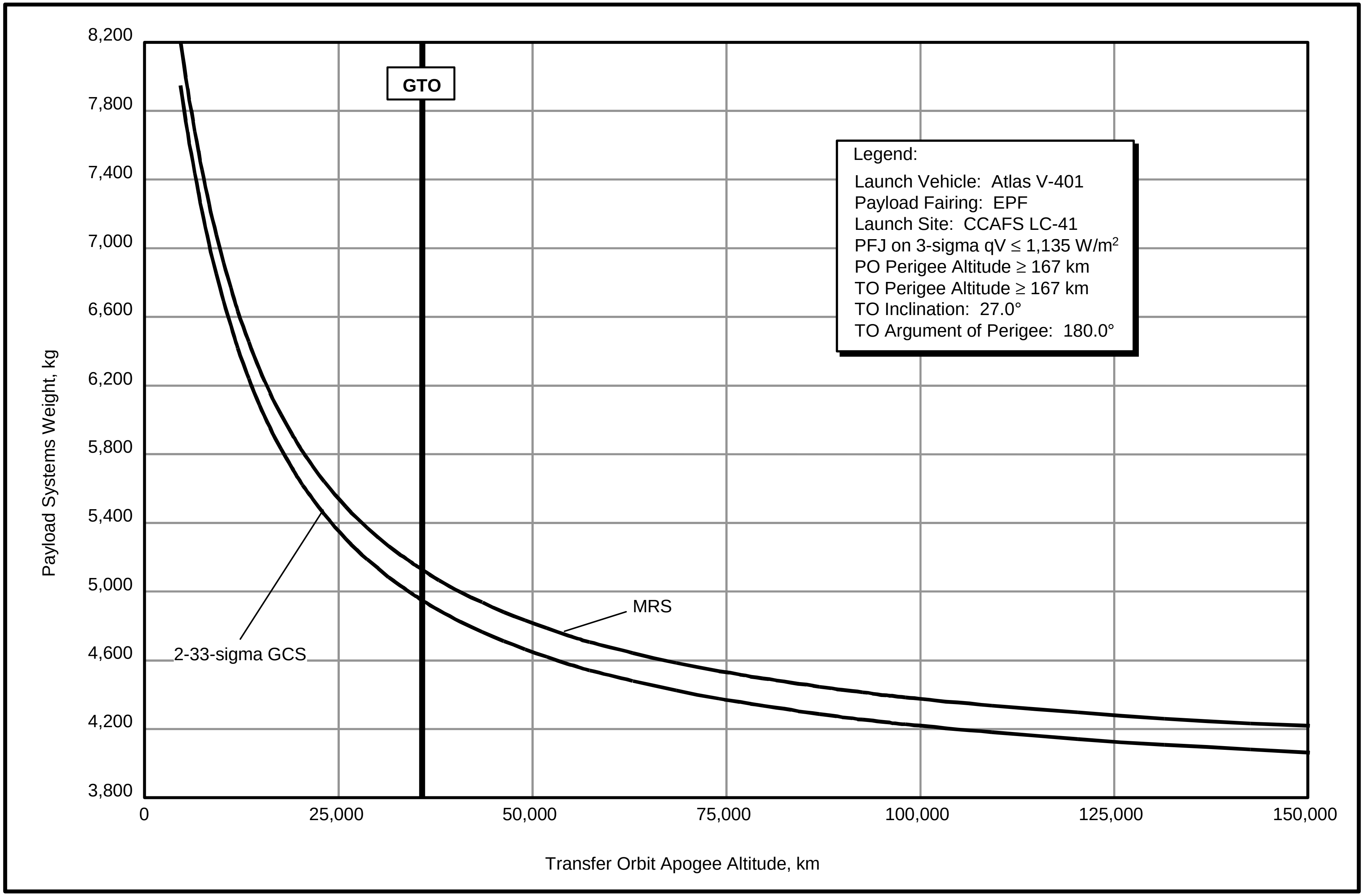


Figure 2.11-2 Atlas V 401 CCAFS Performance to Geotransfer Orbit

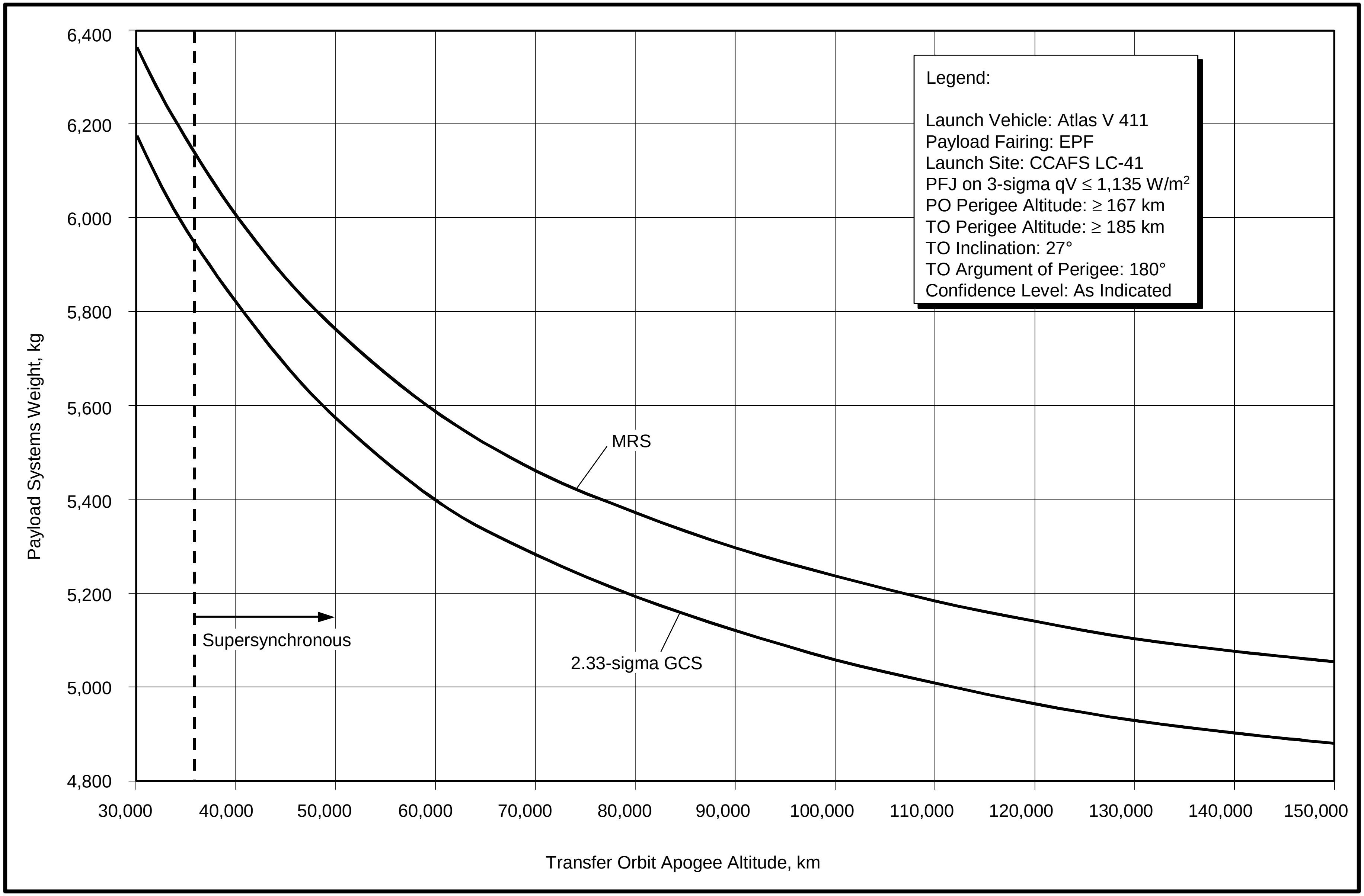


Figure 2.12-2 Atlas V 411 Performance to Geotransfer Orbit with Apogee Variation

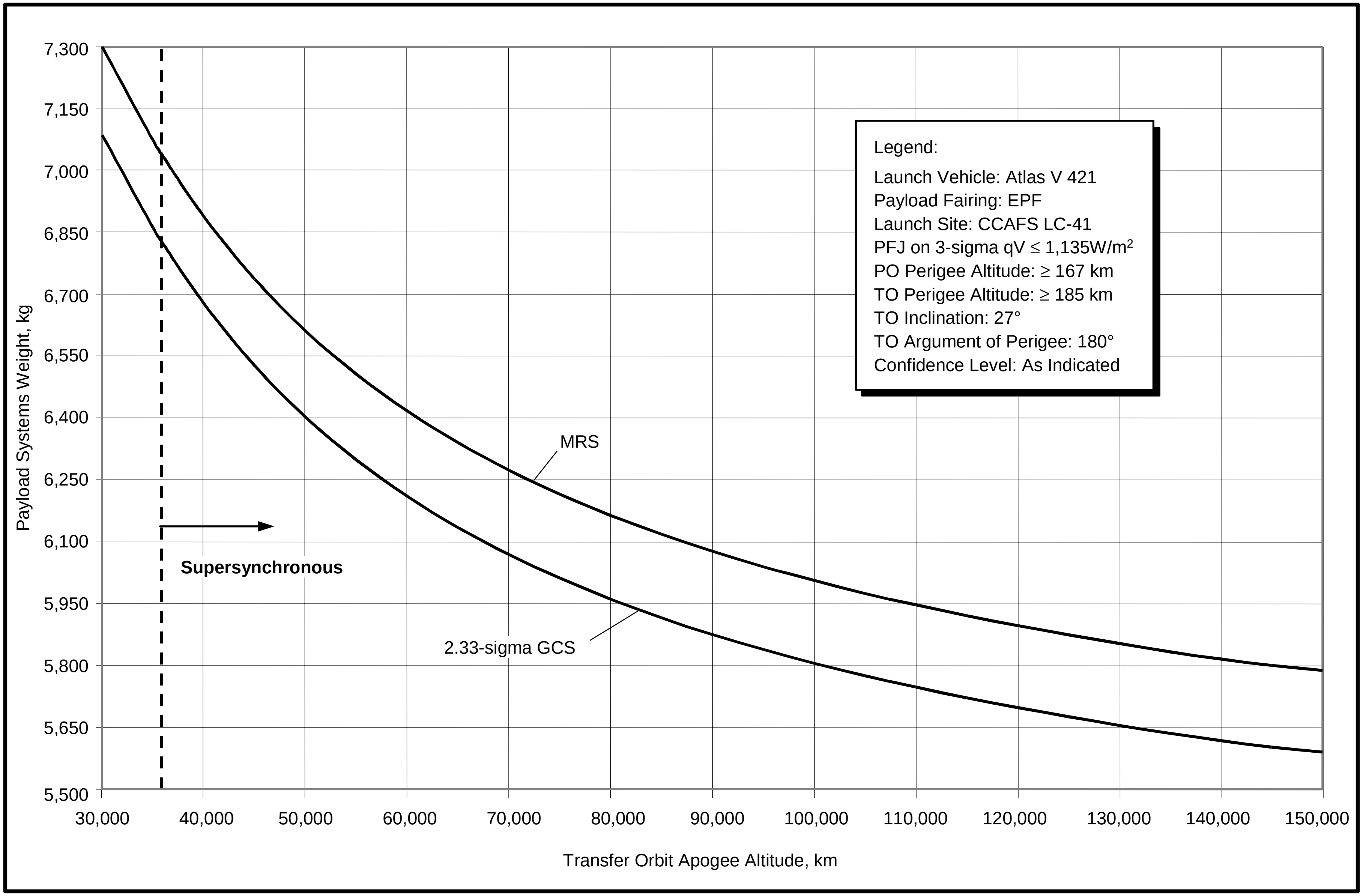


Figure 2.13-2 Atlas V 421 Performance to Geotransfer Orbit with Apogee Variation

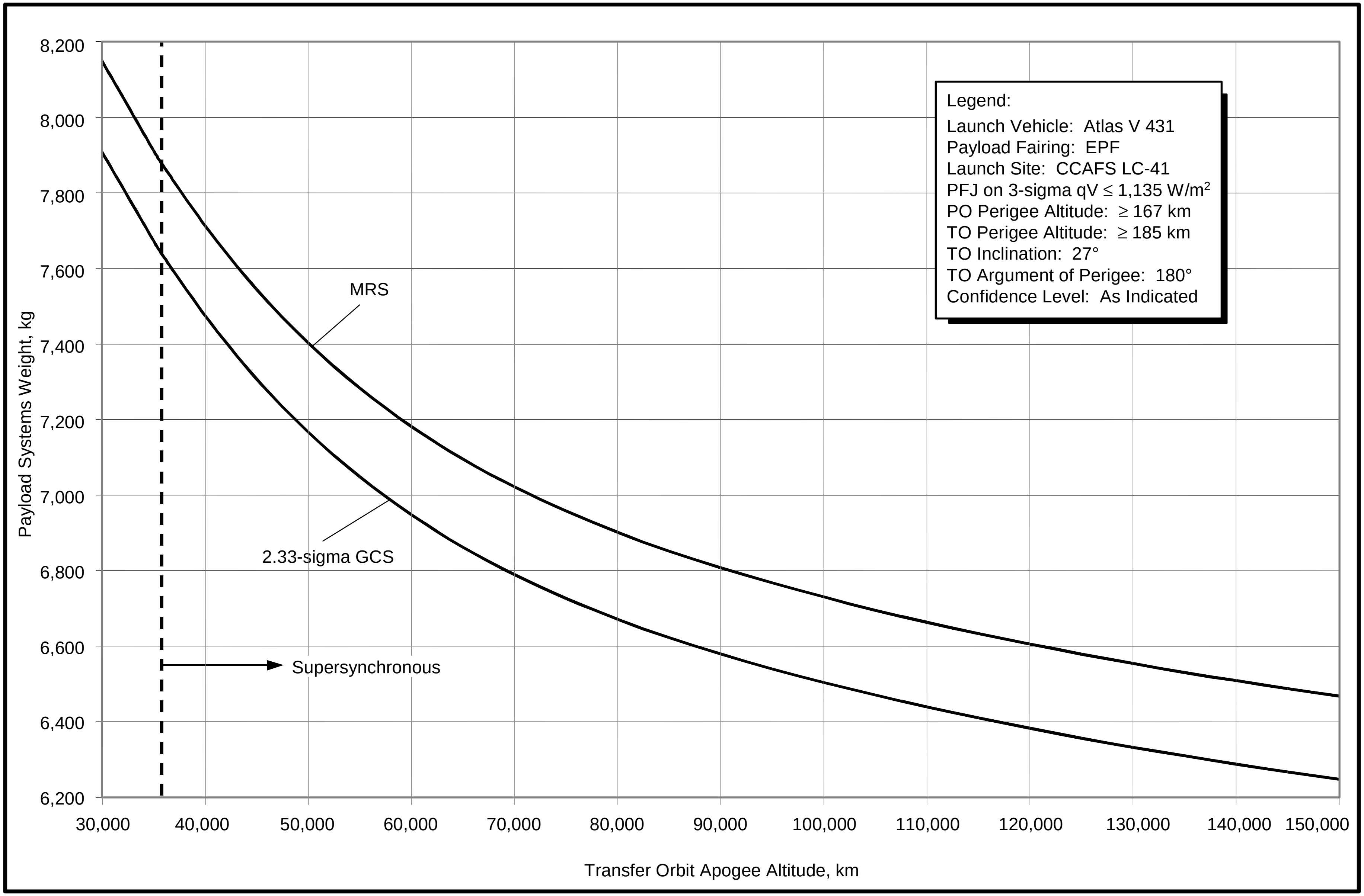


Figure 2.14-2 Atlas V 431 Performance to Geotransfer Orbit with Apogee Variation

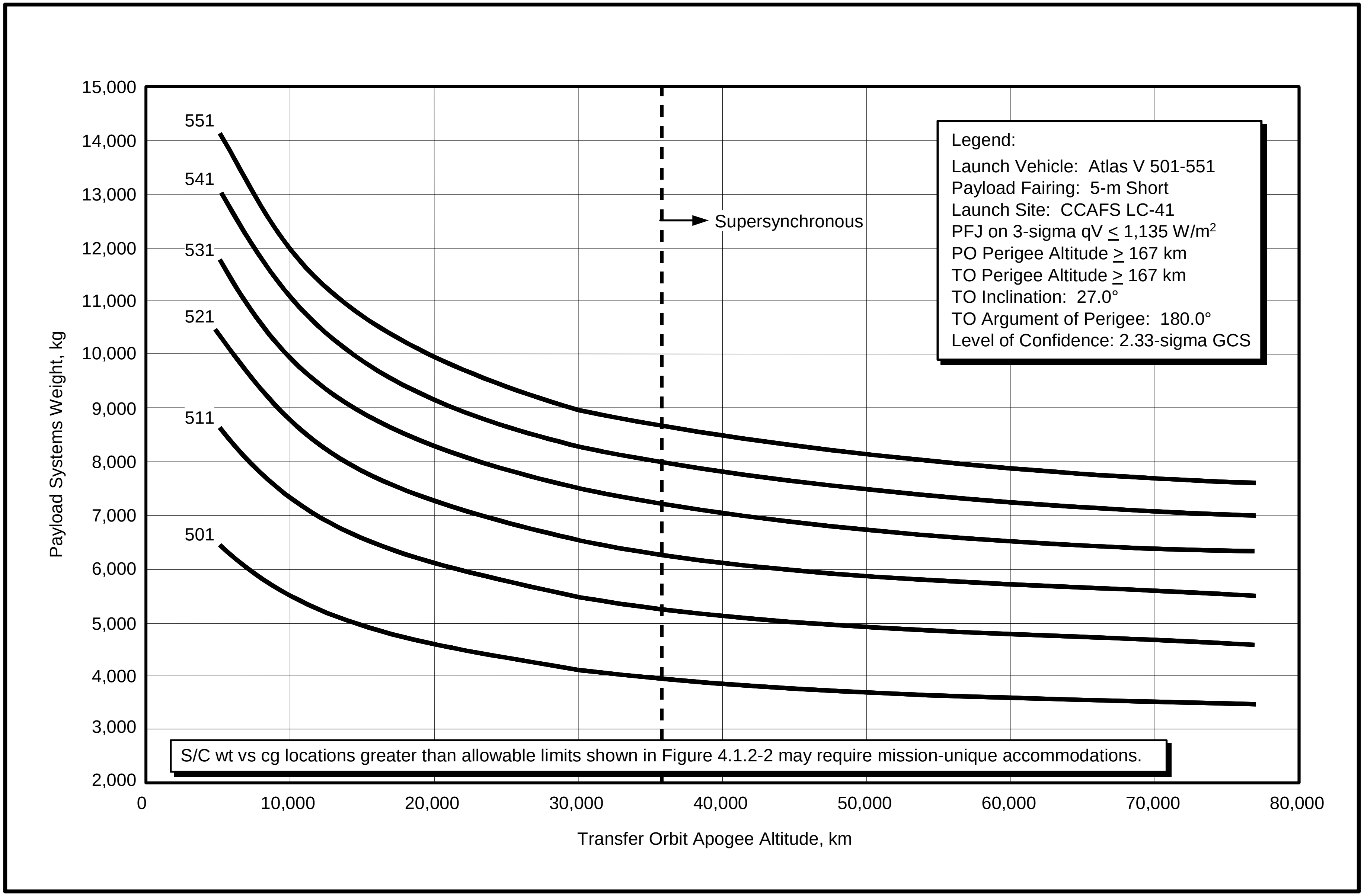


Figure 2.15-2a Atlas V 501-551 Performance to Geotransfer Orbit (GCS)

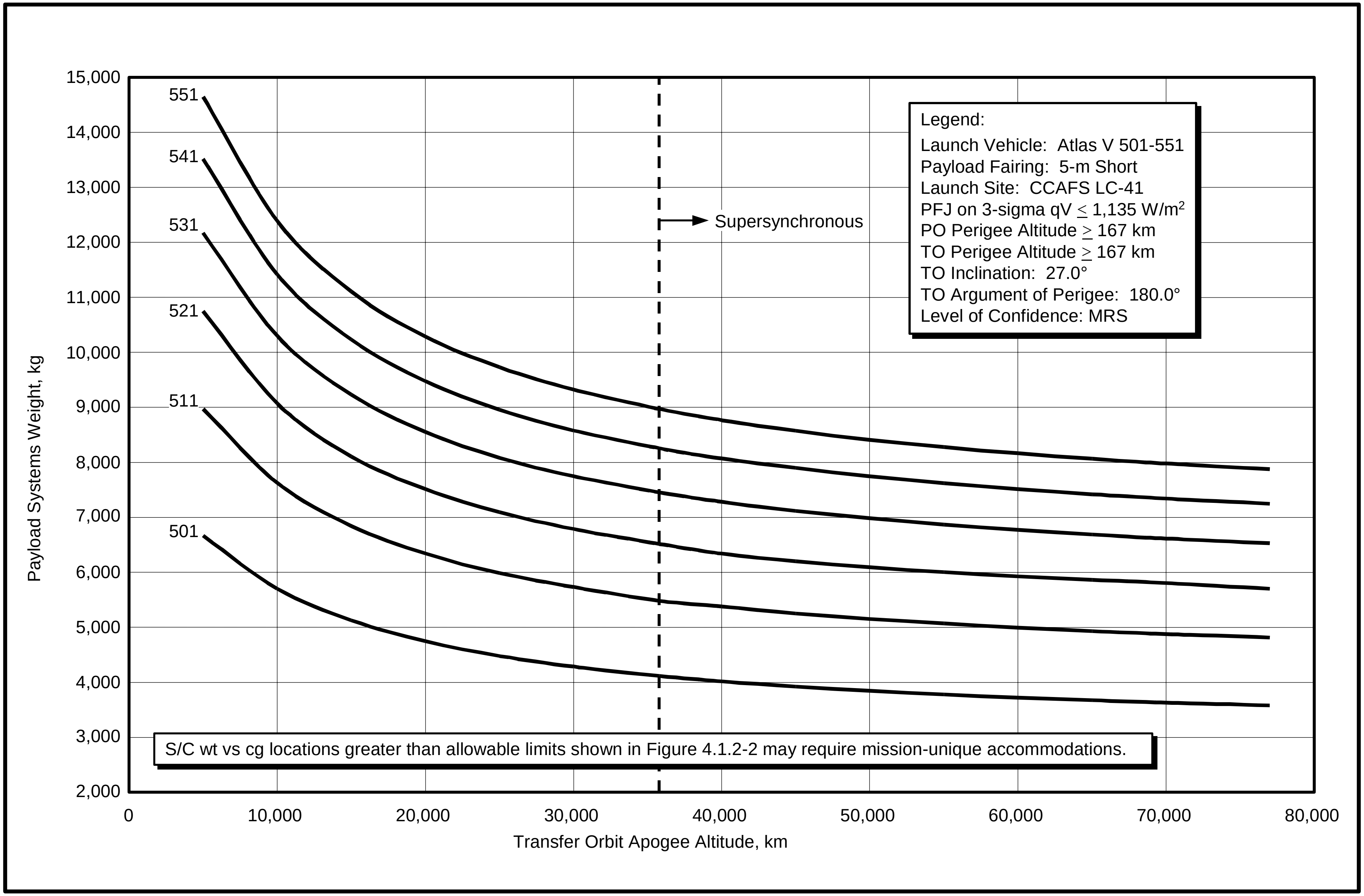


Figure 2.15-2b Atlas V 501-551 Performance to Geotransfer Orbit (MRS)

Table 2.7-2 Atlas IIAS Geotransfer Orbit Performance—PSW vs Inclination

Inclination, °	Payload Systems Weight, kg (lb) Atlas IIAS MRS		GCS	
18.00	3,188	(7,029)	3,118	(6,875)
18.50	3,240	(7,143)	3,169	(6,987)
19.00	3,290	(7,254)	3,219	(7,096)
19.50	3,339	(7,361)	3,266	(7,202)
20.00	3,386	(7,465)	3,313	(7,304)
20.50	3,431	(7,565)	3,357	(7,402)
21.00	3,474	(7,659)	3,399	(7,495)
21.50	3,515	(7,750)	3,440	(7,585)
22.00	3,554	(7,836)	3,479	(7,669)
22.50	3,591	(7,918)	3,515	(7,749)
23.00	3,625	(7,992)	3,548	(7,823)
23.50	3,657	(8,063)	3,579	(7,892)
24.00	3,686	(8,126)	3,608	(7,954)
24.50	3,712	(8,184)	3,634	(8,012)
25.00	3,735	(8,236)	3,657	(8,062)
25.50	3,756	(8,280)	3,676	(8,106)
26.00	3,773	(8,319)	3,694	(8,144)
26.50	3,788	(8,351)	3,708	(8,175)
27.00	**3,799**	**(8,376)**	**3,719**	**(8,200)**
27.50	3,807	(8,393)	3,727	(8,217)
28.00	3,812	(8,404)	3,732	(8,228)
28.50	3,813	(8,408)	3,733	(8,231)
29.00	3,813	(8,407)	3,733	(8,230)
29.50	3,811	(8,403)	3,731	(8,227)
30.00	3,809	(8,399)	3,729	(8,223)

Note: Large (4.2-m) Diameter Payload Fairing Jettison at 3-Sigma qv<1,135 W/m2 (360 Btu/ft2-hr); Parking Orbit Perigee Altitude = 148.2 km (80 nmi); Transfer Orbit Perigee Altitude = 166.7 (90 nmi); Transfer Orbit Apogee Altitude = 35,786 km (19,323 nmi); Argument of Perigee = 180°

Table 2.8-2 Atlas IIIA Geotransfer Orbit Performance—PSW vs Inclination

Inclination, °	Payload Systems Weight, kg (lb) Atlas IIIA MRS		GCS	
18.00	3,474	(7,659)	3,409	(7,515)
18.50	3,528	(7,778)	3,462	(7,631)
19.00	3,580	(7,892)	3,513	(7,745)
19.50	3,630	(8,004)	3,563	(7,854)
20.00	3,679	(8,111)	3,611	(7,960)
20.50	3,726	(8,215)	3,657	(8,062)
21.00	3,771	(8,313)	3,701	(8,159)
21.50	3,814	(8,408)	3,743	(8,252)
22.00	3,854	(8,497)	3,783	(8,340)
22.50	3,892	(8,580)	3,820	(8,423)
23.00	3,928	(8,659)	3,856	(8,500)
23.50	3,961	(8,732)	3,888	(8,572)
24.00	3,991	(8,799)	3,918	(8,638)
24.50	4,019	(8,860)	3,945	(8,697)
25.00	4,043	(8,914)	3,969	(8,751)
25.50	4,065	(8,962)	3,991	(8,798)
26.00	4,084	(9,003)	4,009	(8,839)
26.50	4,099	(9,037)	4,025	(8,873)
27.00	**4,112**	**(9,065)**	**4,037**	**(8,900)**
27.50	4,121	(9,086)	4,046	(8,920)
28.00	4,127	(9,099)	4,052	(8,933)
28.50	4,131	(9,106)	4,055	(8,940)
29.00	4,132	(9,109)	4,057	(8,943)
29.50	4,132	(9,109)	4,056	(8,943)
30.00	4,129	(9,102)	4,054	(8,937)

Note: Extended Payload Fairing Jettison at 3-sigma qv<1,135 W/m^2 (360 Btu/ft^2-hr); Parking Orbit Perigee Altitude ≥ 148.2 km (80 nmi); Transfer Orbit Perigee Altitude ≥ 166.7 (90 nmi); Transfer Orbit Apogee Altitude = 35,786 km (19,323 nmi); Argument of Perigee = 180°

Table 2.9-2 Atlas IIIB (DEC) Geotransfer Orbit Performance—PSW vs Inclination

Inclination, °	Payload Systems Weight, kg (lb) Atlas IIIB (DEC)			
	MRS		GCS	
18.00	3,826	(8,434)	3,730	(8,223)
18.50	3,890	(8,575)	3,793	(8,362
19.00	3,952	(8,713)	3,854	(8,497)
19.50	4,012	(8,846)	3,914	(8,628)
20.00	4,071	(8,975)	3,971	(8,755)
20.50	4,127	(9,099)	4,027	(8,877)
21.00	4,182	(9,220)	4,080	(8,996)
21.50	4,235	(9,336)	4,132	(9,110)
22.00	4,285	(9,447)	4,182	(9,219)
22.50	4,332	(9,551)	4,228	(9,322)
23.00	4,377	(9,649)	4,272	(9,418)
23.50	4,418	(9,741)	4,313	(9,508)
24.00	4,456	(9,824)	4,350	(9,590)
24.50	4,491	(9,901	4,384	(9,665)
25.00	4,522	(9,969)	4,415	(9,733)
25.50	4,549	(10,029)	4,442	(9,792)
26.00	4,573	(10,082)	4,465	(9,844)
26.50	4,593	(10,126)	4,484	(9,887)
27.00	**4,609**	**(10,161)**	**4,500**	**(9,920)**
27.50	4,621	(10,187)	4,512	(9,947)
28.00	4,628	(10,204)	4,519	(9,964)
28.50	4,633	(10,213)	4,523	(9,972)
29.00	4,632	(10,211)	4,522	(9,970)
29.50	4,628	(10,203)	4,519	(9,963)
30.00	4,624	(10,195)	4,515	(9,954)

Note: Large Payload Fairing Jettison at 3-sigma qv<1,135 W/m^2 (360 Btu/ft^2-hr);
Parking Orbit Perigee Altitude ≥ 148.2 km (80 nmi);
Transfer Orbit Perigee Altitude ≥ 166.7 (90 nmi);
Transfer Orbit Apogee Altitude = 35,786 km (19,323 mn);
Argument of Perigee = 180°

Table 2.10-2 Atlas IIIB (SEC) Geotransfer Orbit Performance—PSW vs Inclination

Inclination, °	Payload Systems Weight, kg (lb) Atlas IIIB (SEC)			
	MRS		GCS	
18.00	3,602	(7,942)	3,534	(7,792)
18.50	3,653	(8,054)	3,585	(7,903)
19.00	3,702	(8,162)	3,633	(8,010)
19.50	3,750	(8,267)	3,680	(8,114)
20.00	3,795	(8,368)	3,725	(8,213)
20.50	3,839	(8,464)	3,769	(8,308)
21.00	3,881	(8,556)	3,810	(8,399)
21.50	3,920	(8,643)	3,849	(8,485)
22.00	3,958	(8,725)	3,886	(8,567)
22.50	3,993	(8,802)	3,920	(8,643)
23.00	4,025	(8,874)	3,953	(8,714)
23.50	4,056	(8,941)	3,983	(8,780)
24.00	4,083	(9,002)	4,010	(8,840)
24.50	4,108	(9,057)	4,035	(8,895)
25.00	4,131	(9,107)	4,057	(8,944)
25.50	4,151	(9,150)	4,077	(8,987)
26.00	4,167	(9,188)	4,093	(9,024)
26.50	4,182	(9,219)	4,107	(9,055)
27.00	**4,193**	**(9,244)**	**4,119**	**(9,080)**
27.50	4,201	(9,263)	4,127	(9,098)
28.00	4,208	(9,276)	4,133	(9,112)
28.50	4,212	(9,286)	4,137	(9,121)
29.00	4,214	(9,290)	4,139	(9,125)
29.50	4,213	(9,287)	4,138	(9,123)
30.00	4,209	(9,279)	4,134	(9,115)

Note: Large Payload Fairing Jettison at 3-sigma qv<1,135W/m^2 (360 Btu/ft^2-hr);
Parking Orbit Perigee Altitude = 148.2 km (80 nmi);
Transfer Orbit Perigee Altitude = 166.7 (90 nmi);
Transfer Orbit Apogee Altitude = 35,786 km (19,323 nmi);
Argument of Perigee = 180°

Table 2.11-2 Atlas V 401 Geotransfer Orbit Performance—PSW vs Inclination

Inclination, °	Payload Systems Weight, kg (lb) Atlas V 401			
	MRS		2.33-sigma GCS	
18.0	4,235	(9,336)	4,082	(9,000)
18.5	4,308	(9,497)	4,153	(9,156)
19.0	4,378	(9,652)	4,222	(9,307)
19.5	4,447	(9,803)	4,288	(9,454)
20.0	4,513	(9,950)	4,353	(9,596)
20.5	4,577	(10,090)	4,415	(9,733)
21.0	4,638	(10,226)	4,475	(9,865)
21.5	4,697	(10,355)	4,532	(9,991)
22.0	4,753	(10,478)	4,586	(10,111)
22.5	4,806	(10,595)	4,638	(10,224)
23.0	4,856	(10,705)	4,686	(10,331)
23.5	4,902	(10,808)	4,732	(10,432)
24.0	4,946	(10,904)	4,774	(10,525)
24.5	4,986	(10,991)	4,813	(10,610)
25.0	5,022	(11,071)	4,848	(10,687)
25.5	5,054	(11,143)	4,879	(10,757)
26.0	5,083	(11,205)	4,907	(10,818)
26.5	5,107	(11,259)	4,930	(10,870)
27.0	**5,127**	**(11,304)**	**4,950**	**(10,913)**
27.5	5,140	(11,332)	4,963	(10,942)
28.0	5,149	(11,351)	4,972	(10,961)
28.5	5,155	(11,365)	4,978	(10,974)
29.0	5,158	(11,371)	4,981	(10,981)
29.5	5,160	(11,375)	4,982	(10,984)
30.0	5,160	(11,376)	4,983	(10,985)

Note: EPF Jettison at 3-sigma qV < 1,135 W/M^2 (360 Btu/ft^2-hr)
Parking Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Apogee Altitude = 35,786 km (19,323 mn)
Argument of Perigee = 180°

Table 2.12-2 Atlas V 411 Geotransfer Orbit Performance—PSW vs Inclination

Inclination, °	Payload Systems Weight, kg (lb) Atlas V 411			
	MRS		2.33-sigma GCS	
15.0	4,647	(10,245)	4,478	(9,872)
15.5	4,747	(10,465)	4,576	(10,088)
16.0	4,847	(10,685)	4,674	(10,304)
16.5	4,944	(10,899)	4,770	(10,516)
17.0	5,039	(11,110)	4,863	(10,722)
17.5	5,131	(11,312)	4,954	(10,922)
18.0	5,220	(11,507)	5,041	(11,113)
18.5	5,303	(11,692)	5,123	(11,295)
19.0	5,384	(11,869)	5,202	(11,468)
19.5	5,458	(12,033)	5,276	(11,632)
20.0	5,529	(12,189)	5,346	(11,785)
20.5	5,594	(12,333)	5,411	(11,930)
21.0	5,657	(12,471)	5,473	(12,065)
21.5	5,714	(12,597)	5,530	(12,192)
22.0	5,769	(12,718)	5,584	(12,310)
22.5	5,819	(12,829)	5,634	(12,421)
23.0	5,868	(12,936)	5,682	(12,526)
23.5	5,912	(13,033)	5,726	(12,623)
24.0	5,954	(13,126)	5,767	(12,714)
24.5	5,993	(13,211)	5,806	(12,799)
25.0	6,028	(13,290)	5,841	(12,878)
25.5	6,062	(13,364)	5,874	(12,950)
26.0	6,091	(13,429)	5,903	(13,014)
26.5	6,117	(13,485)	5,929	(13,071)
27.0	**6,139**	**(13,534)**	**5,950**	**(13,118)**
27.5	6,155	(13,569)	5,966	(13,153)
28.0	6,165	(13,591)	5,976	(13,174)
28.5	6,168	(13,598)	5,979	(13,181)
29.0	6,171	(13,605)	5,982	(13,188)
29.5	6,167	(13,596)	5,978	(13,179)
30.0	6,166	(13,594)	5,977	(13,177)

Notes: EPF Jettison at 3-sigma qv ≤ 1,135 W/m^2 (360 Btu/ft^2-hr)
Parking Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 km (100 nmi)
Transfer Orbit Apogee Altitude = 35,786 km (19,323 nmi)
Argument of Perigee = 180°

Table 2.13-2 Atlas V 421 Geosynchronous Transfer Orbit Performance—PSW vs Orbit Inclination

Inclination, °	Payload Systems Weight, kg (lb) Atlas V 421			
	MRS		2.33-sigma GCS	
15.0	5,445	(12,003)	5,253	(11,580)
15.5	5,542	(12,218)	5,348	(11,791)
16.0	5,637	(12,429)	5,442	(11,999)
16.5	5,732	(12,637)	5,535	(12,203)
17.0	5,824	(12,840)	5,626	(12,404)
17.5	5,915	(13,039)	5,716	(12,601)
18.0	6,003	(13,234)	5,803	(12,793)
18.5	6,089	(13,423)	5,888	(12,980)
19.0	6,172	(13,608)	5,970	(13,163)
19.5	6,254	(13,787)	6,051	(13,339)
20.0	6,332	(13,959)	6,128	(13,510)
20.5	6,407	(14,126)	6,202	(13,674)
21.0	6,480	(14,286)	6,274	(13,831)
21.5	6,549	(14,438)	6,342	(13,982)
22.0	6,615	(14,583)	6,407	(14,125)
22.5	6,676	(14,718)	6,468	(14,260)
23.0	6,735	(14,847)	6,526	(14,386)
23.5	6,788	(14,965)	6,579	(14,504)
24.0	6,838	(15,074)	6,629	(14,614)
24.5	6,884	(15,176)	6,674	(14,713)
25.0	6,925	(15,266)	6,715	(14,803)
25.5	6,962	(15,348)	6,751	(14,883)
26.0	6,993	(15,418)	6,782	(14,953)
26.5	7,020	(15,476)	6,809	(15,011)
27.0	**7,042**	**(15,526)**	**6,830**	**(15,058)**
27.5	7,058	(15,561)	6,846	(15,094)
28.0	7,069	(15,585)	6,857	(15,117)
28.5	7,075	(15,598)	6,862	(15,129)
29.0	7,074	(15,597)	6,861	(15,127)
29.5	7,070	(15,587)	6,858	(15,119)
30.0	7,066	(15,578)	6,854	(15,110)

Note: EPF Jettison at 3-sigma qv ≤ 1,135 W/m^2 (360 Btu/ft^2-hr)
Parking Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 185 (100 nmi)
Transfer Orbit Perigee Altitude = 35,786 km (19,323 nmi)
Argument of Perigee = 180°

Table 2.14-2 Atlas V 431 Geosynchronous Transfer Orbit Performance—PSW vs Orbit Inclination

Inclination, °	Payload Systems Weight, kg (lb) Atlas V 431			
	MRS		2.33-sigma GCS	
12.0	5,390	(11,883)	5,185	(11,431)
12.5	5,504	(12,135)	5,297	(11,678)
13.0	5,617	(12,384)	5,409	(11,926)
13.5	5,731	(12,635)	5,521	(12,172)
14.0	5,844	(12,884)	5,632	(12,417)
14.5	5,956	(13,130)	5,743	(12,660)
15.0	6,066	(13,373)	5,852	(12,901)
15.5	6,176	(13,616)	5,960	(13,139)
16.0	6,284	(13,853)	6,067	(13,375)
16.5	6,390	(14,087)	6,172	(13,606)
17.0	6,494	(14,316)	6,275	(13,834)
17.5	6,596	(14,542)	6,376	(14,057)
18.0	6,696	(14,762)	6,475	(14,275)
18.5	6,793	(14,977)	6,571	(14,487)
19.0	6,888	(15,186)	6,665	(14,694)
19.5	6,980	(15,389)	6,756	(14,895)
20.0	7,069	(15,585)	6,844	(15,089)
20.5	7,155	(15,773)	6,929	(15,275)
21.0	7,237	(15,955)	7,010	(15,454)
21.5	7,315	(16,128)	7,087	(15,625)
22.0	7,390	(16,292)	7,161	(15,788)
22.5	7,461	(16,448)	7,231	(15,941)
23.0	7,527	(16,594)	7,296	(16,085)
23.5	7,589	(16,731)	7,357	(16,219)
24.0	7,646	(16,857)	7,413	(16,343)
24.5	7,698	(16,972)	7,464	(16,456)
25.0	7,745	(17,076)	7,510	(16,558)
25.5	7,787	(17,168)	7,551	(16,648)
26.0	7,824	(17,248)	7,587	(16,726)
26.5	7,854	(17,316)	7,616	(16,791)
27.0	**7,879**	**(17,371)**	**7,640**	**(16,844)**
27.5	7,897	(17,410)	7,658	(16,883)
28.0	7,908	(17,435)	7,669	(16,908)
28.5	7,914	(17,448)	7,674	(16,919)
29.0	7,911	(17,441)	7,672	(16,914)
29.5	7,906	(17,430)	7,667	(16,903)
30.0	7,903	(17,423)	7,664	(16,896)

Note: EPF Jettison at 3-sigma qv < 1,135 W/m^2 (360 Btu/ft^2-hr)
Parking Orbit Perigee Altitude > 167 km (90 nmi)
Transfer Orbit Perigee Altitude > 185 km (100 nmi)
Transfer Orbit Apogee Altitude = 35,786 km (19,323 nmi)
Argument of Perigee = 180°

Table 2.15-2a Atlas V 501, 511, 521 Geotransfer Orbit Performance—PSW vs Inclination

	Payload Systems Weight, kg (lb)											
	Atlas V 501				Atlas V 511				Atlas V 521			
Inclination, °	**MRS**		**2.33-sigma GCS**		**MRS**		**2.33-sigma GCS**		**MRS**		**2.33-sigma GCS**	
18.0	3,475	(7,660)	3,344	(7,372)	4,437	(9,783)	4,257	(9,384)	5,367	(11,832)	5,167	(11,392)
18.5	3,530	(7,782)	3,398	(7,491)	4,518	(9,961)	4,335	(9,557)	5,459	(12,036)	5,257	(11,590)
19.0	3,584	(7,901)	3,450	(7,606)	4,598	(10,136)	4,412	(9,727)	5,550	(12,236)	5,345	(11,785)
19.5	3,637	(8,018)	3,502	(7,721)	4,677	(10,311)	4,489	(9,896)	5,640	(12,434)	5,433	(11,977)
20.0	3,686	(8,126)	3,550	(7,826)	4,751	(10,474)	4,560	(10,053)	5,723	(12,618)	5,514	(12,156)
20.5	3,732	(8,228)	3,595	(7,925)	4,821	(10,630)	4,629	(10,204)	5,803	(12,793)	5,591	(12,327)
21.0	3,776	(8,326)	3,638	(8,020)	4,890	(10,781)	4,695	(10,350)	5,880	(12,962)	5,666	(12,491)
21.5	3,819	(8,420)	3,680	(8,112)	4,957	(10,929)	4,760	(10,494)	5,954	(13,127)	5,739	(12,652)
22.0	3,859	(8,507)	3,718	(8,197)	5,020	(11,067)	4,821	(10,628)	6,024	(13,280)	5,806	(12,801)
22.5	3,905	(8,610)	3,763	(8,297)	5,092	(11,227)	4,891	(10,783)	6,105	(13,459)	5,885	(12,974)
23.0	3,944	(8,694)	3,801	(8,379)	5,154	(11,363)	4,951	(10,915)	6,173	(13,609)	5,951	(13,120)
23.5	3,977	(8,767)	3,833	(8,449)	5,210	(11,485)	5,004	(11,033)	6,233	(13,741)	6,010	(13,249)
24.0	4,006	(8,832)	3,861	(8,513)	5,260	(11,597)	5,054	(11,141)	6,288	(13,862)	6,063	(13,366)
24.5	4,040	(8,907)	3,895	(8,586)	5,318	(11,724)	5,109	(11,264)	6,350	(13,999)	6,124	(13,500)
25.0	4,064	(8,961)	3,918	(8,638)	5,363	(11,822)	5,152	(11,359)	6,397	(14,103)	6,169	(13,601)
25.5	4,085	(9,006)	3,938	(8,682)	5,402	(11,910)	5,191	(11,444)	6,438	(14,193)	6,209	(13,689)
26.0	4,100	(9,038)	3,953	(8,714)	5,435	(11,981)	5,222	(11,513)	6,470	(14,264)	6,240	(13,758)
26.5	4,113	(9,068)	3,966	(8,743)	5,466	(12,050)	5,252	(11,579)	6,500	(14,331)	6,270	(13,823)
27.0	**4,118**	**(9,078)**	**3,970**	**(8,752)**	**5,485**	**(12,092)**	**5,270**	**(11,618)**	**6,517**	**(14,367)**	**6,285**	**(13,856)**
27.5	4,124	(9,092)	3,977	(8,767)	5,493	(12,111)	5,279	(11,638)	6,527	(14,390)	6,296	(13,880)
28.0	4,127	(9,098)	3,979	(8,773)	5,497	(12,119)	5,283	(11,646)	6,531	(14,399)	6,300	(13,889)
28.5	4,124	(9,092)	3,977	(8,767)	5,493	(12,111)	5,279	(11,638)	6,527	(14,390)	6,296	(13,880)
29.0	4,118	(9,078)	3,970	(8,753)	5,485	(12,091)	5,271	(11,620)	6,517	(14,367)	6,286	(13,858)
29.5	4,102	(9,044)	3,956	(8,720)	5,464	(12,047)	5,251	(11,576)	6,492	(14,313)	6,262	(13,806)
30.0	4,095	(9,029)	3,949	(8,706)	5,455	(12,026)	5,242	(11,556)	6,481	(14,289)	6,252	(13,782)

Note: 5-m Short PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 Btu/ft^2-hr)
Parking Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Apogee Altitude = 35,786 km (19,323 nmi)
Argument of Perigee = 180°

Table 2.15-2b Atlas V 531, 541, 551 Geotransfer Orbit Performance—PSW vs Inclination

	Payload Systems Weight, kg (lb)											
	Atlas V 531				Atlas V 541				Atlas V 551			
Inclination, °	**MRS**		**2.33-sigma GCS**		**MRS**		**2.33-sigma GCS**		**MRS**		**2.33-sigma GCS**	
18.0	6,196	(13,659)	5,974	(13,171)	6,862	(15,128)	6,621	(14,598)	7,457	(16,440)	7,194	(15,860)
18.5	6,299	(13,888)	6,075	(13,393)	6,977	(15,381)	6,733	(14,844)	7,582	(16,715)	7,315	(16,128)
19.0	6,401	(14,111)	6,174	(13,611)	7,089	(15,629)	6,843	(15,086)	7,704	(16,984)	7,434	(16,390)
19.5	6,501	(14,333)	6,272	(13,827)	7,200	(15,874)	6,951	(15,325)	7,825	(17,250)	7,552	(16,650)
20.0	6,594	(14,537)	6,362	(14,026)	7,303	(16,101)	7,052	(15,546)	7,936	(17,496)	7,661	(16,890)
20.5	6,682	(14,732)	6,448	(14,216)	7,401	(16,316)	7,147	(15,756)	8,042	(17,730)	7,765	(17,118)
21.0	6,767	(14,919)	6,531	(14,398)	7,495	(16,523)	7,239	(15,958)	8,145	(17,956)	7,864	(17,338)
21.5	6,850	(15,101)	6,612	(14,576)	7,586	(16,725)	7,328	(16,155)	8,244	(18,175)	7,961	(17,552)
22.0	6,926	(15,269)	6,686	(14,740)	7,671	(16,911)	7,410	(16,337)	8,336	(18,378)	8,051	(17,750)
22.5	7,016	(15,467)	6,773	(14,932)	7,770	(17,130)	7,507	(16,550)	8,444	(18,615)	8,156	(17,981)
23.0	7,090	(15,631)	6,846	(15,093)	7,853	(17,312)	7,588	(16,728)	8,533	(18,813)	8,244	(18,174)
23.5	7,155	(15,775)	6,910	(15,233)	7,925	(17,471)	7,658	(16,884)	8,612	(18,986)	8,320	(18,343)
24.0	7,214	(15,905)	6,967	(15,360)	7,990	(17,615)	7,722	(17,024)	8,683	(19,142)	8,390	(18,496)
24.5	7,282	(16,054)	7,033	(15,506)	8,065	(17,781)	7,795	(17,186)	8,764	(19,322)	8,469	(18,672)
25.0	7,332	(16,165)	7,082	(15,613)	8,121	(17,903)	7,849	(17,305)	8,825	(19,455)	8,528	(18,801)
25.5	7,375	(16,260)	7,124	(15,706)	8,169	(18,008)	7,896	(17,408)	8,877	(19,570)	8,579	(18,913)
26.0	7,408	(16,332)	7,156	(15,777)	8,205	(18,089)	7,932	(17,487)	8,916	(19,657)	8,618	(18,998)
26.5	7,439	(16,400)	7,187	(15,844)	8,239	(18,164)	7,965	(17,560)	8,953	(19,739)	8,654	(19,079)
27.0	**7,454**	**(16,433)**	**7,200**	**(15,873)**	**8,255**	**(18,200)**	**7,980**	**(17,593)**	**8,971**	**(19,778)**	**8,670**	**(19,114)**
27.5	7,466	(16,459)	7,213	(15,901)	8,268	(18,229)	7,994	(17,624)	8,985	(19,809)	8,685	(19,148)
28.0	7,470	(16,470)	7,217	(15,911)	8,274	(18,240)	7,999	(17,635)	8,991	(19,822)	8,691	(19,160)
28.5	7,466	(16,459)	7,213	(15,901)	8,268	(18,229)	7,994	(17,624)	8,985	(19,809)	8,685	(19,148)
29.0	7,454	(16,432)	7,201	(15,875)	8,255	(18,199)	7,981	(17,595)	8,971	(19,777)	8,671	(19,117)
29.5	7,426	(16,372)	7,174	(15,816)	8,225	(18,132)	7,951	(17,530)	8,938	(19,704)	8,639	(19,045)
30.0	7,413	(16,344)	7,162	(15,789)	8,211	(18,101)	7,938	(17,500)	8,922	(19,671)	8,624	(19,012)

Note: 5-m Short PLF Jettison at 3-sigma qV ≤ 1,135 W/m^2 (360 Btu/ft^2-hr)
Parking Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Perigee Altitude ≥ 167 km (90 nmi)
Transfer Orbit Apogee Altitude = 35,786 km (19,323 nmi)
Argument of Perigee = 180°

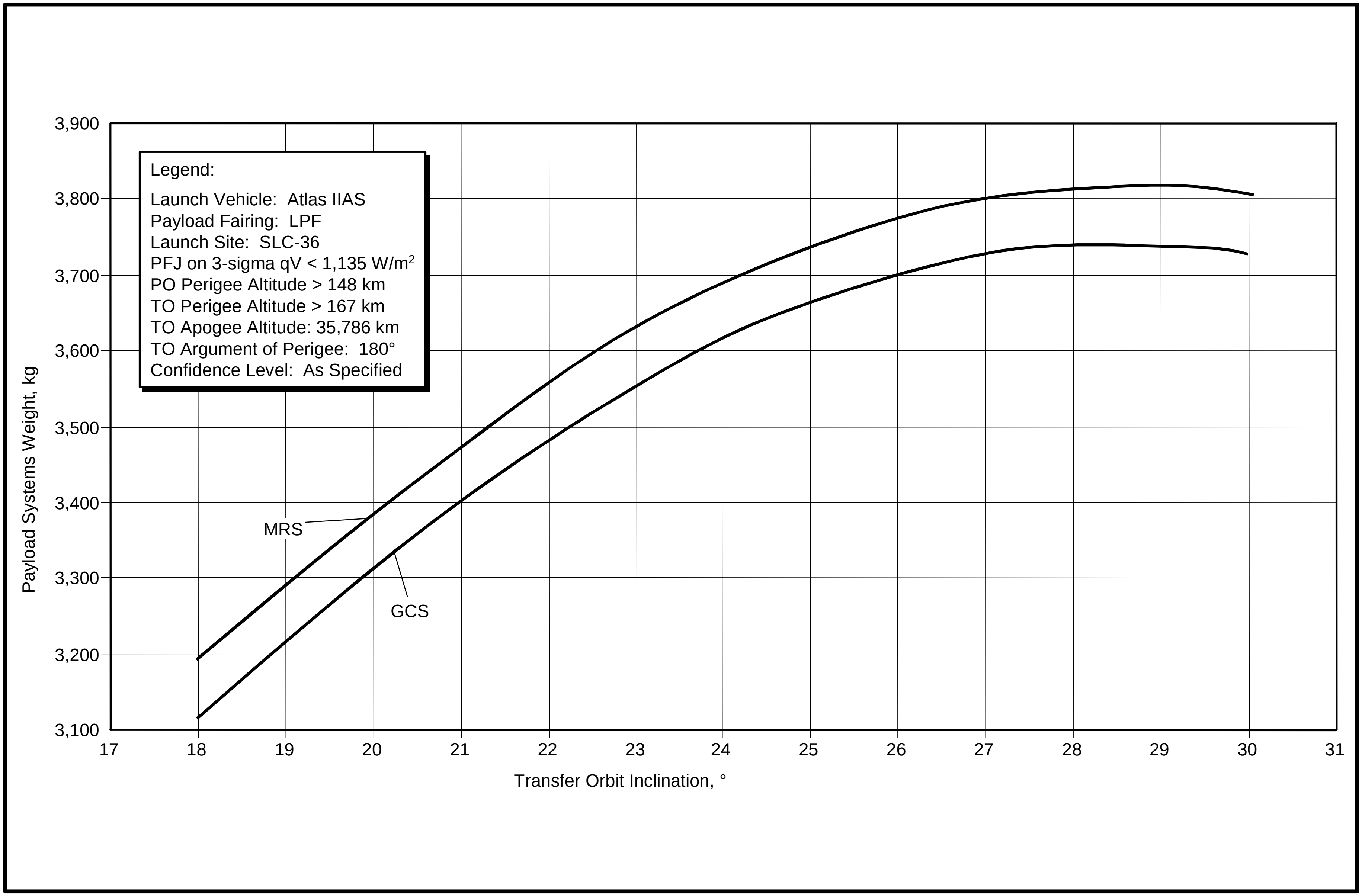


Figure 2.7-3 Atlas IIAS Reduced Inclination Performance to Geotransfer Orbit

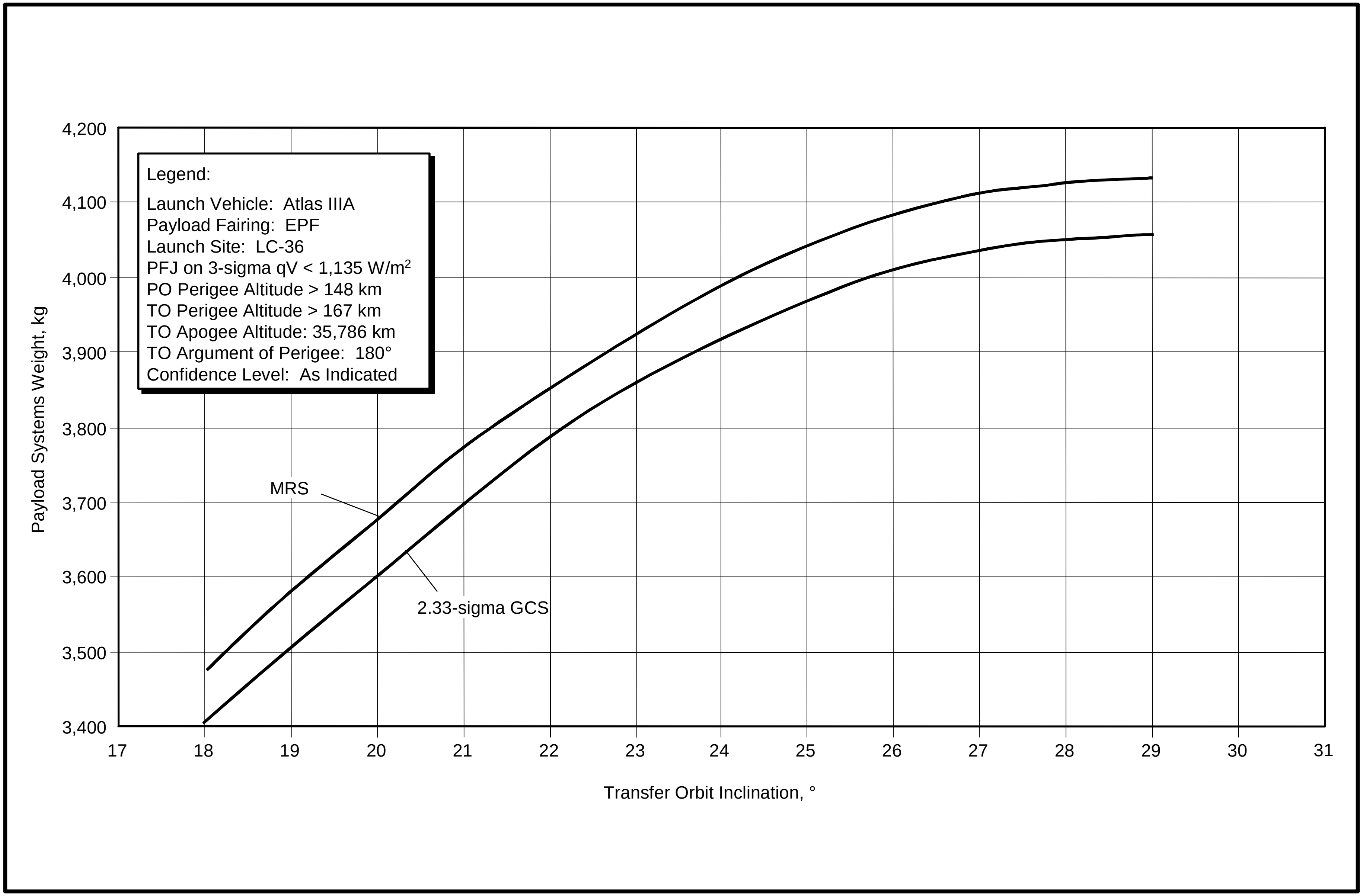


Figure 2.8-3 Atlas IIIA Reduced Inclination Performance to Geotransfer Orbit

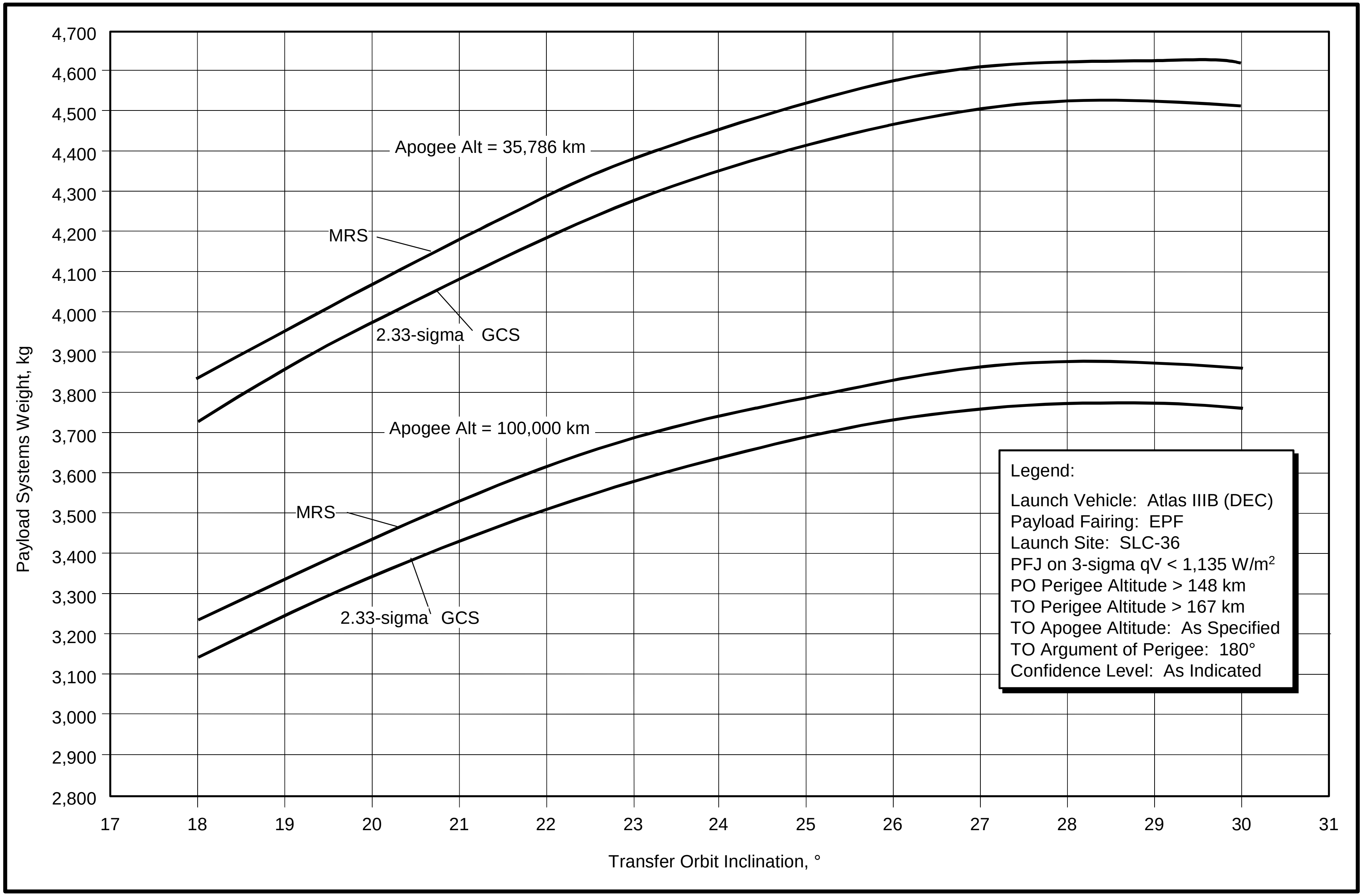


Figure 2.9-3 Atlas IIIB (DEC) Reduced Inclination Performance to Geotransfer Orbit

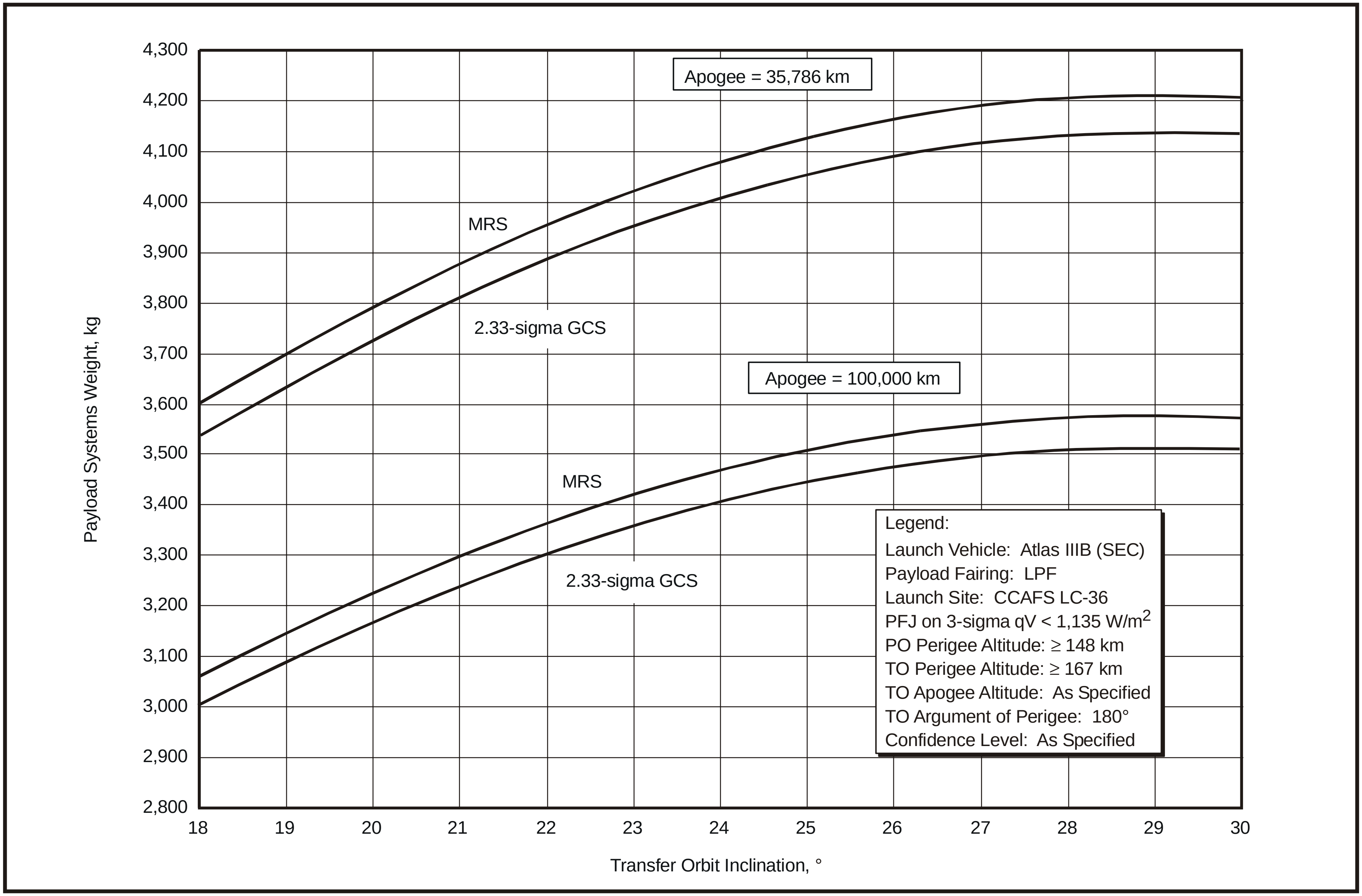


Figure 2.10-3 Atlas IIIB (SEC) CCAFS Reduced Inclination to Geotransfer Orbit

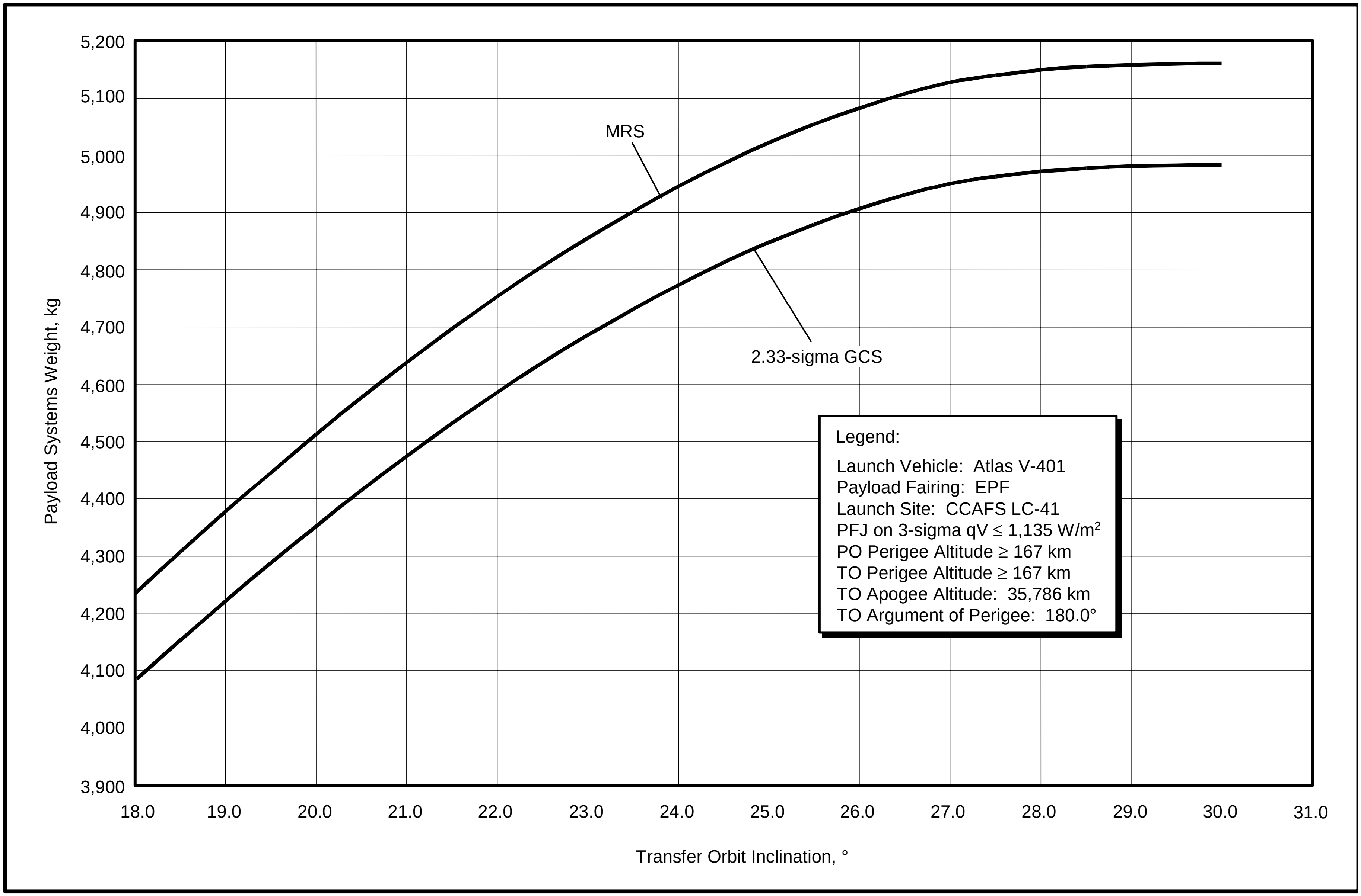


Figure 2.11-3 Atlas V 401 Reduced Inclination Performance to Geotransfer Orbit

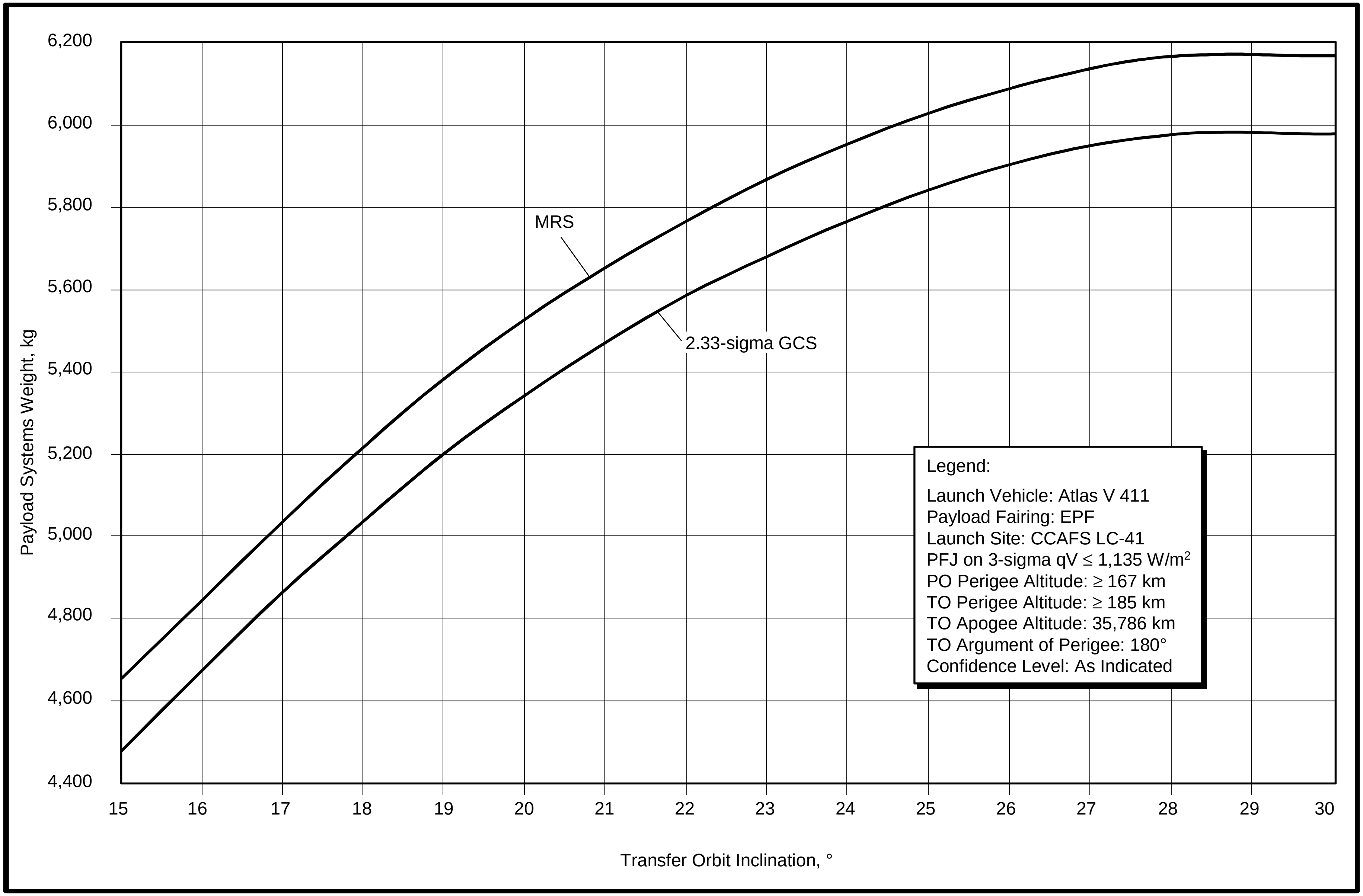


Figure 2.12-3 Atlas V 411 Reduced Inclination Performance to Geotransfer Orbit

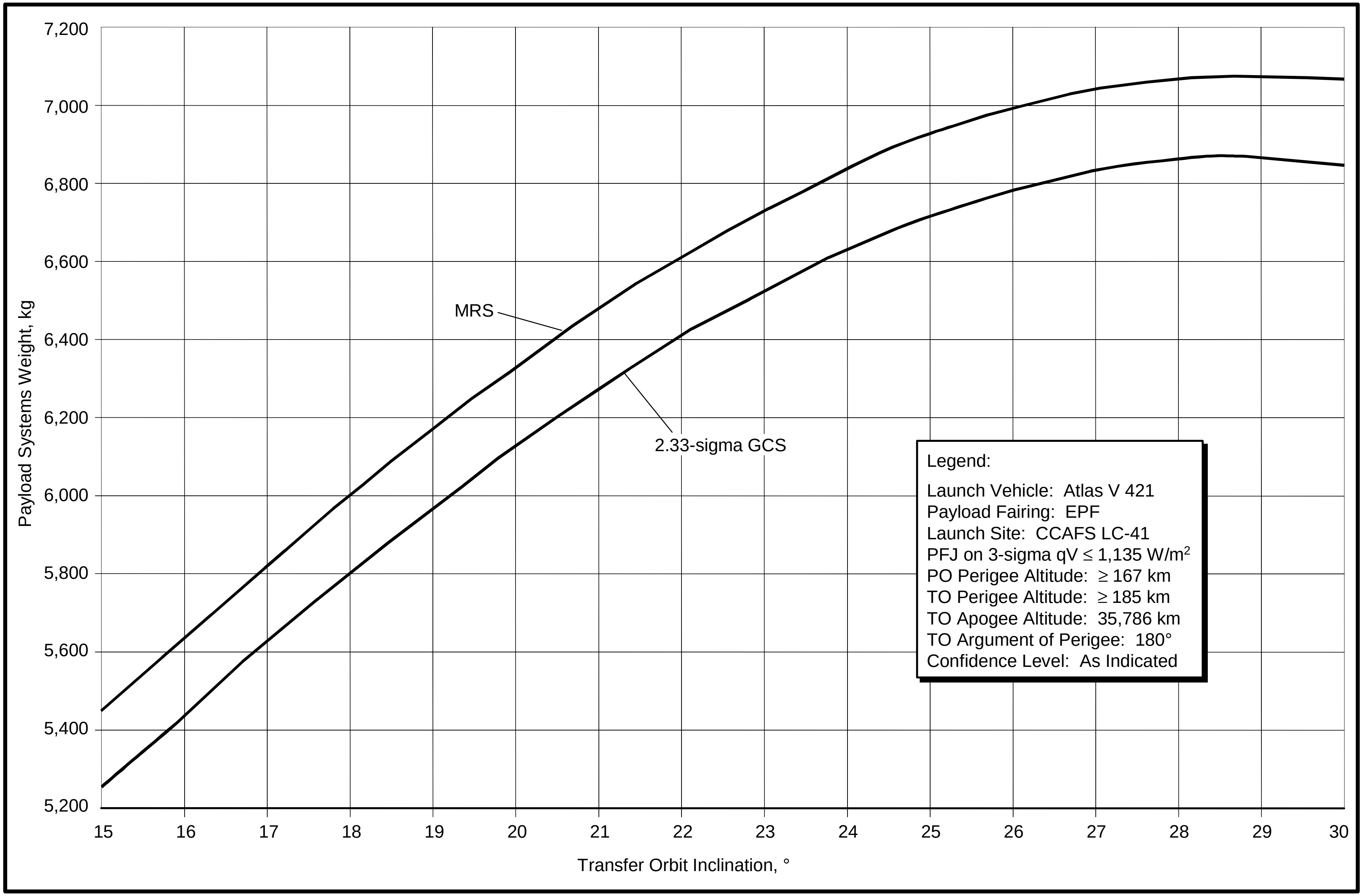


Figure 2.13-3 Atlas V 421 Reduced Inclination Performance to Geotransfer Orbit

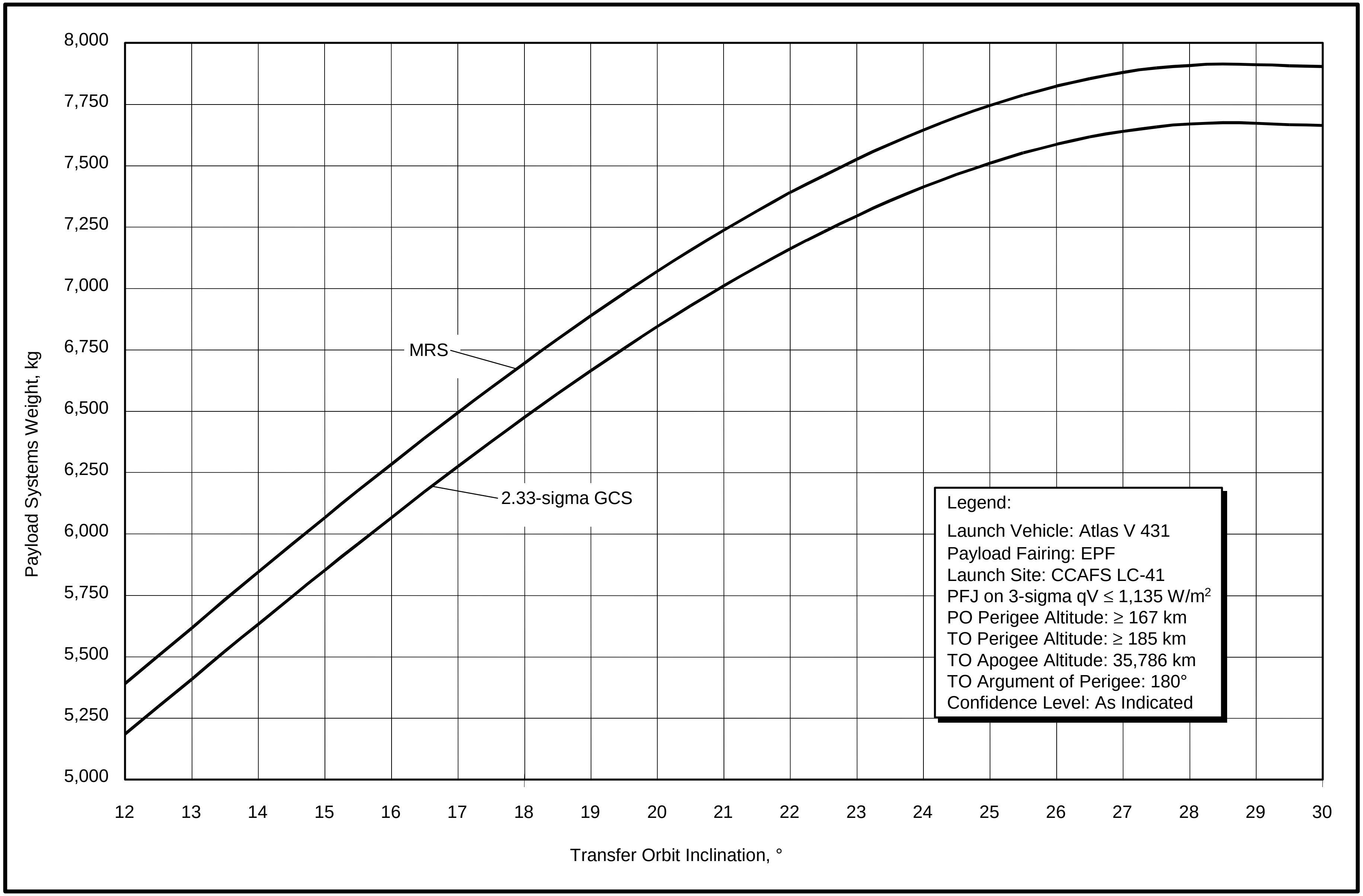


Figure 2.14-3 Atlas V 431 Reduced Inclination Performance to Geotransfer Orbit

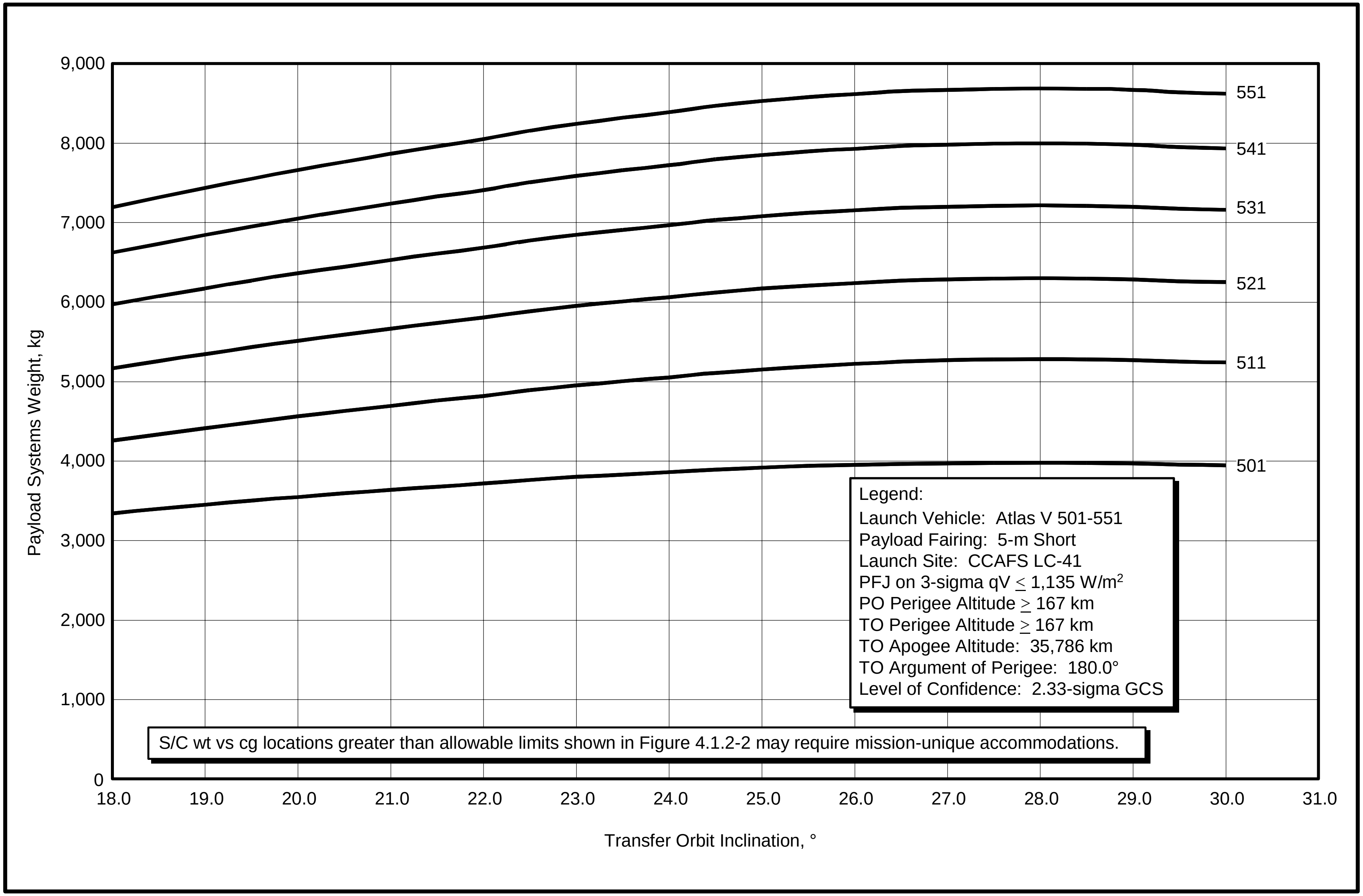


Figure 2.15-3a Atlas V 501-551 Reduced Inclination Performance to Geotransfer Orbit (GCS)

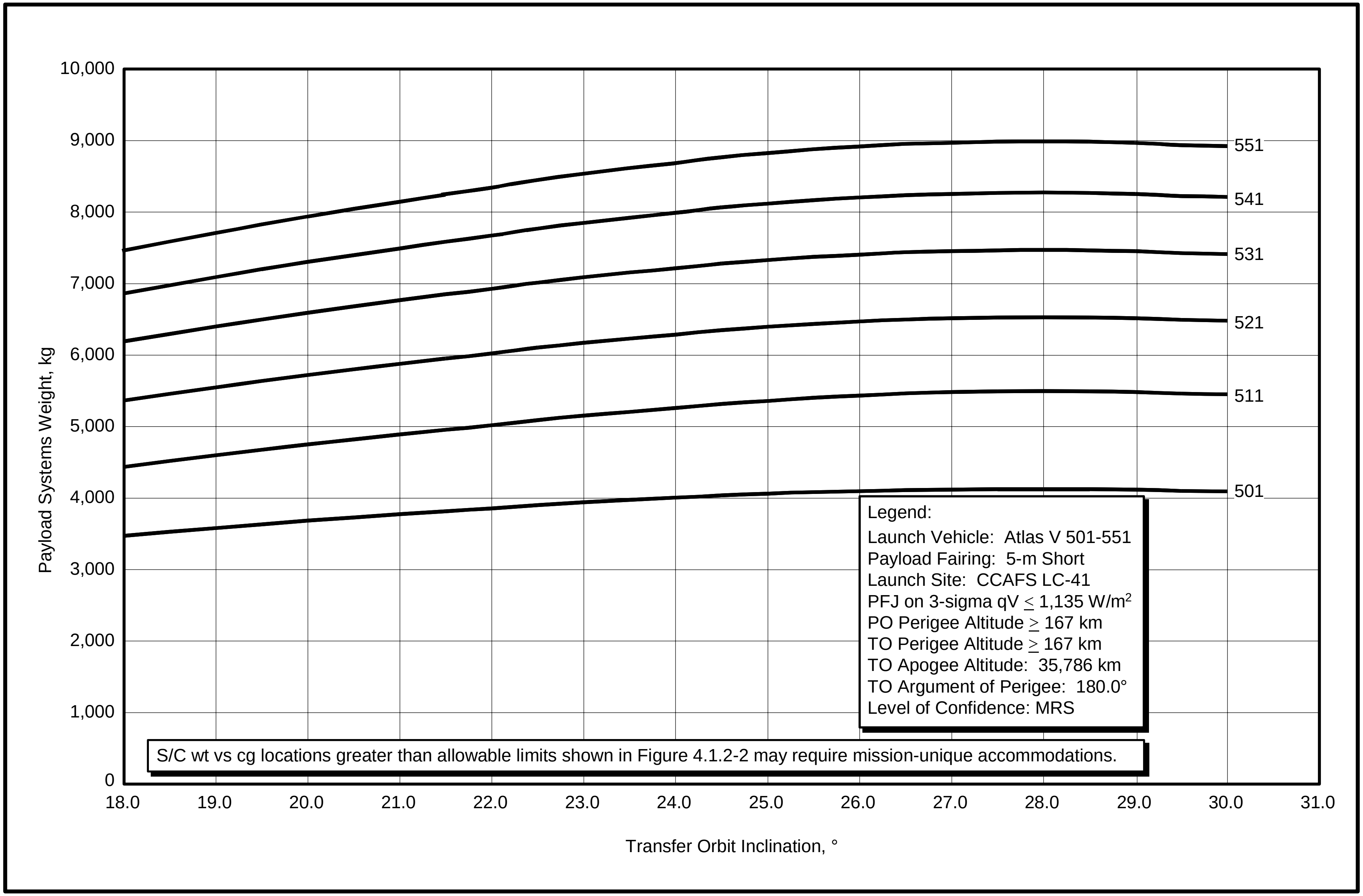


Figure 2.15-3b Atlas V 501-551 Reduced Inclination Performance to Geotransfer Orbit (MRS)

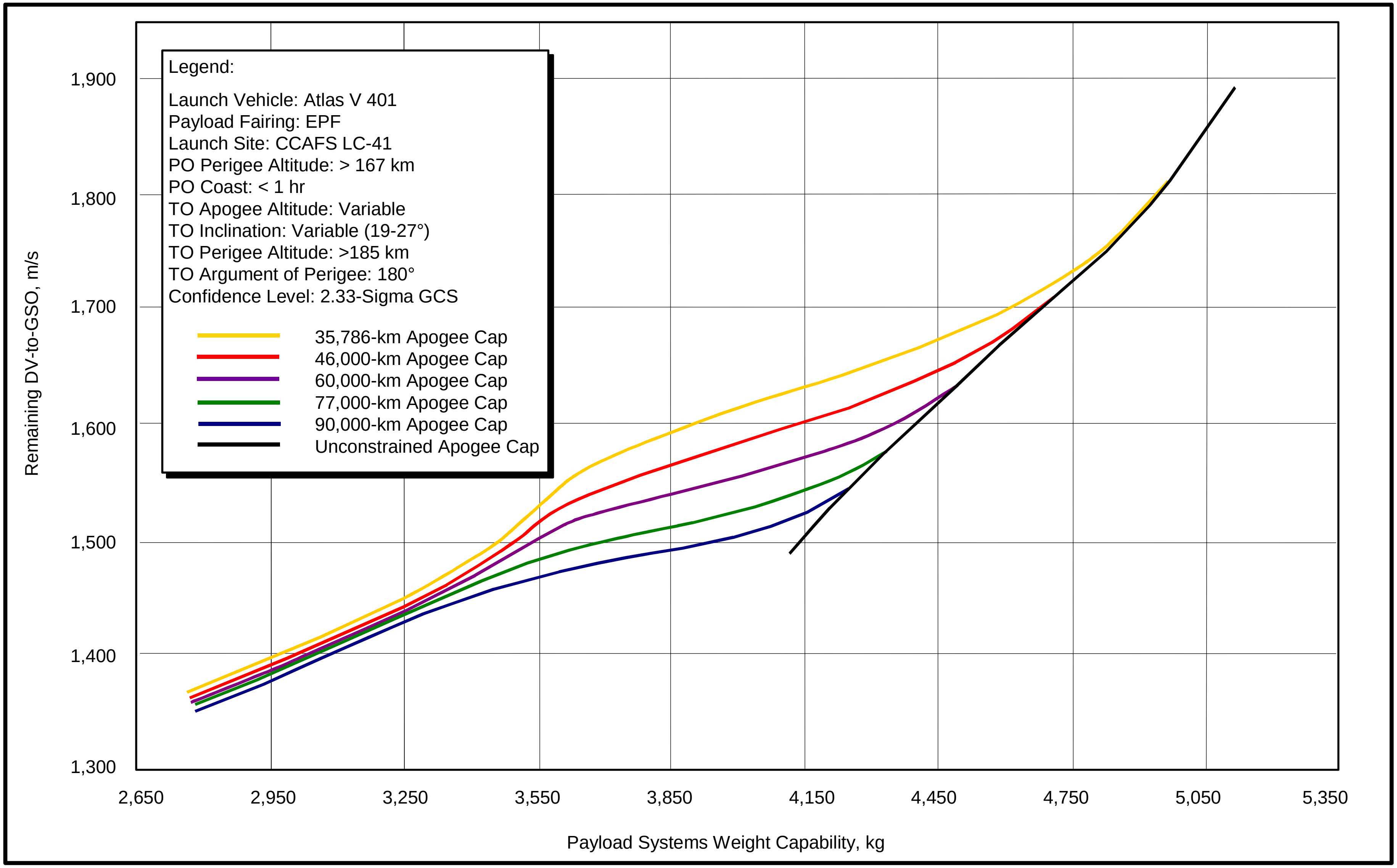


Figure 2.11-4 Atlas V 401 Minimum Delta-V to Geosynchronous Orbit

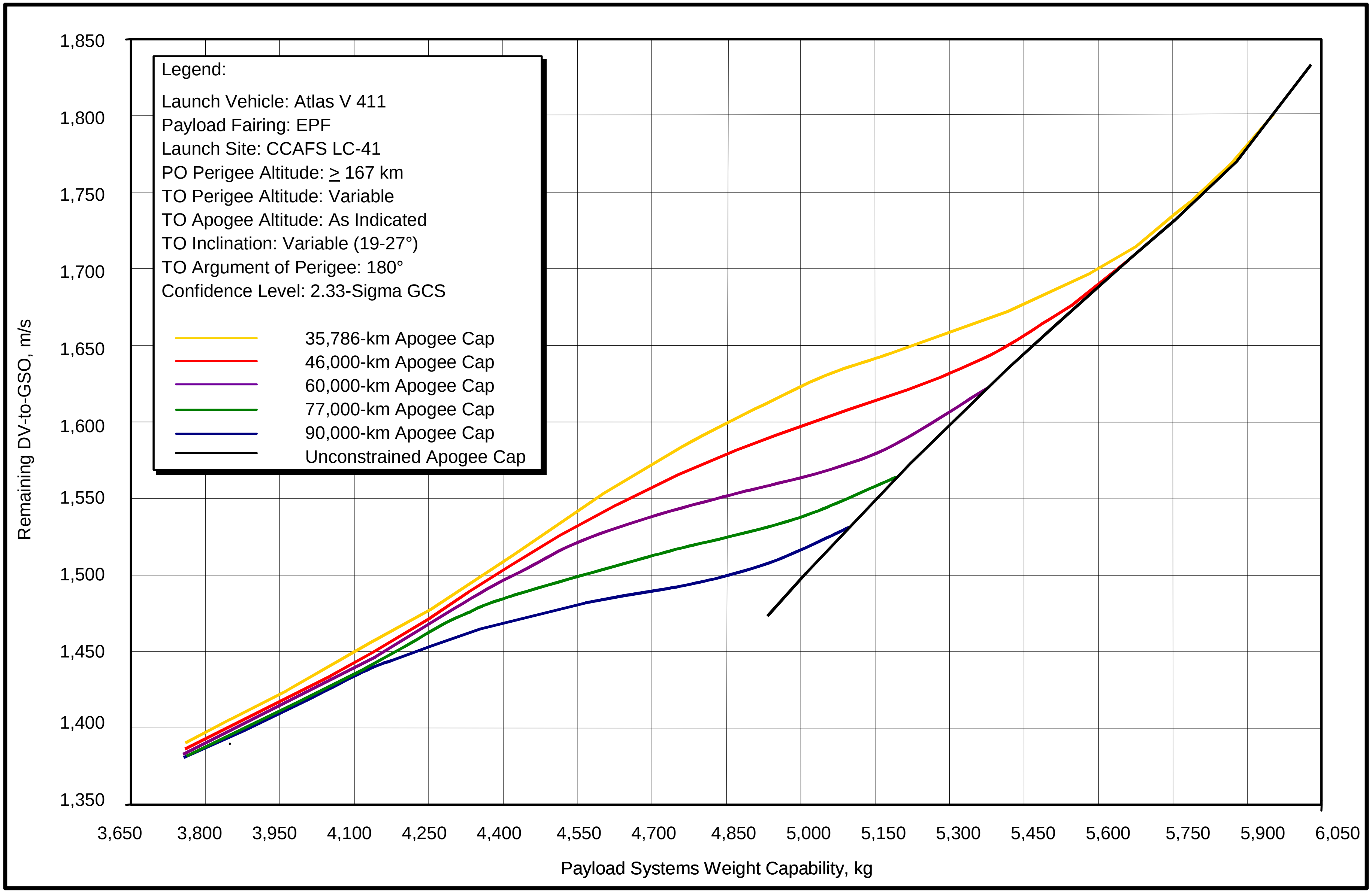


Figure 2.12-4 Atlas V 411 Minimum Delta-V to Geosynchronous Orbit

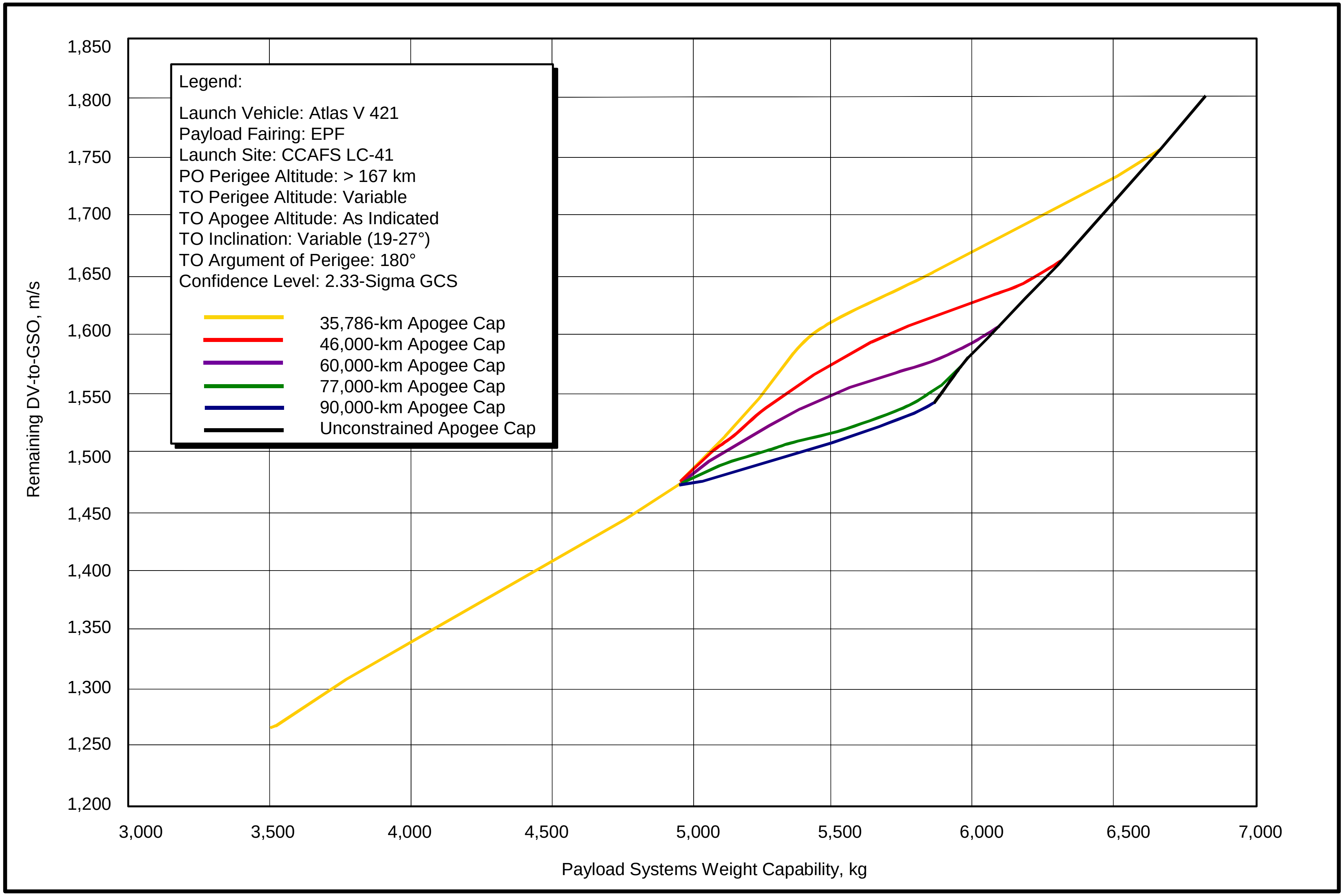


Figure 2.13-4 Atlas V 421 Minimum Delta-V to Geosynchronous Orbit

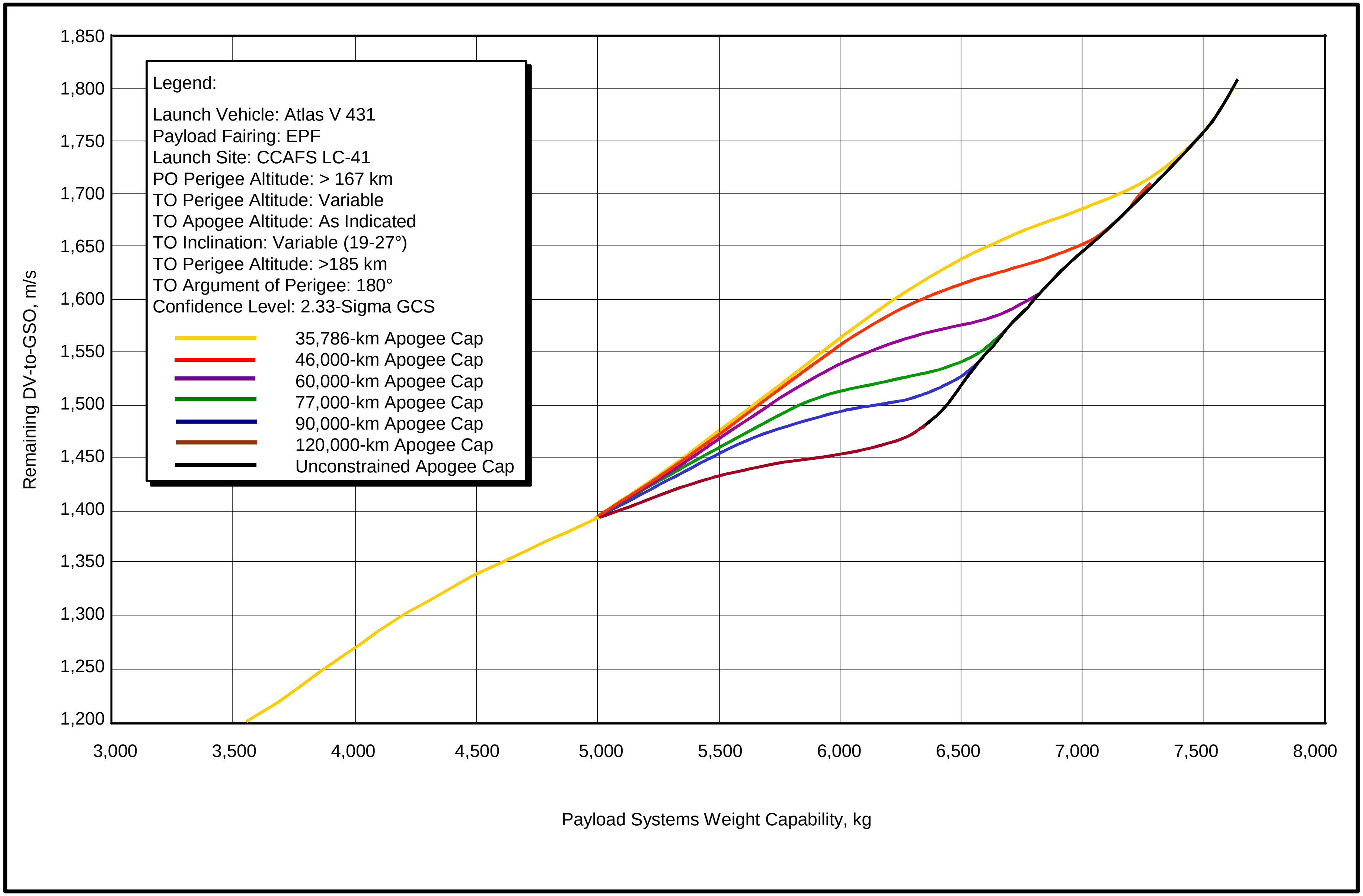


Figure 2.14-4 Atlas V 431 Minimum Delta-V to Geosynchronous Orbit

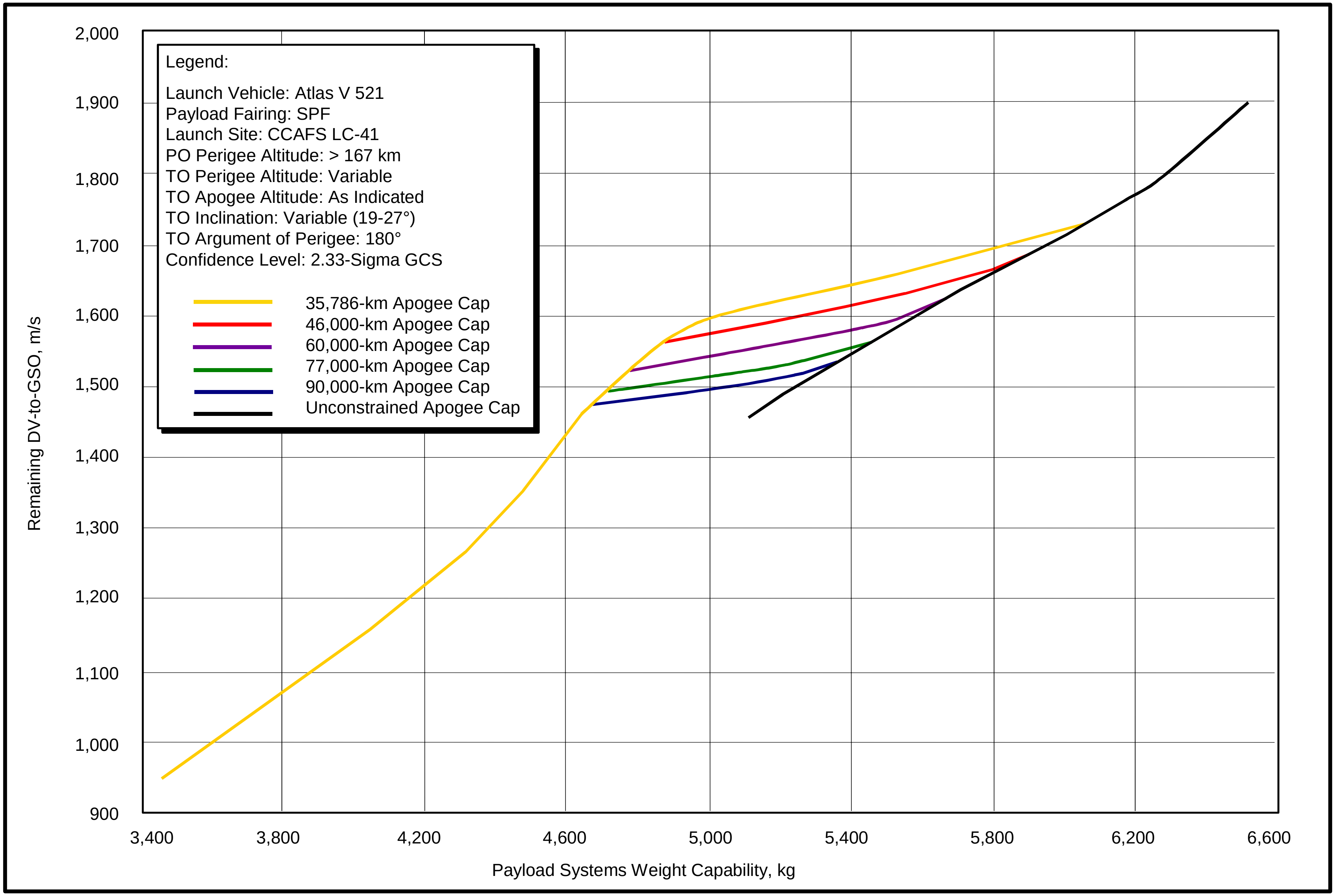


Figure 2.17-4 Atlas V 521 Minimum Delta-V to Geosynchronous Orbit

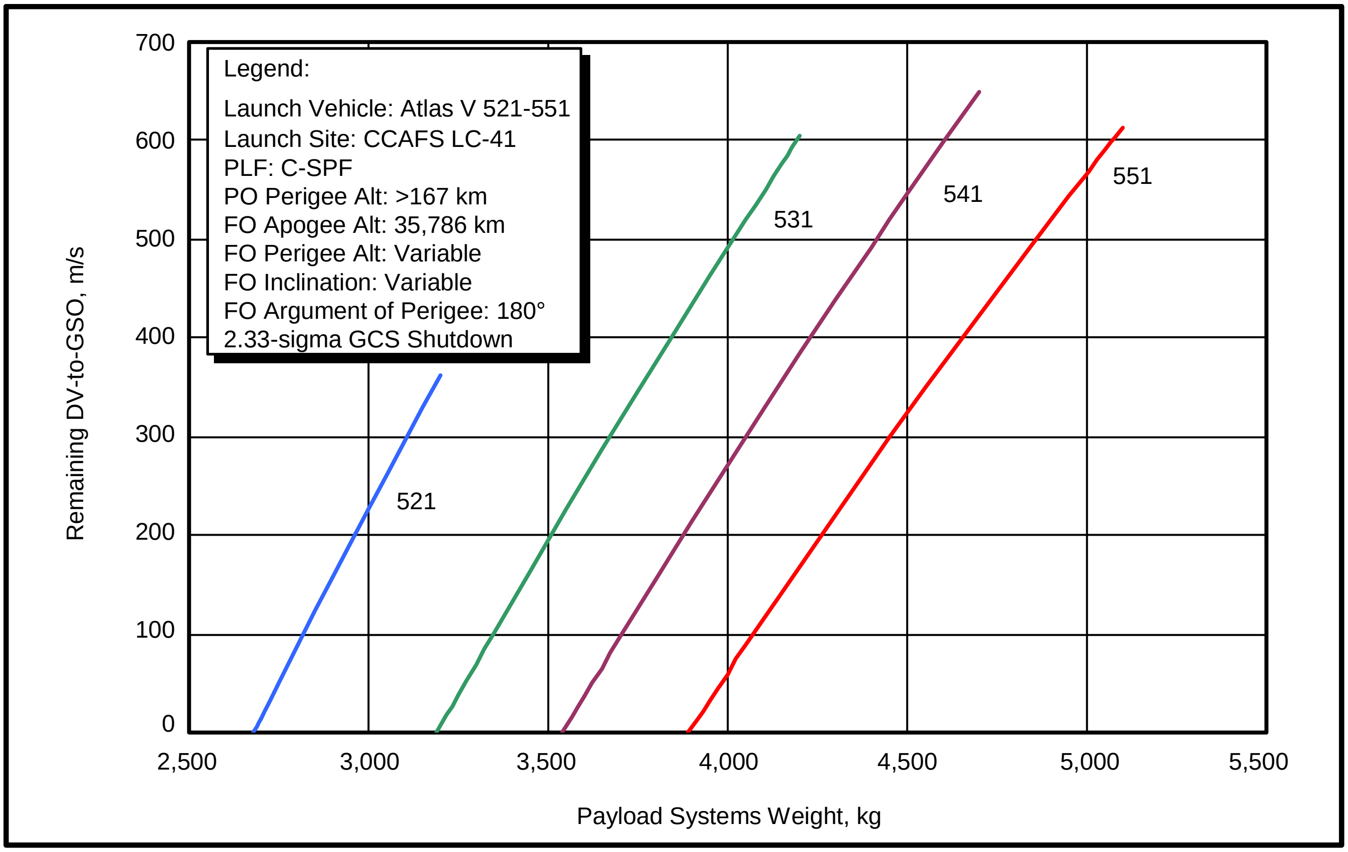


Figure 2.15-5 Atlas V 521-551 3-Burn Delta-V to Geosynchronous Orbit

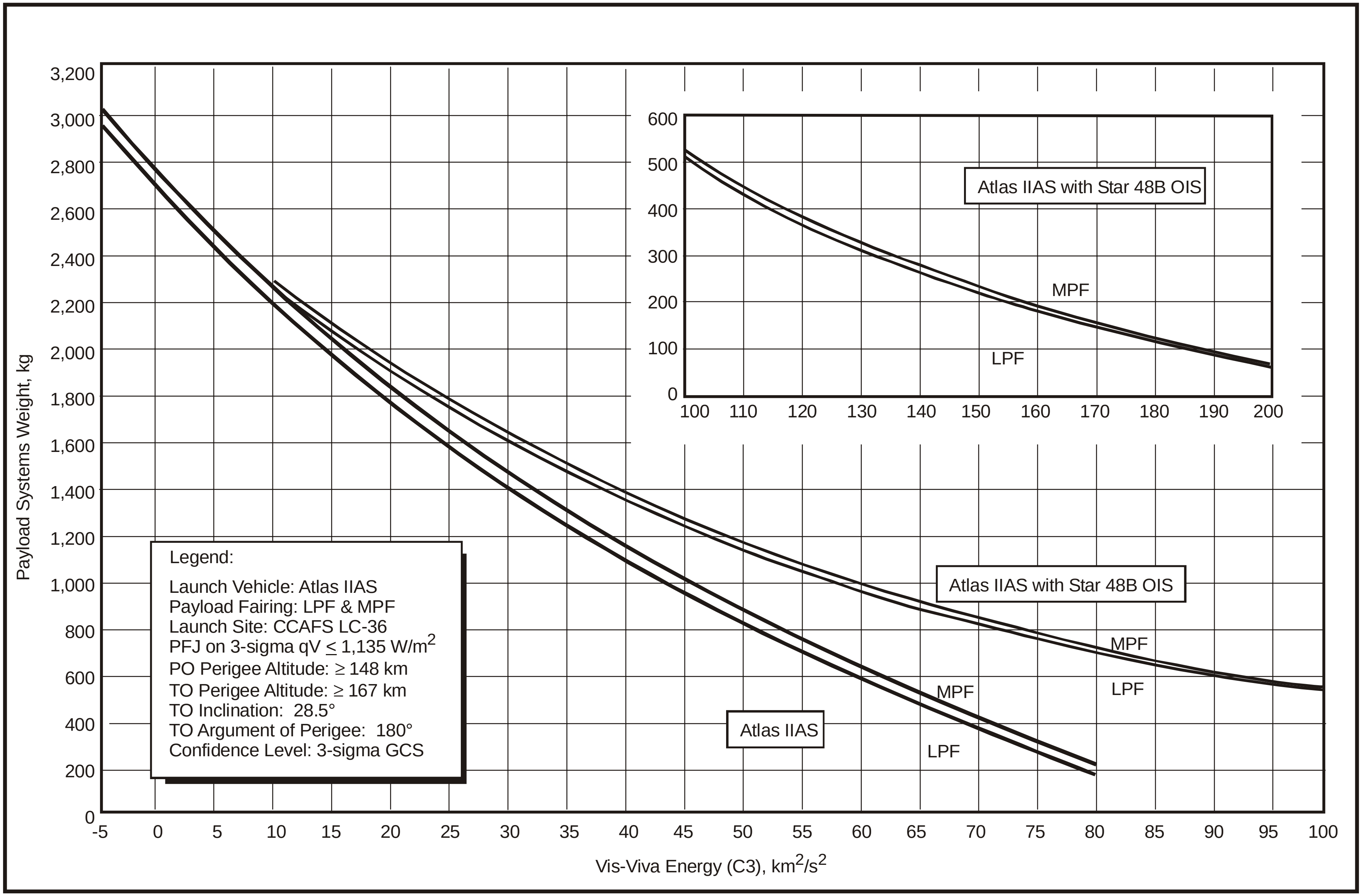


Figure 2.7-6 Atlas IIAS CCAFS Earth-Escape Performance

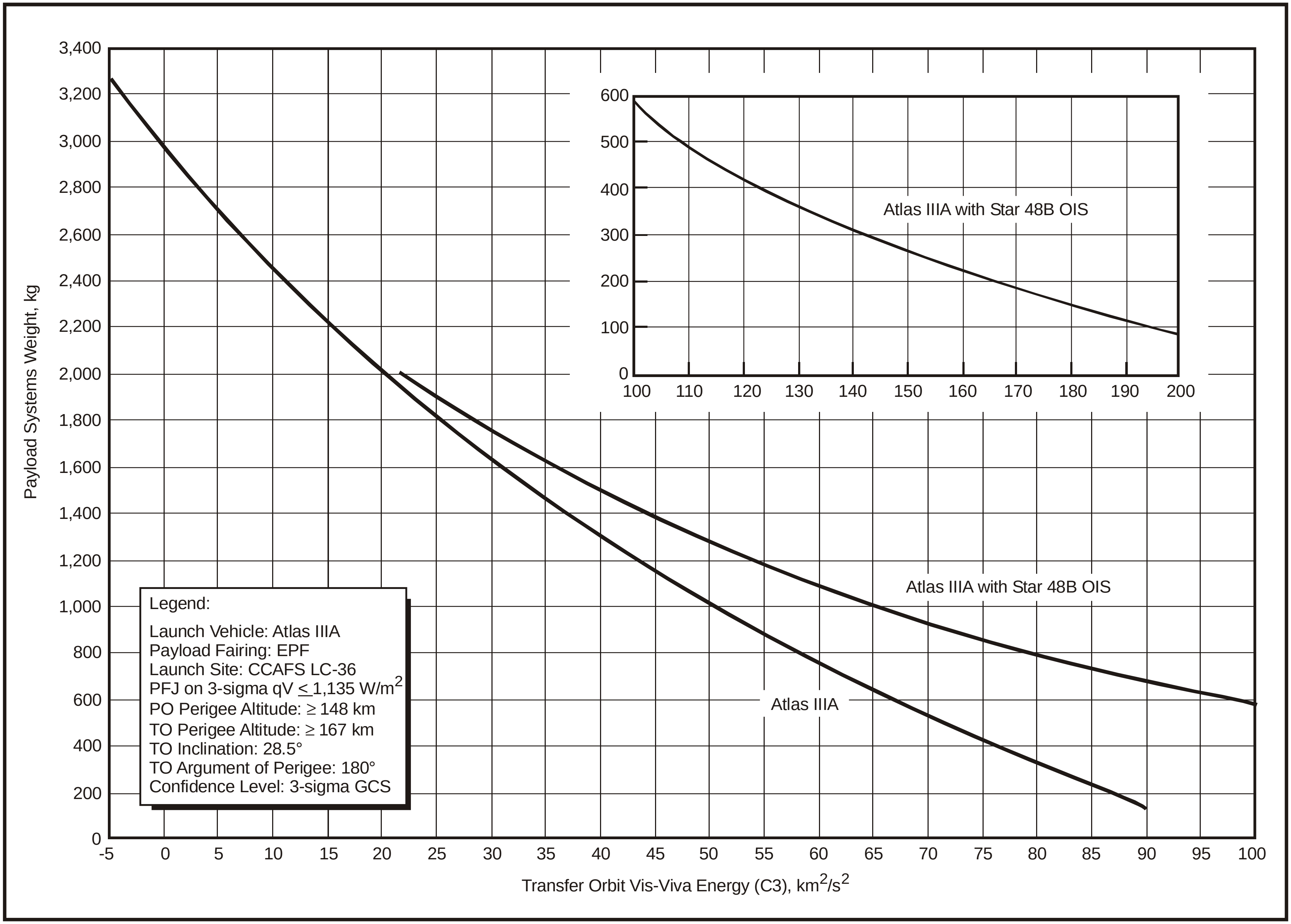


Figure 2.8-6 Atlas IIIA CCAFS Earth-Escape Performance

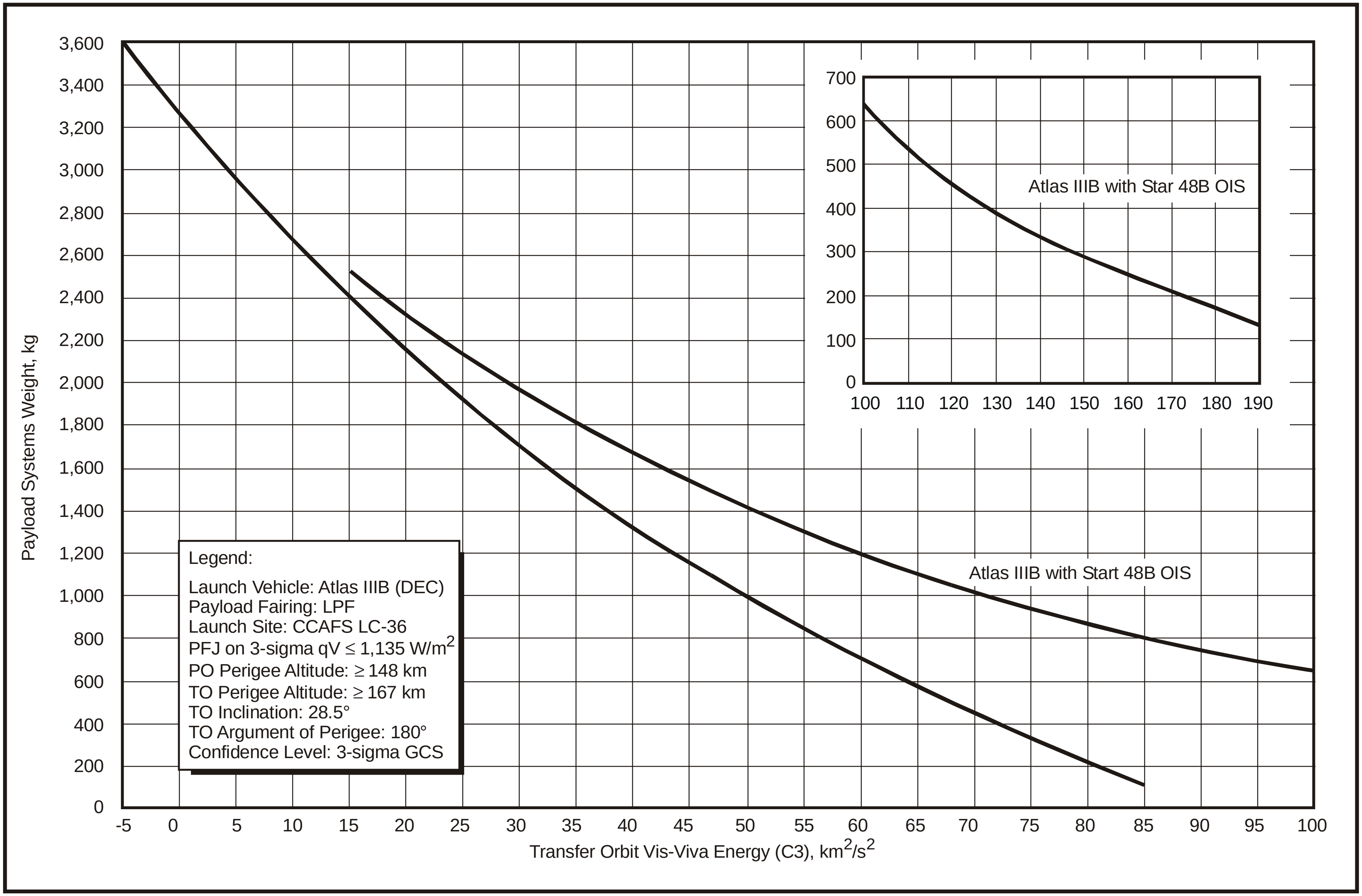


Figure 2.9-6 Atlas IIIB (DEC) CCAFS Earth-Escape Performance

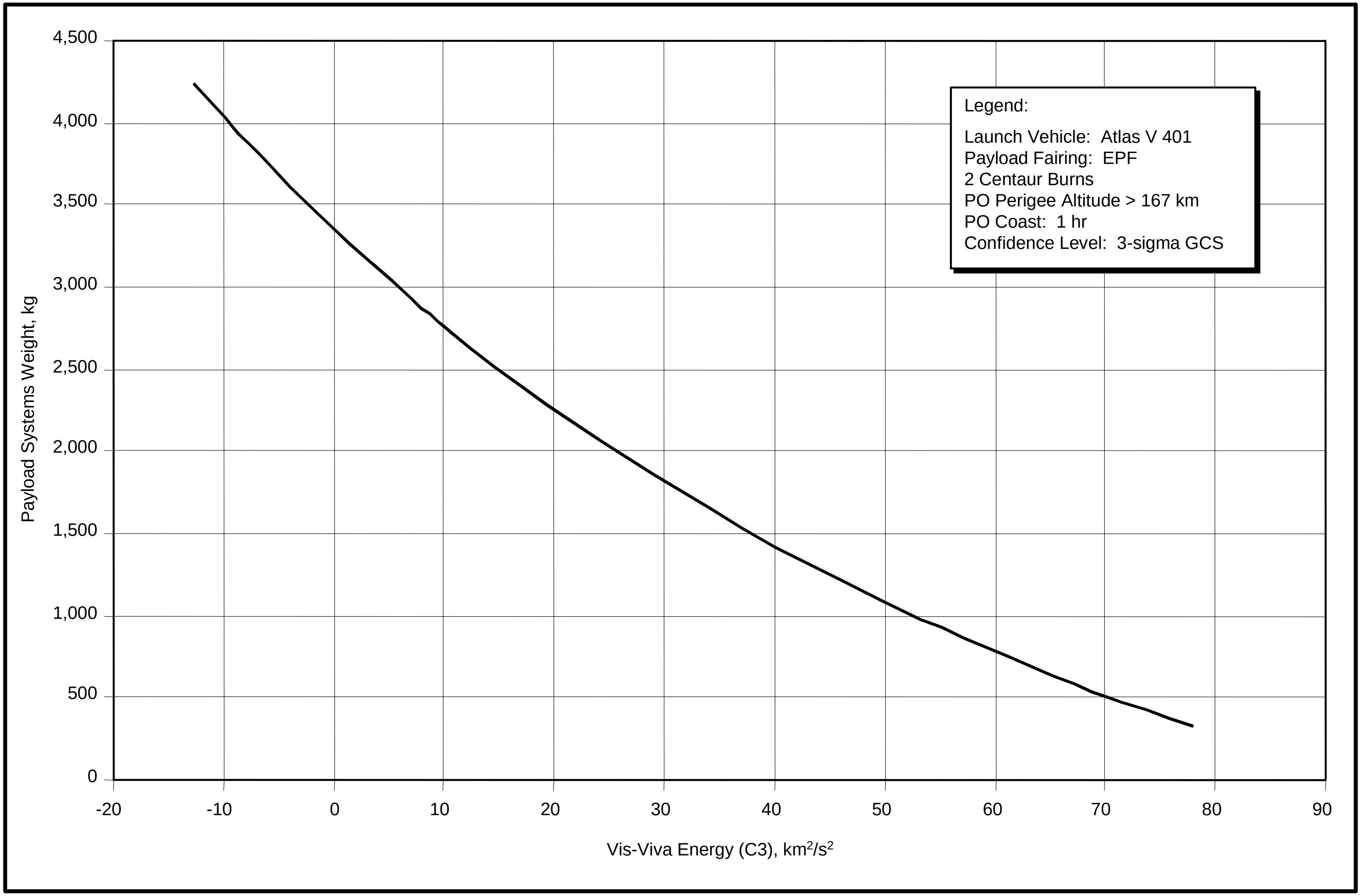


Figure 2.11-6 Atlas V 401 Earth Escape Performance (C3 Curves)

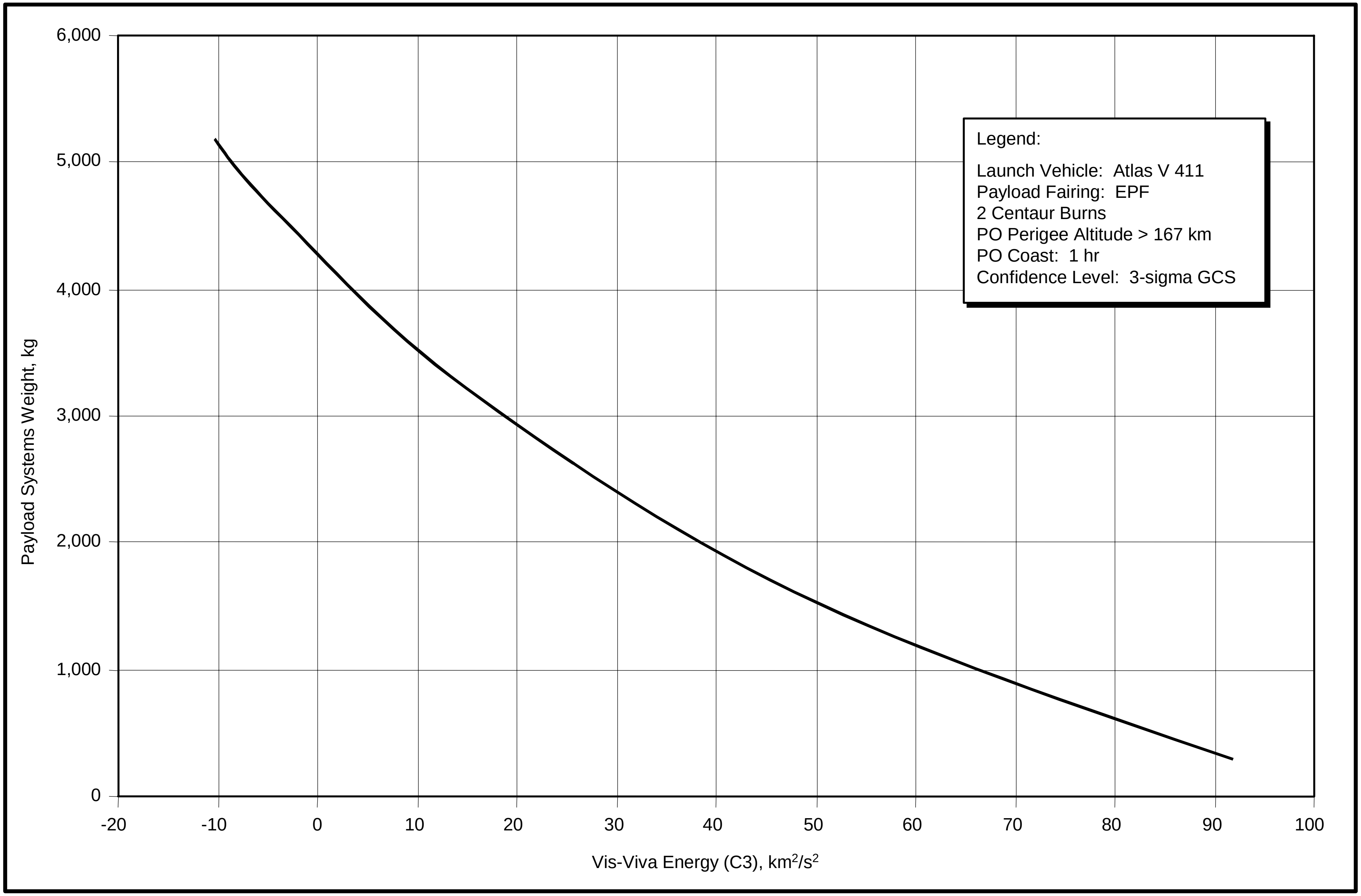


Figure 2.12-6 Atlas V 411 Earth Escape Performance (C3 Curves)

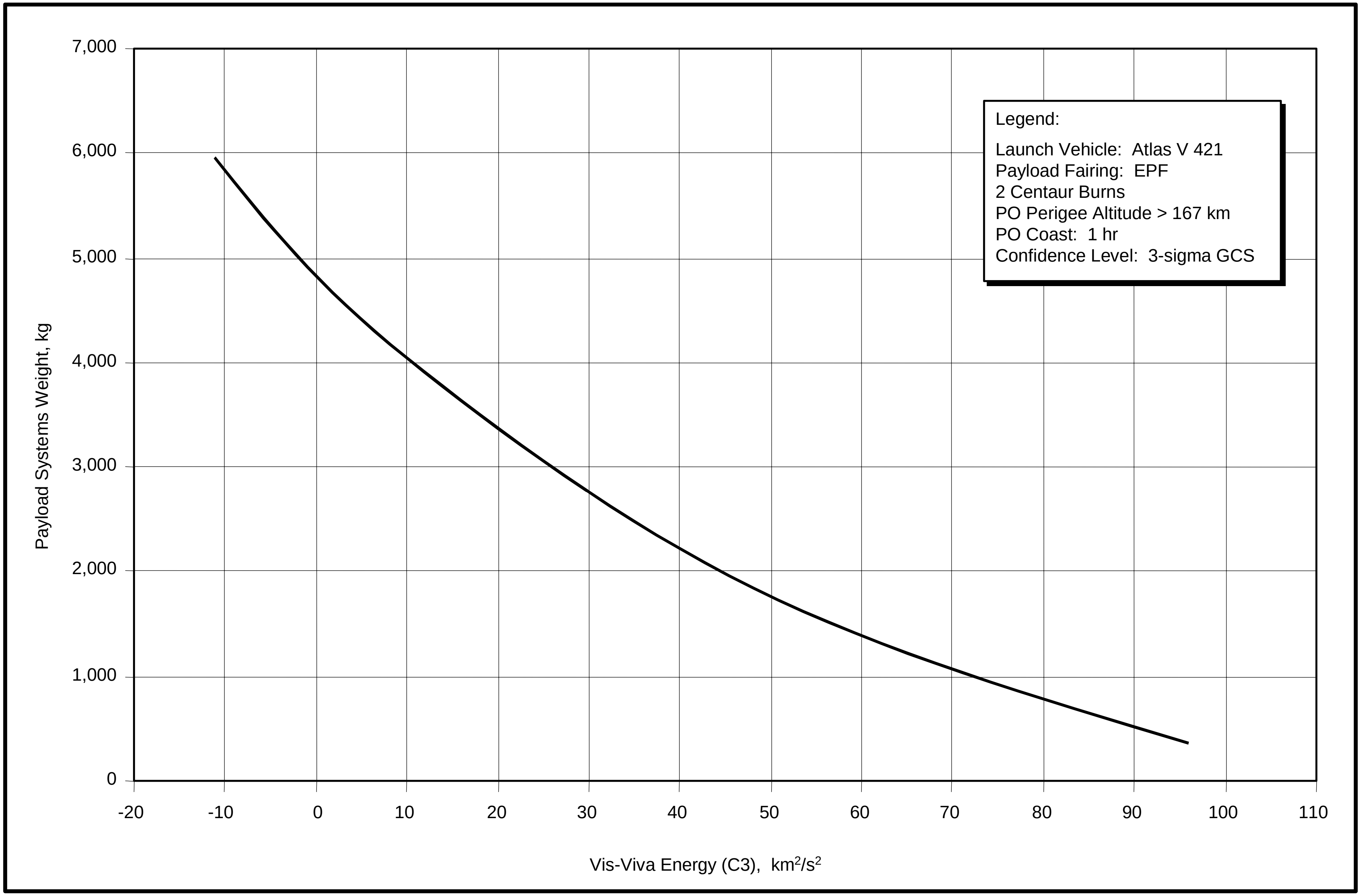


Figure 2.13-6 Atlas V 421 Earth Escape Performance (C3 Curves)

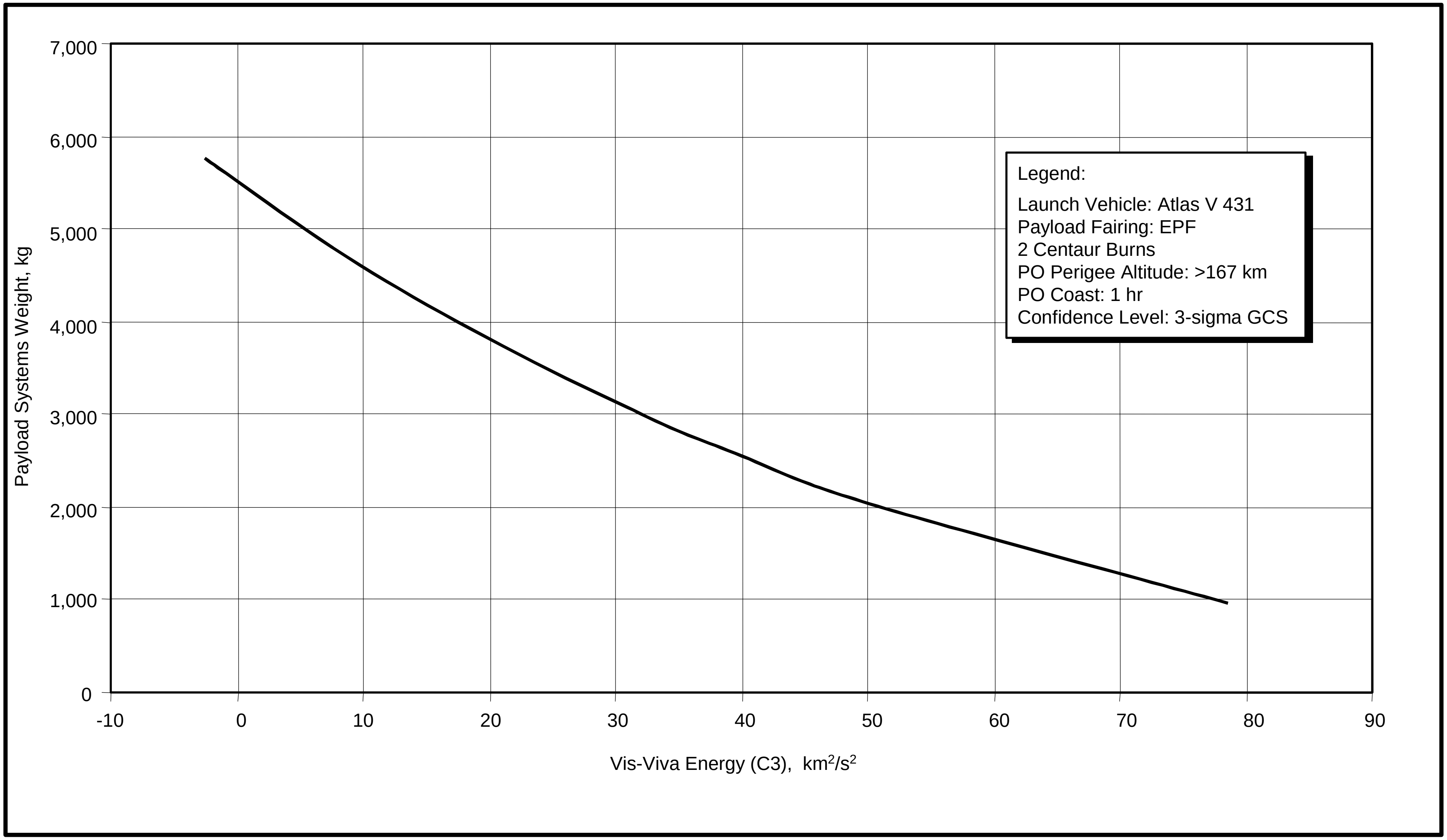


Figure 2.14-6 Atlas V 431 Earth Escape Performance (C3 Curves)

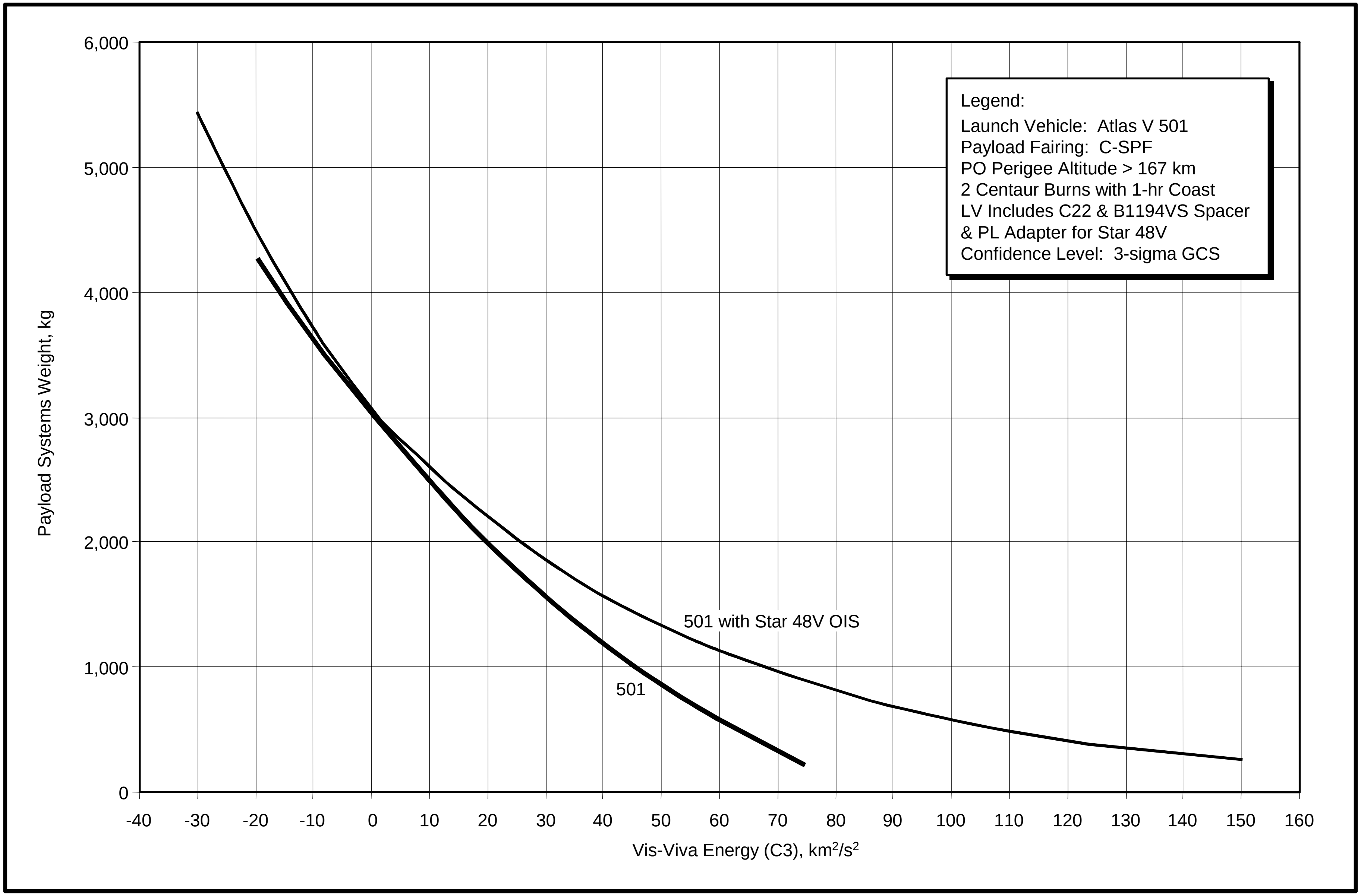


Figure 2.15-6 Atlas V 501 Earth Escape Performance (C3 Curves)

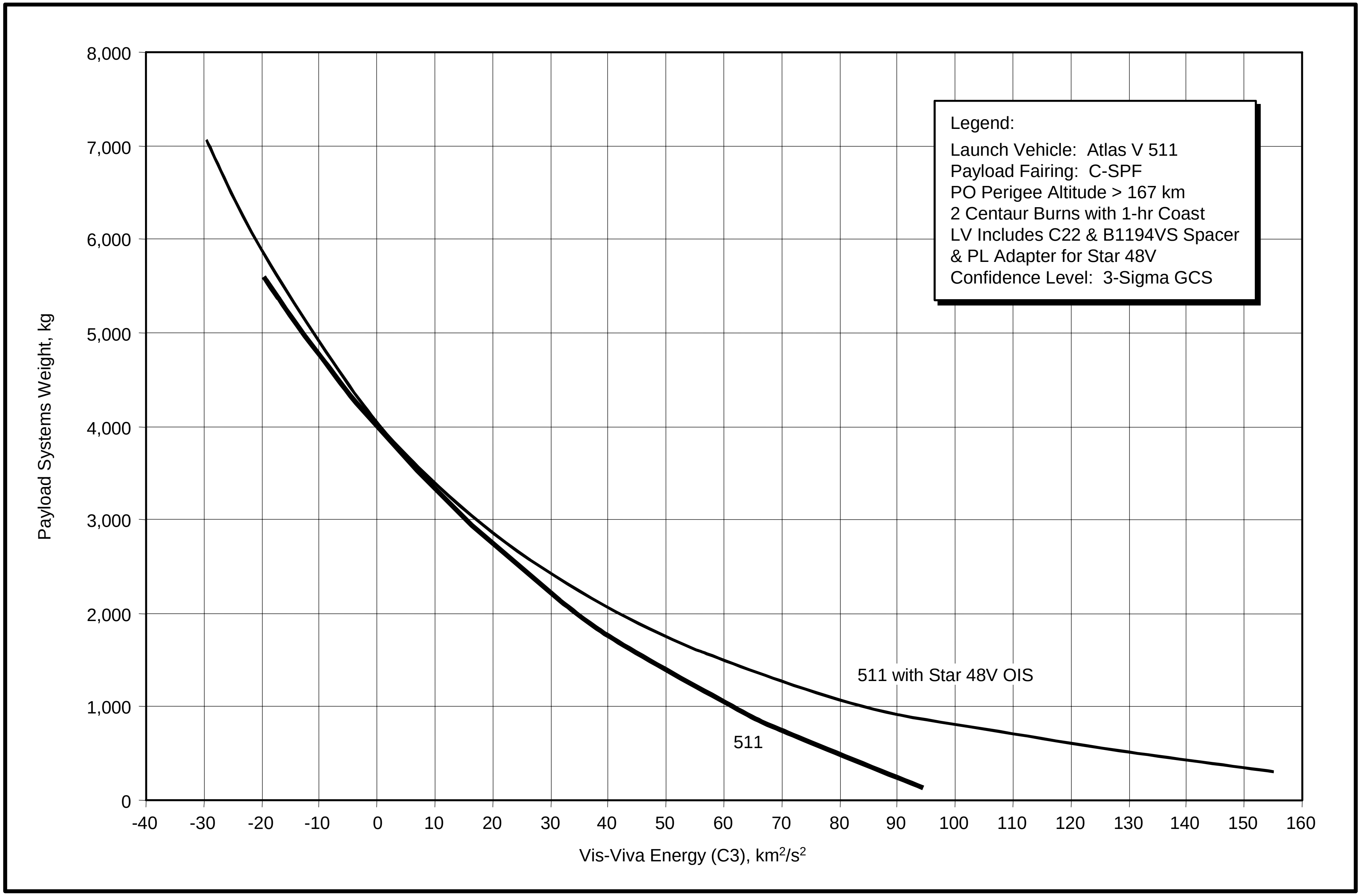


Figure 2.16-6 Atlas V 511 Earth Escape Performance (C3 Curves)

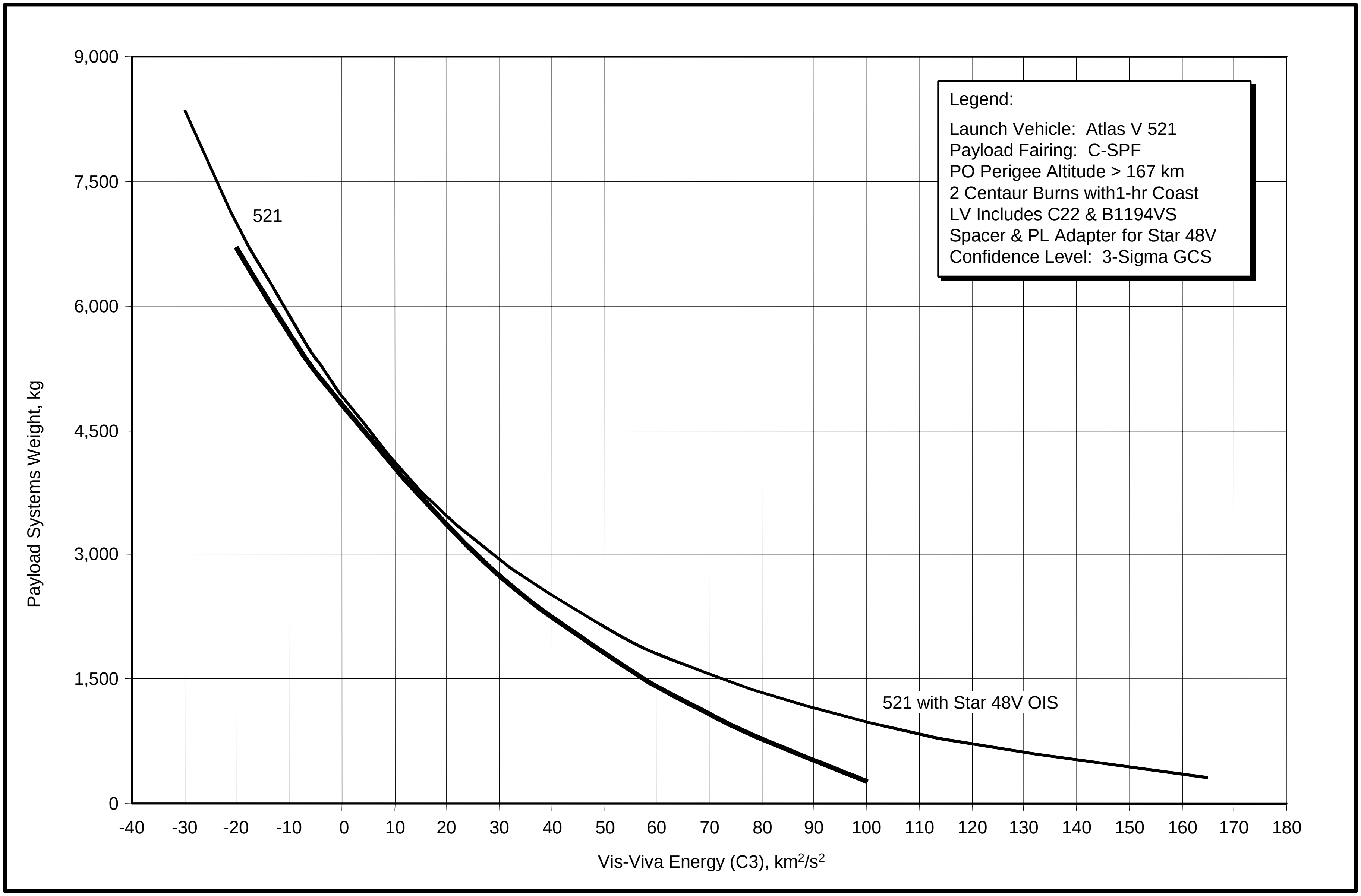


Figure 2.17-6 Atlas V 521 Earth Escape Performance (C3 Curves)

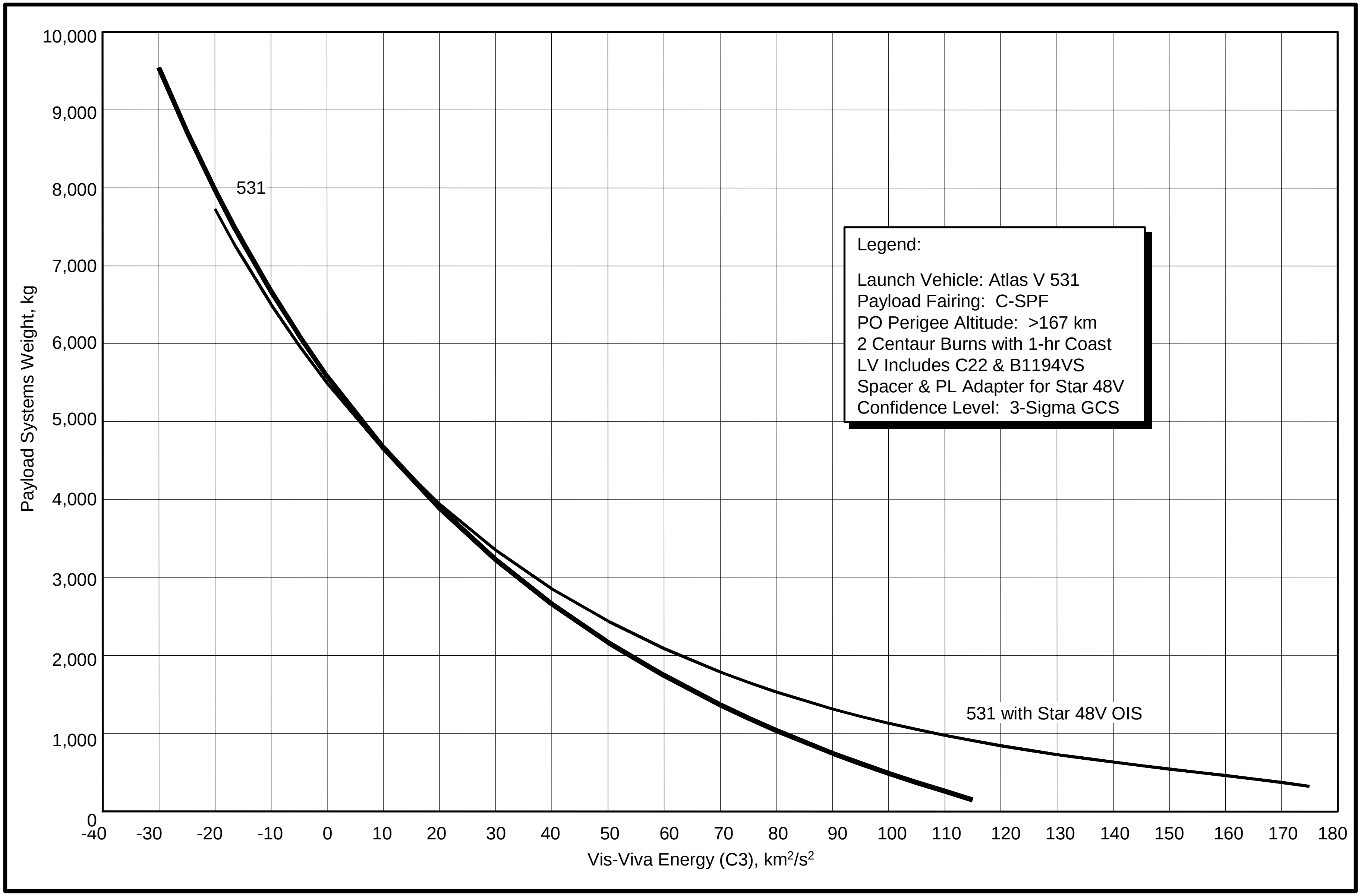


Figure 2.18-6 Atlas V 531 Earth Escape Performance (C3 Curves)

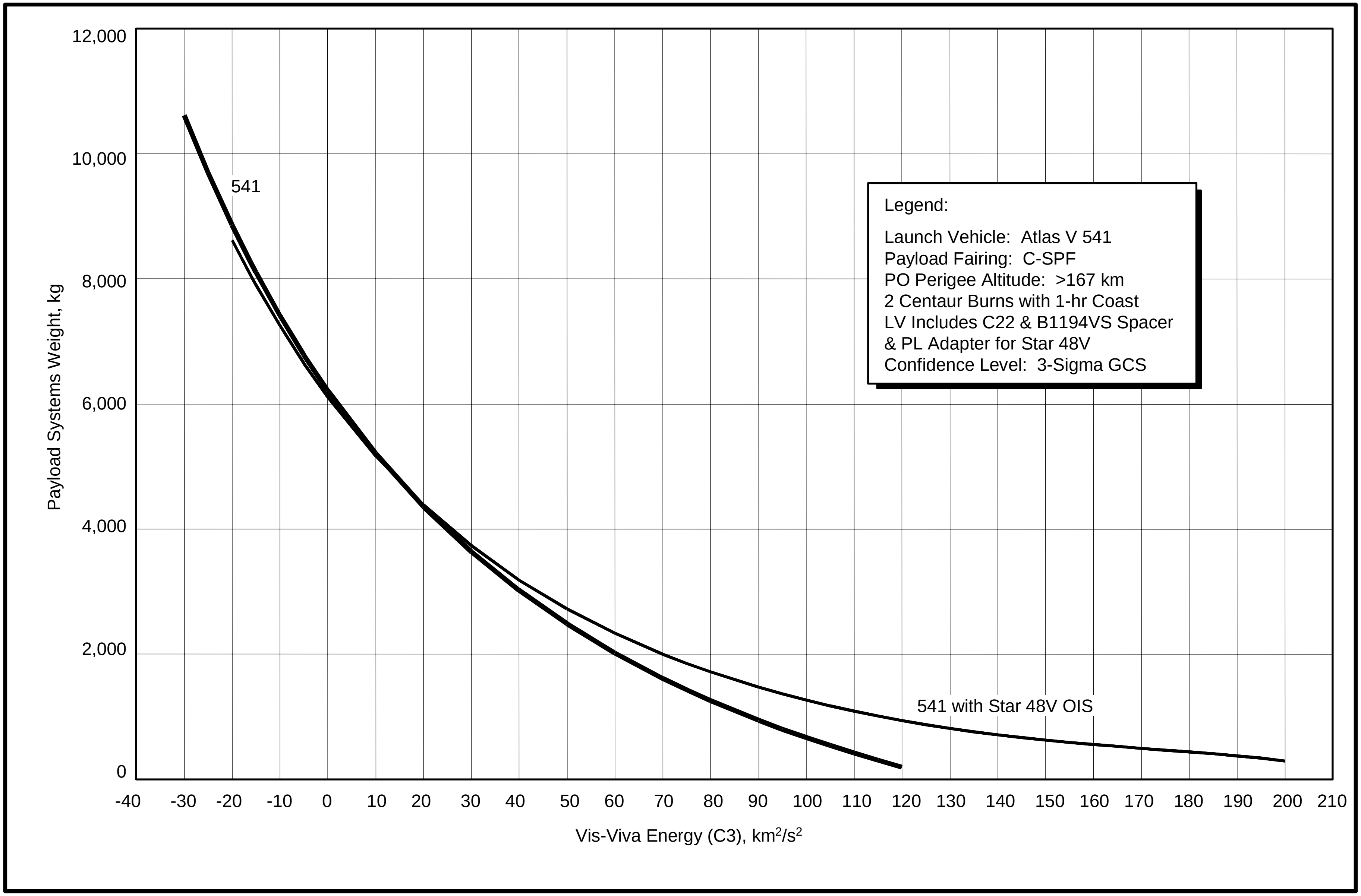


Figure 2.19-6 Atlas V 541 Earth Escape Performance (C3 Curves)

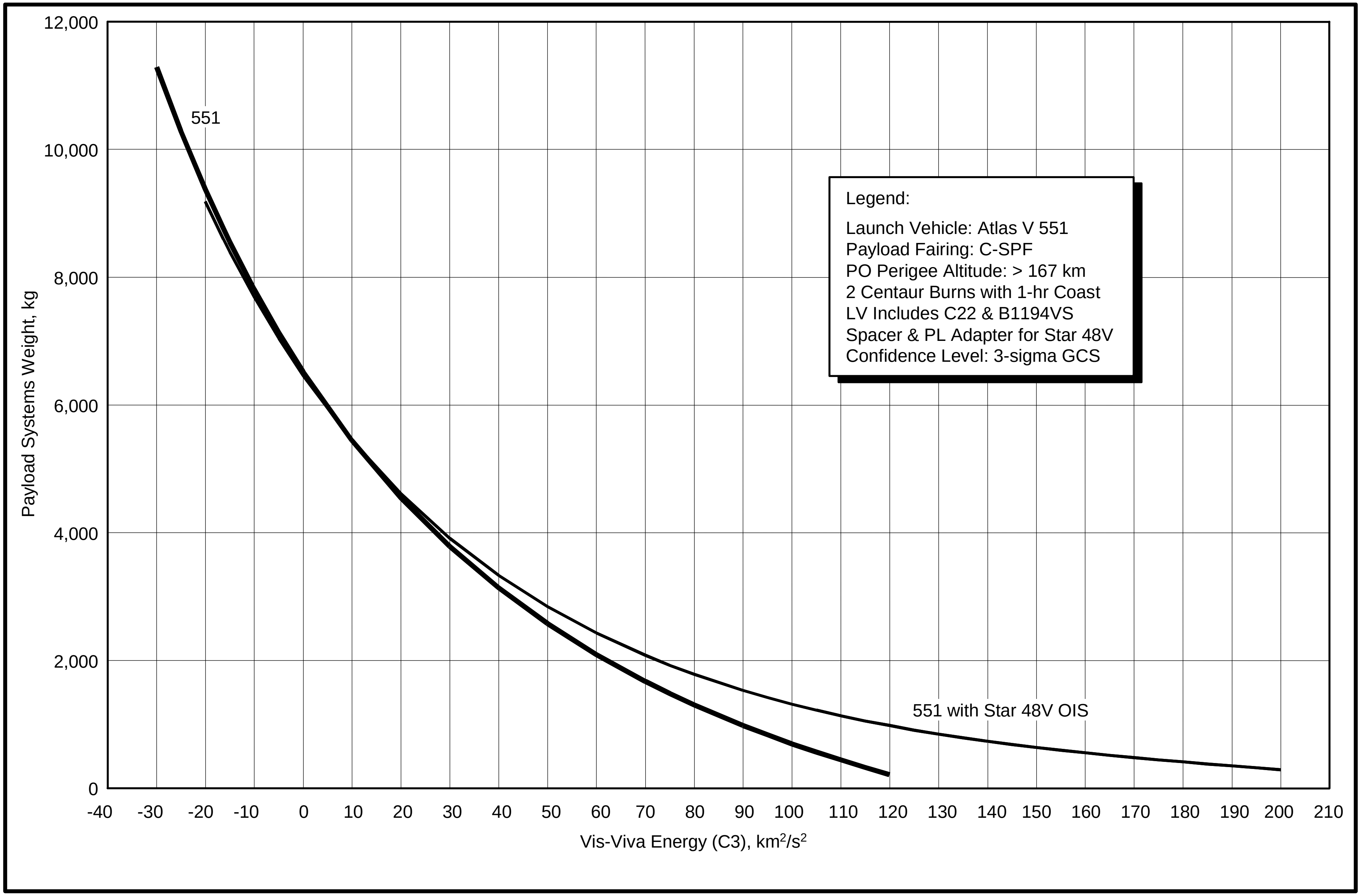


Figure 2.20-6 Atlas V 551 Earth Escape Performance (C3 Curves)

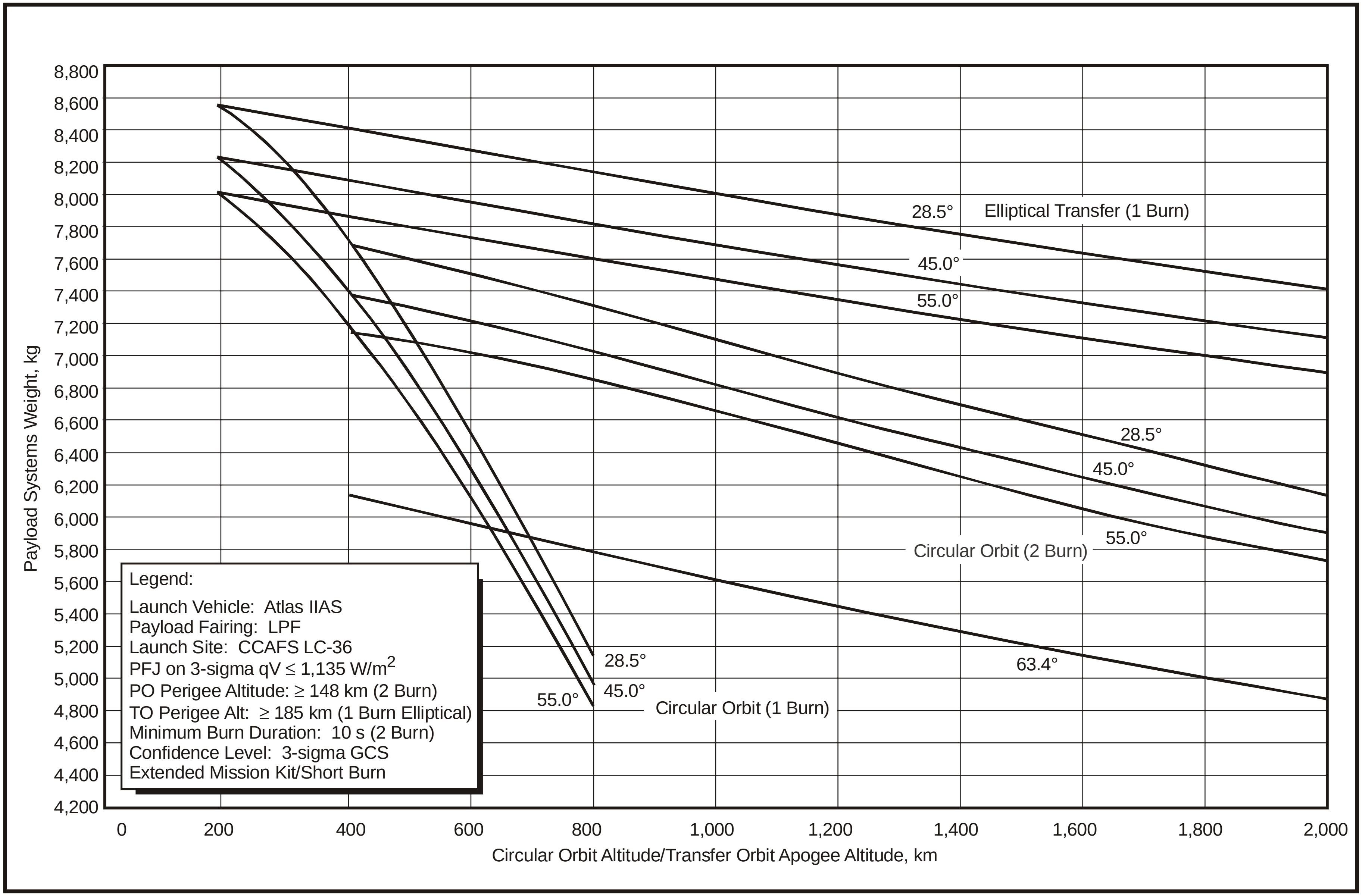


Figure 2.7-7 Atlas IIAS CCAFS Low-Earth Orbit Performance

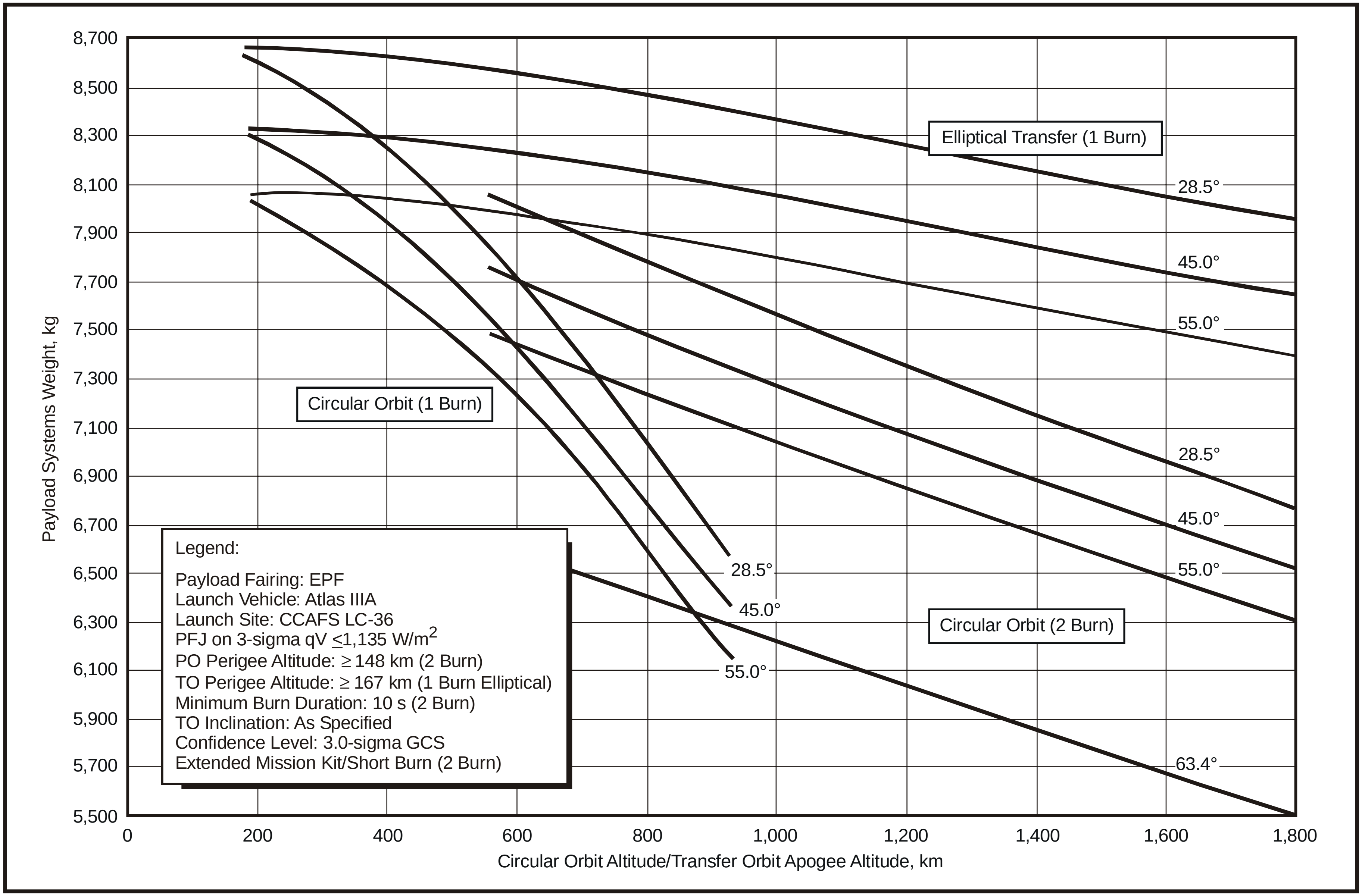


Figure 2.8-7 Atlas IIIA CCAFS Low-Earth Orbit Performance

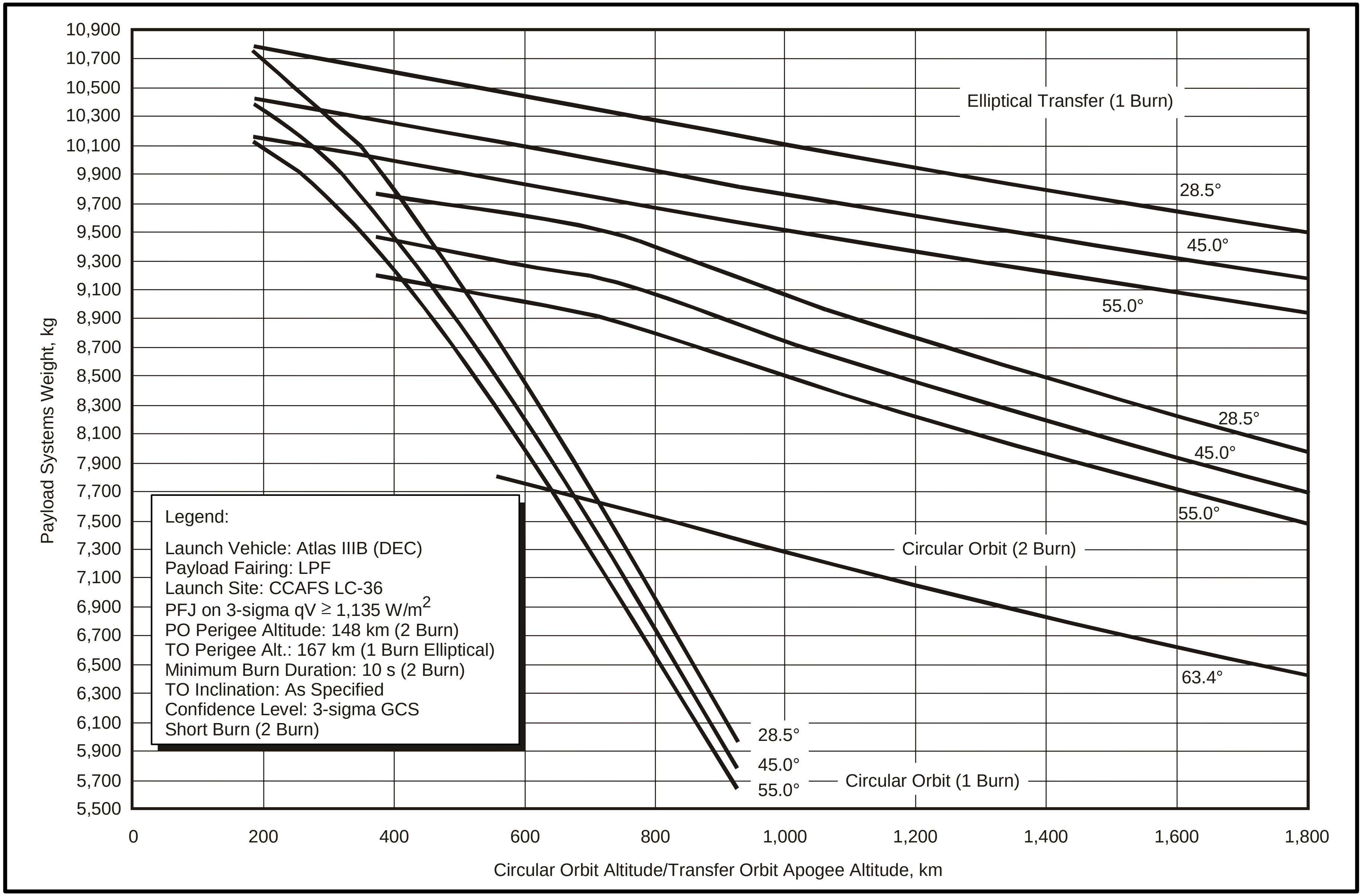


Figure 2.9-7 Atlas IIIB (DEC) CCAFS Low-Earth Orbit Performance

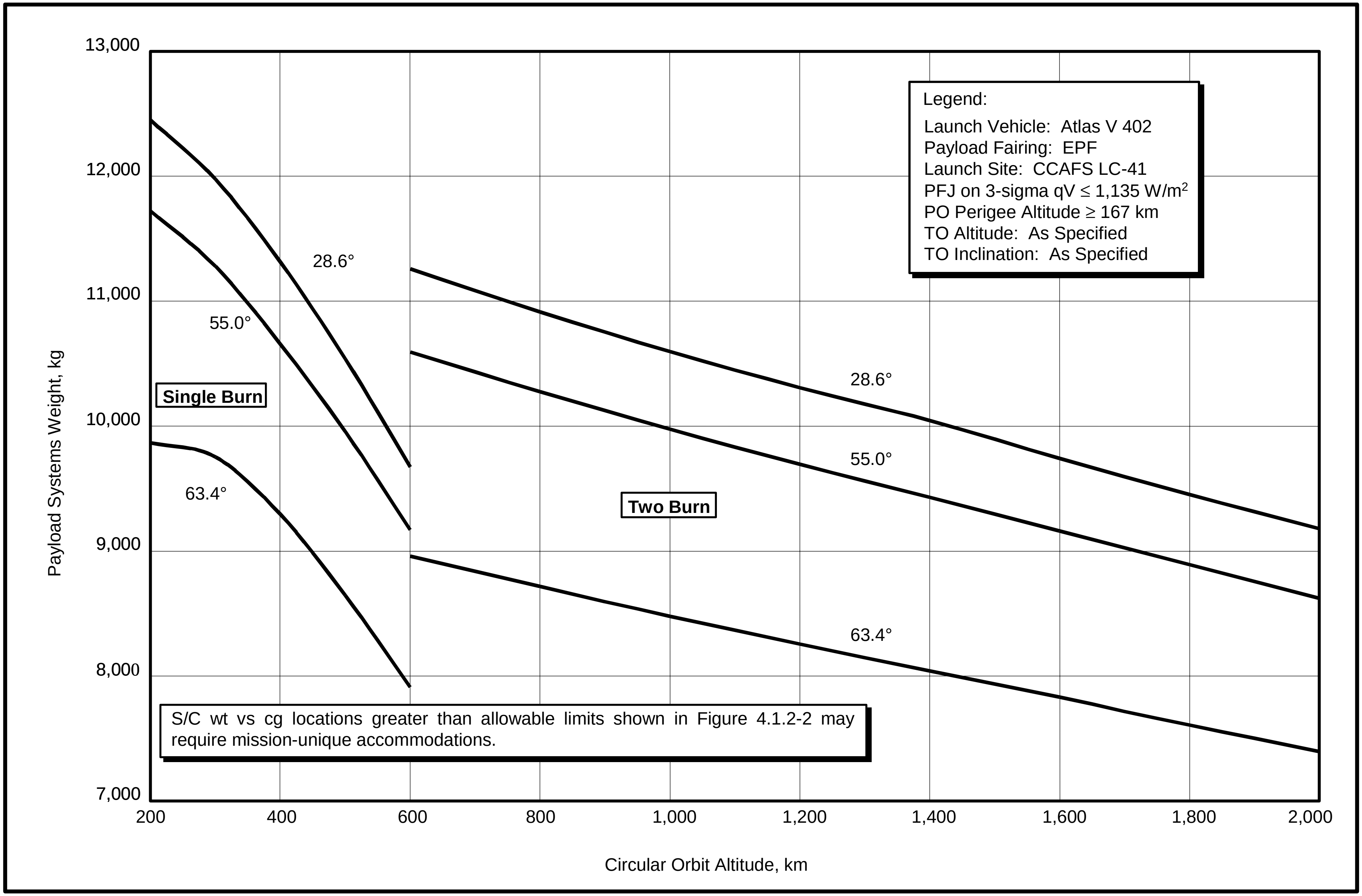


Figure 2.11-7 Atlas V 402 Low-Earth Orbit Performance

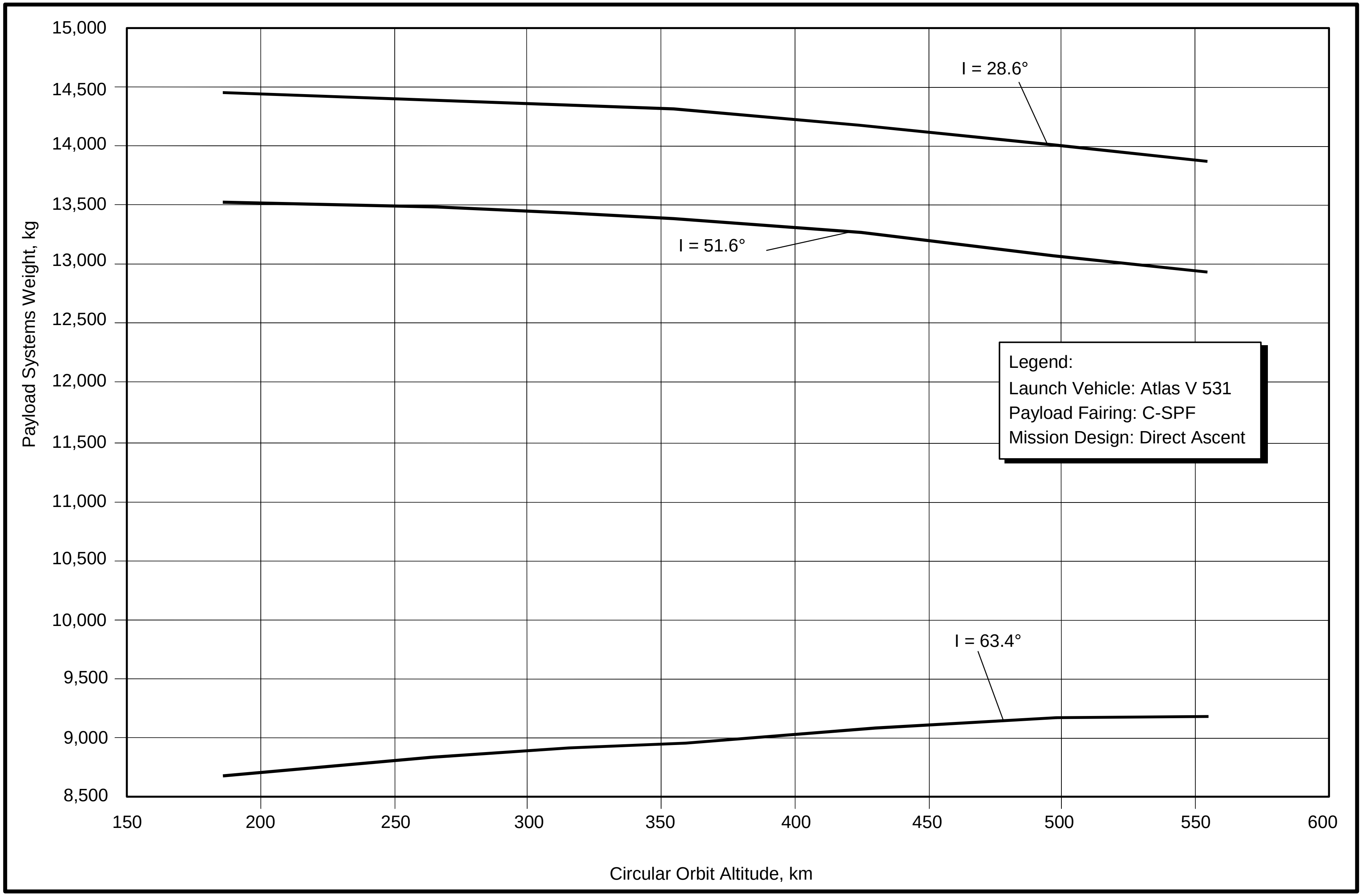


Figure 2.18-7 Atlas V 531 Low-Earth Orbit Performance

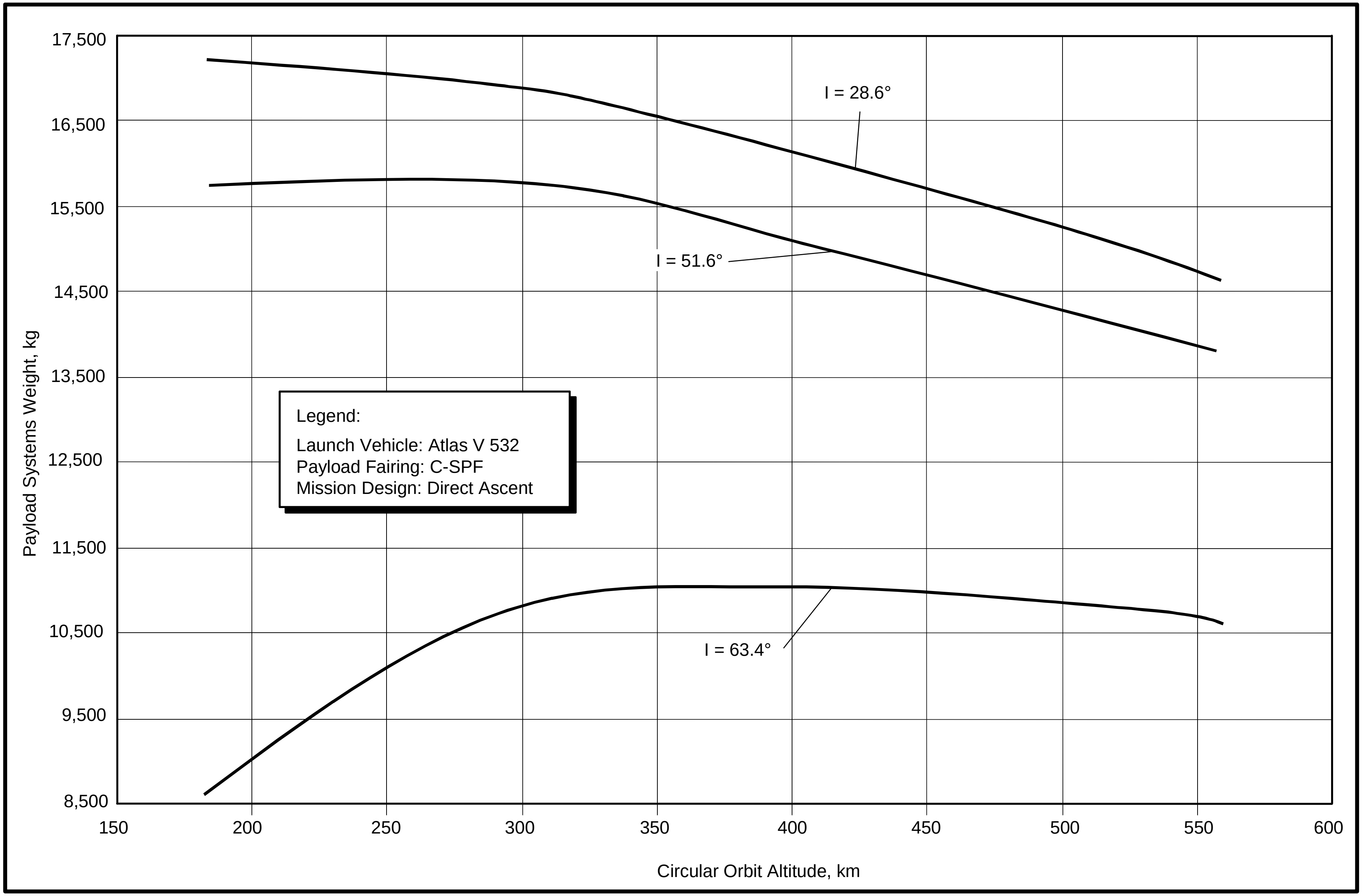


Figure 2.24-7 Atlas V 532 Low-Earth Orbit Performance

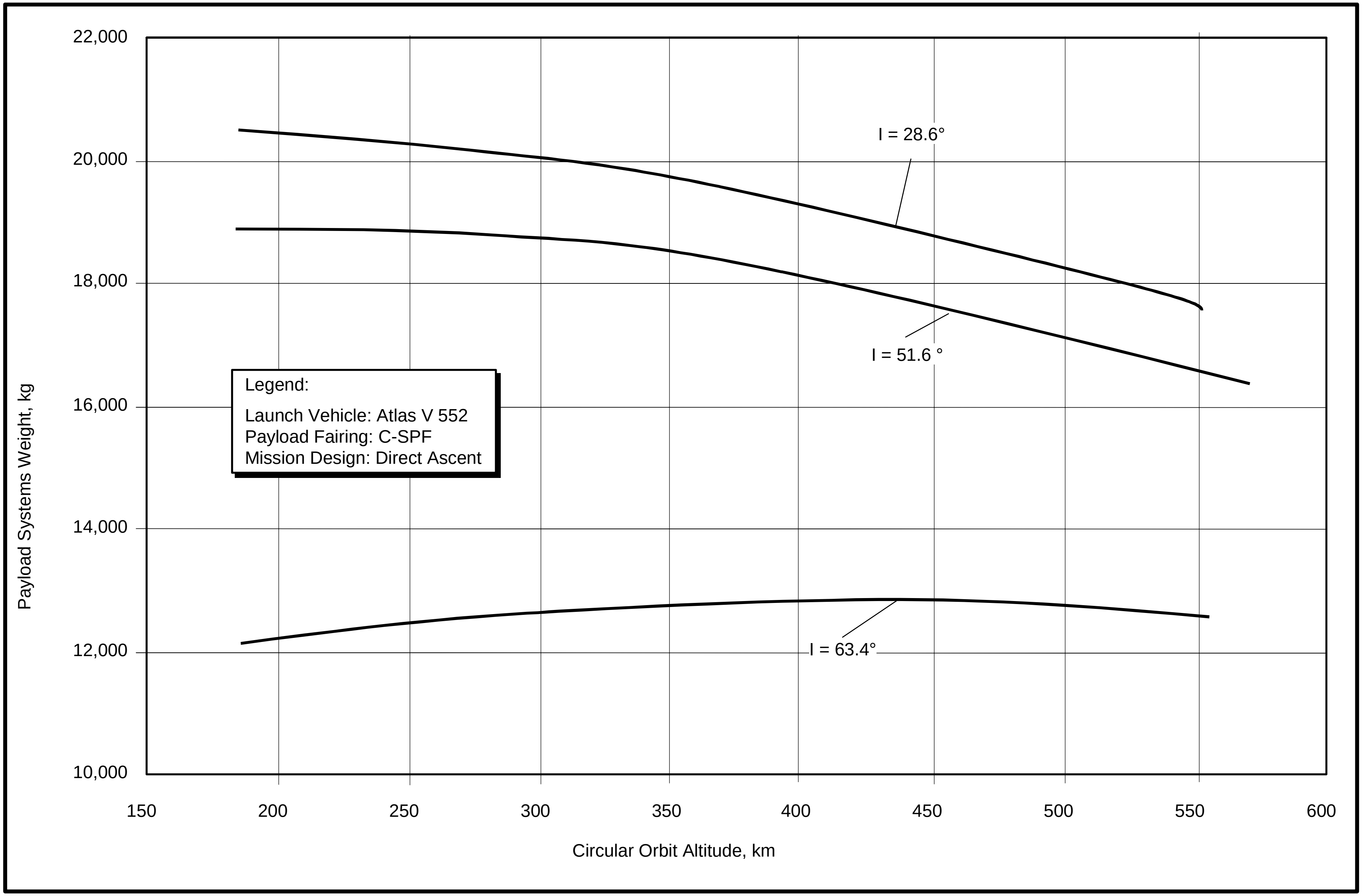


Figure 2.26-7 Atlas V 552 Low-Earth Orbit Performance

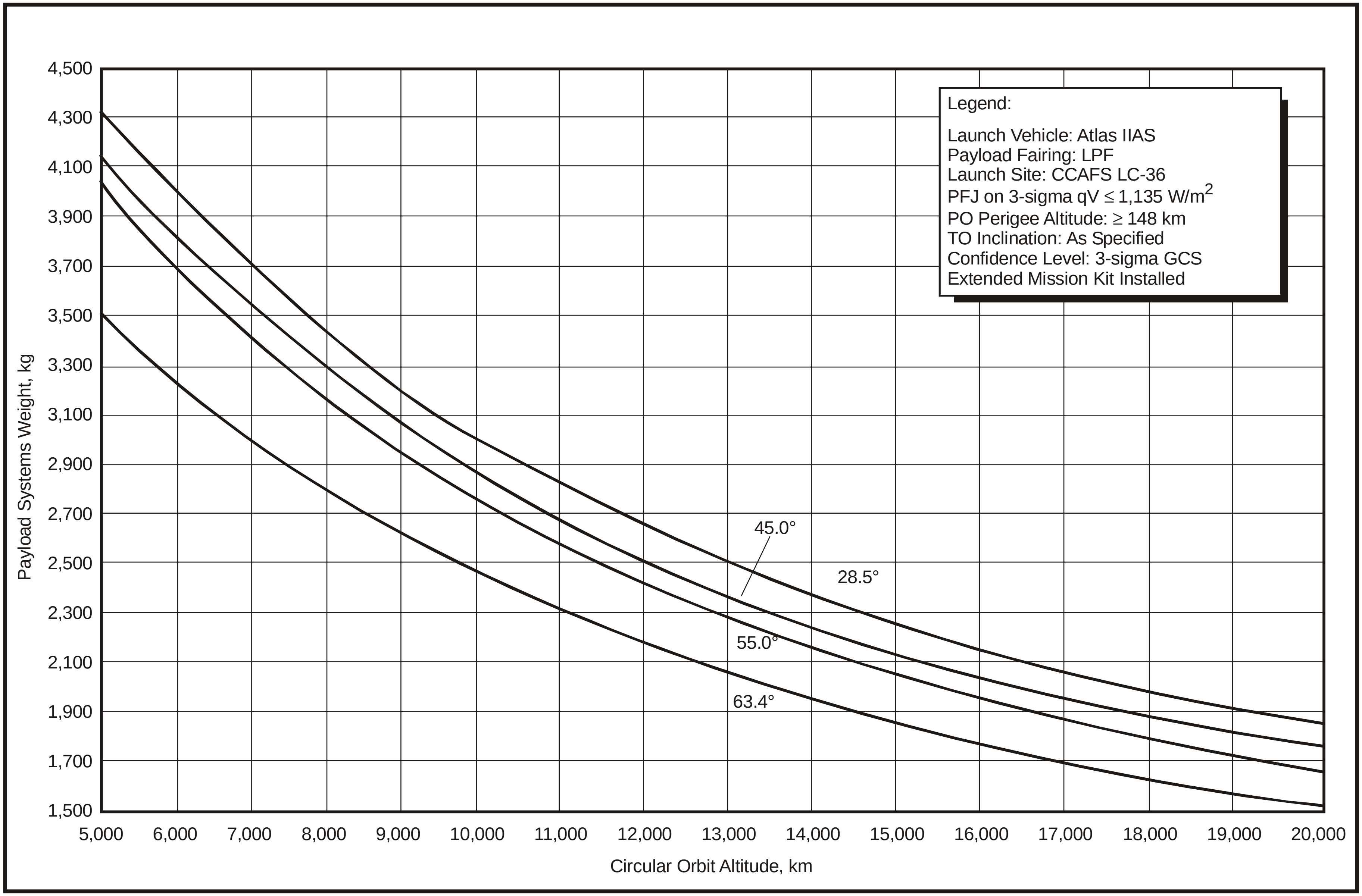


Figure 2.7-8 Atlas IIAS CCAFS Intermediate Circular Orbit Performance

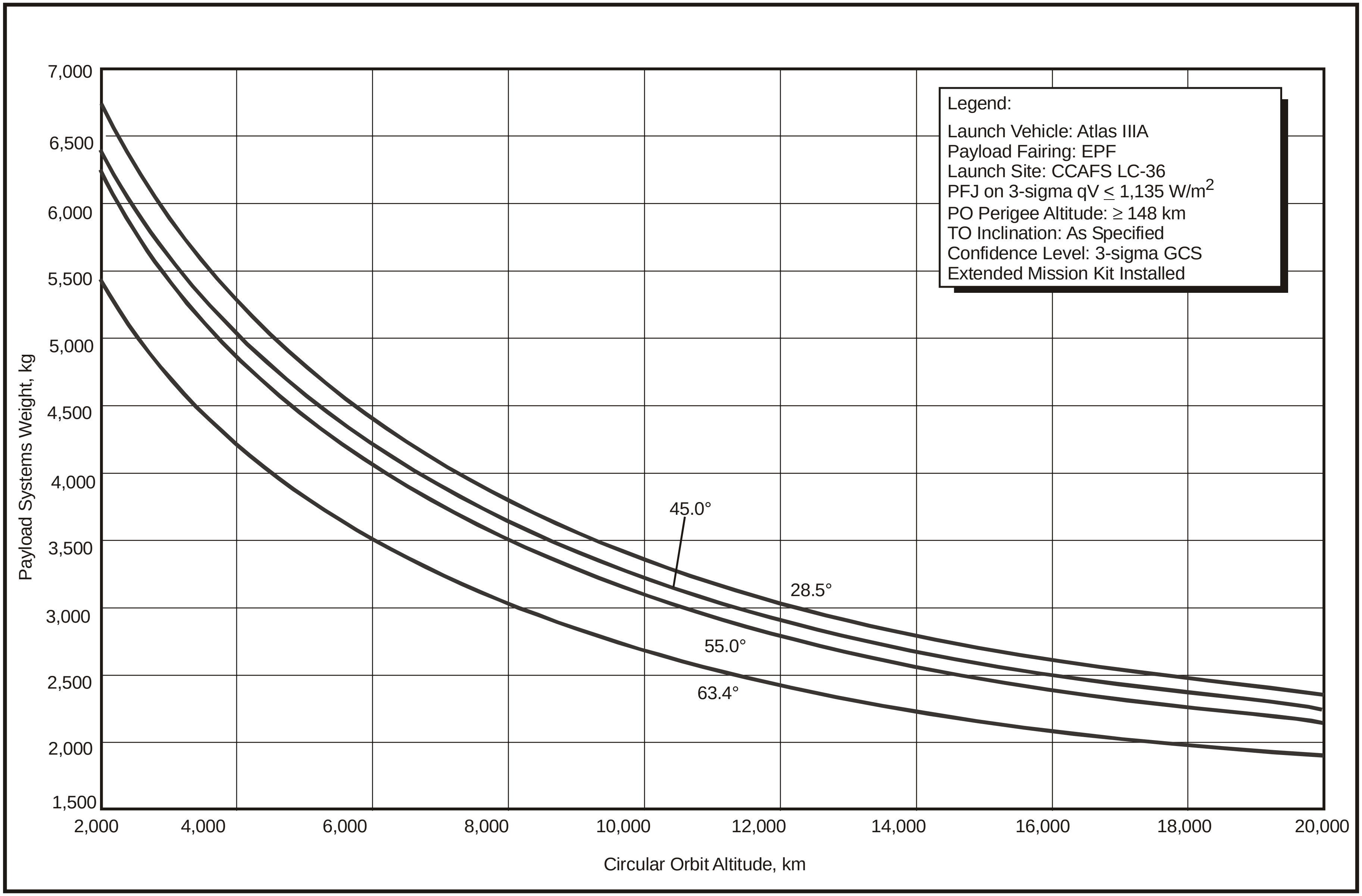


Figure 2.8-8 Atlas IIIA CCAFS Intermediate Circular Orbit Performance

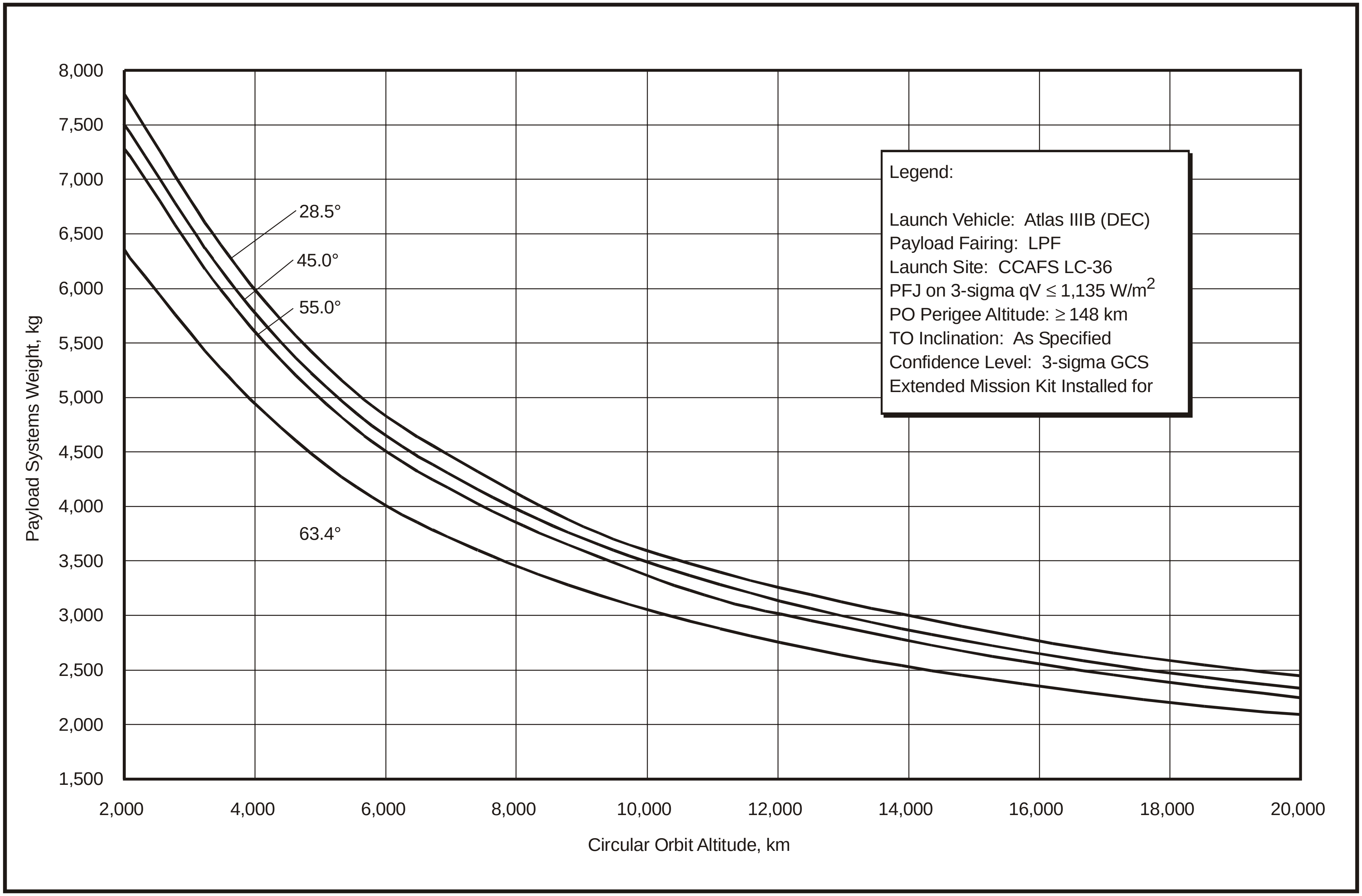


Figure 2.9-8 Atlas IIIB (DEC) CCAFS Intermediate Circular Orbit Performance

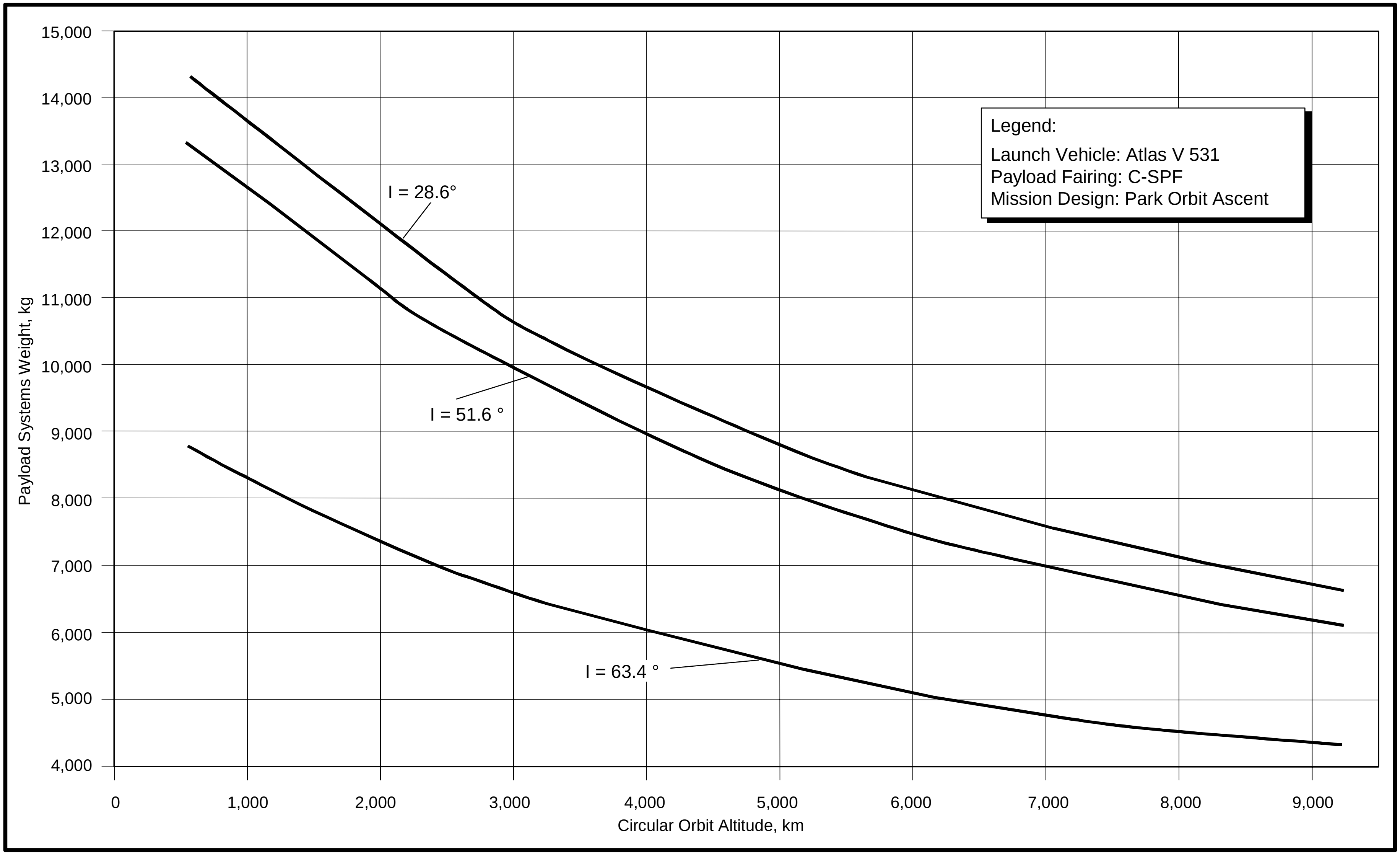


Figure 2.18-8 Atlas V 531 Intermediate Orbit Performance

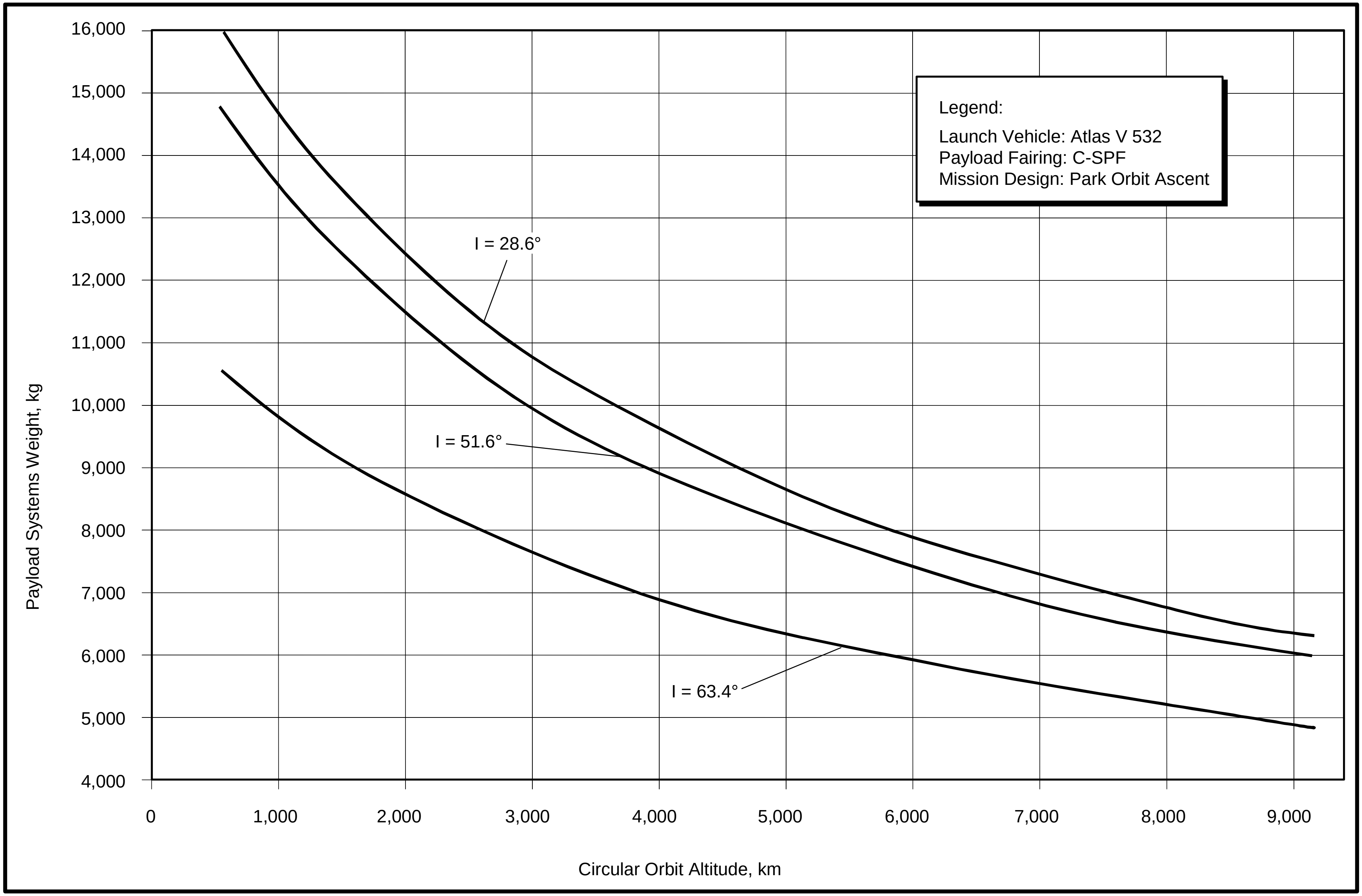


Figure 2.24-8 Atlas V 532 Intermediate Orbit Performance

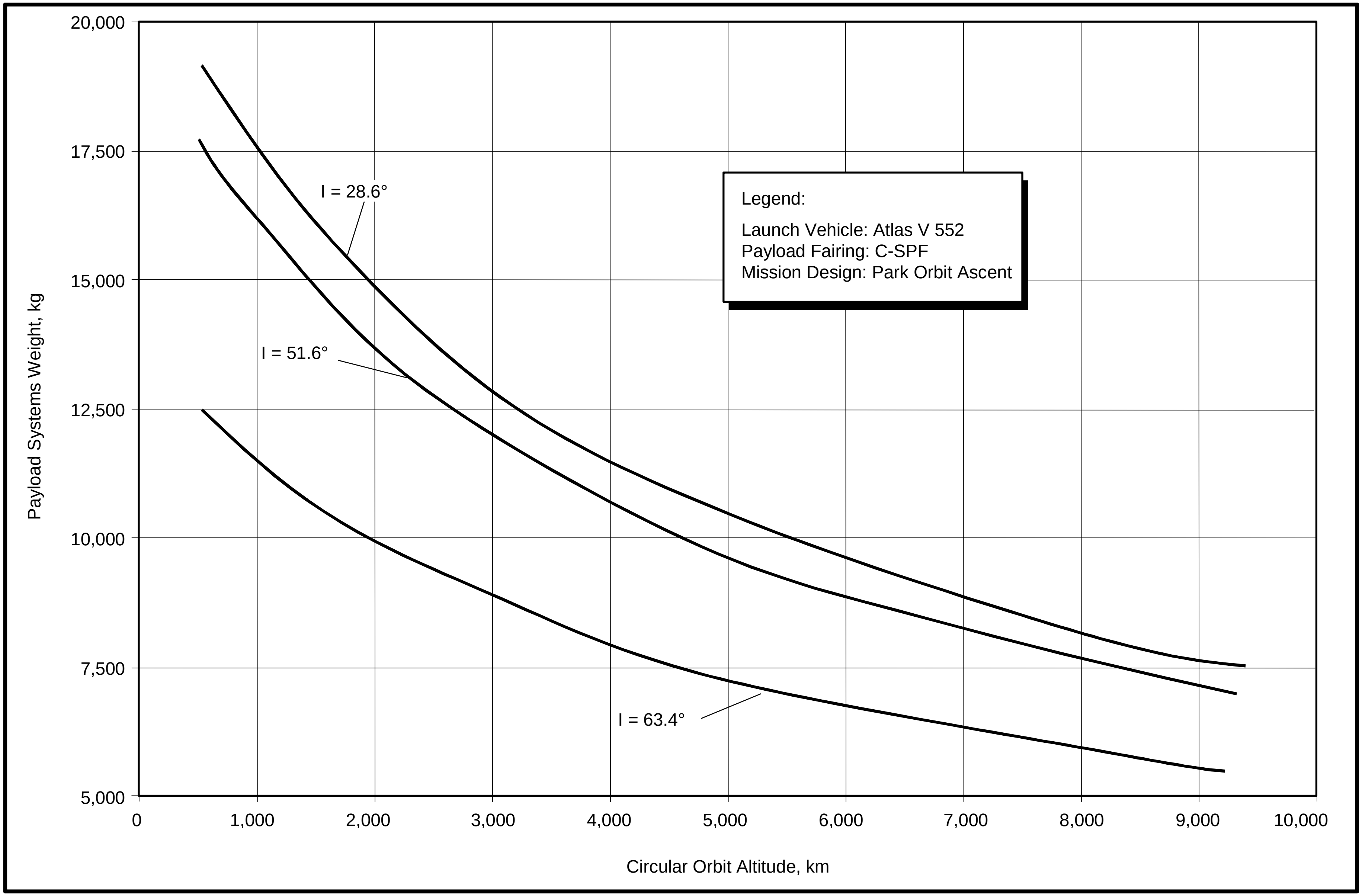


Figure 2.26-8 Atlas V 552 Intermediate Orbit Performance

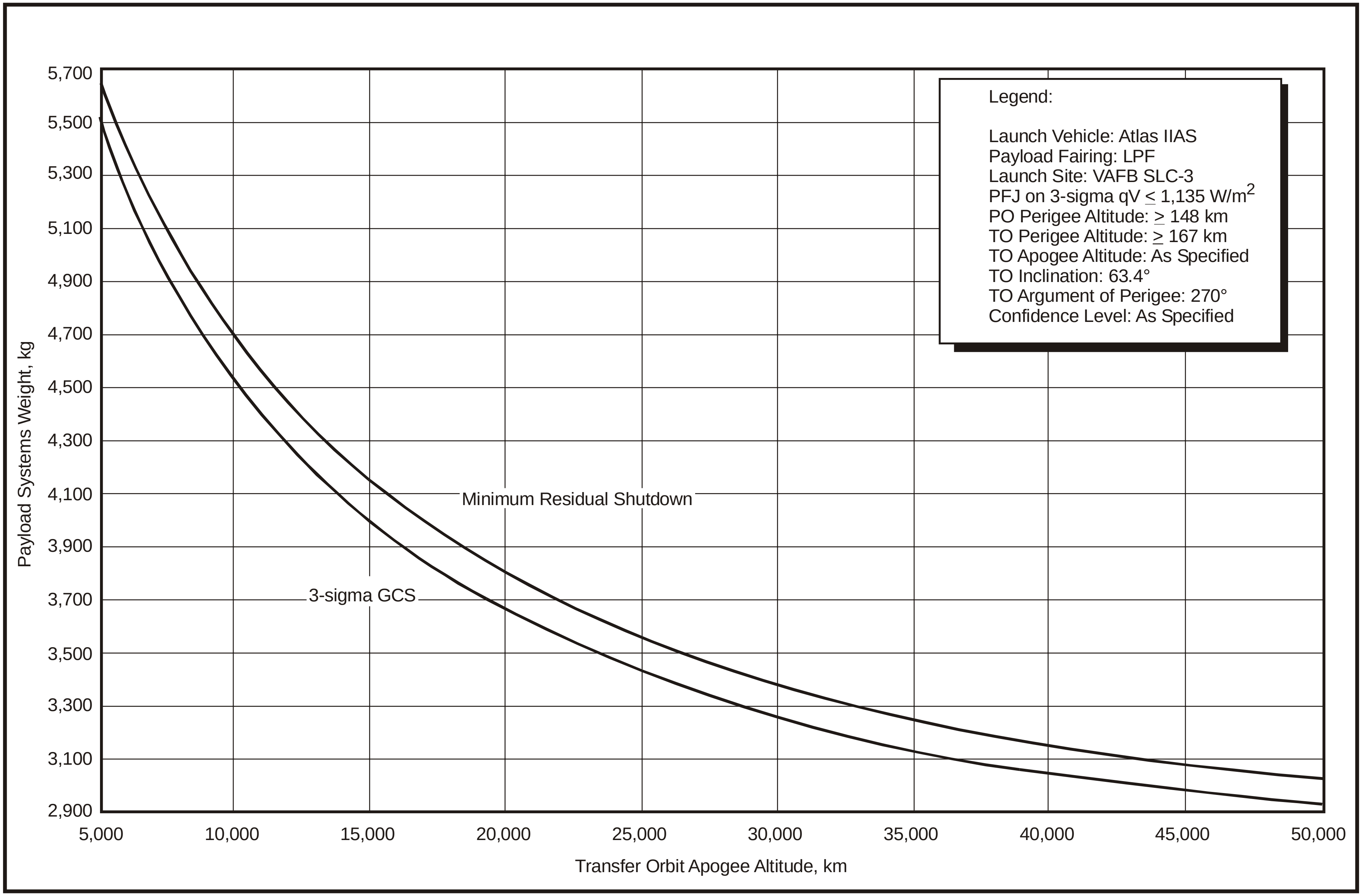


Figure 2.7-9 Atlas IIAS VAFB Elliptical Orbit Performance

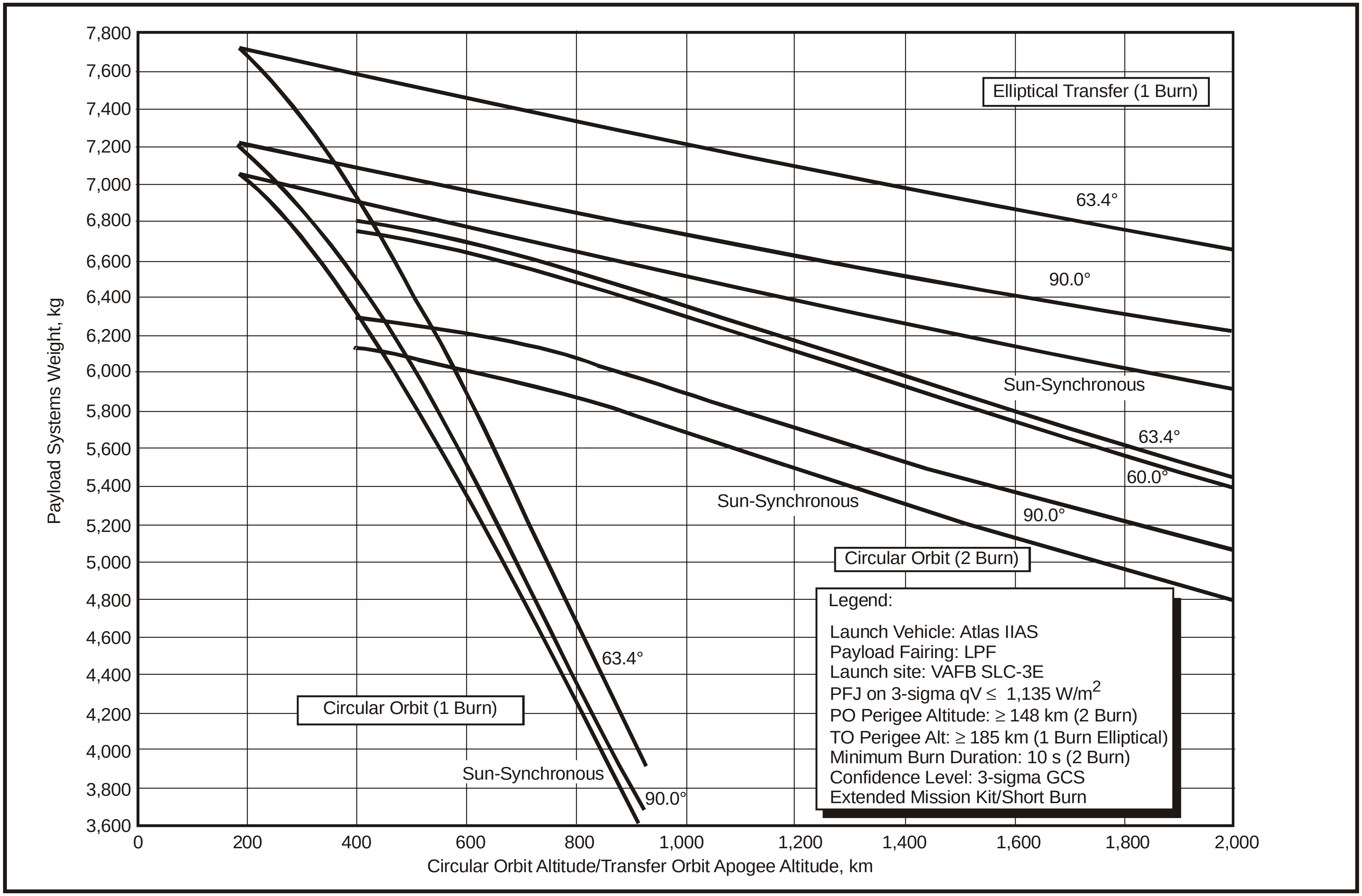


Figure 2.7-10 Atlas IIAS VAFB Performance to Low-Earth Orbit

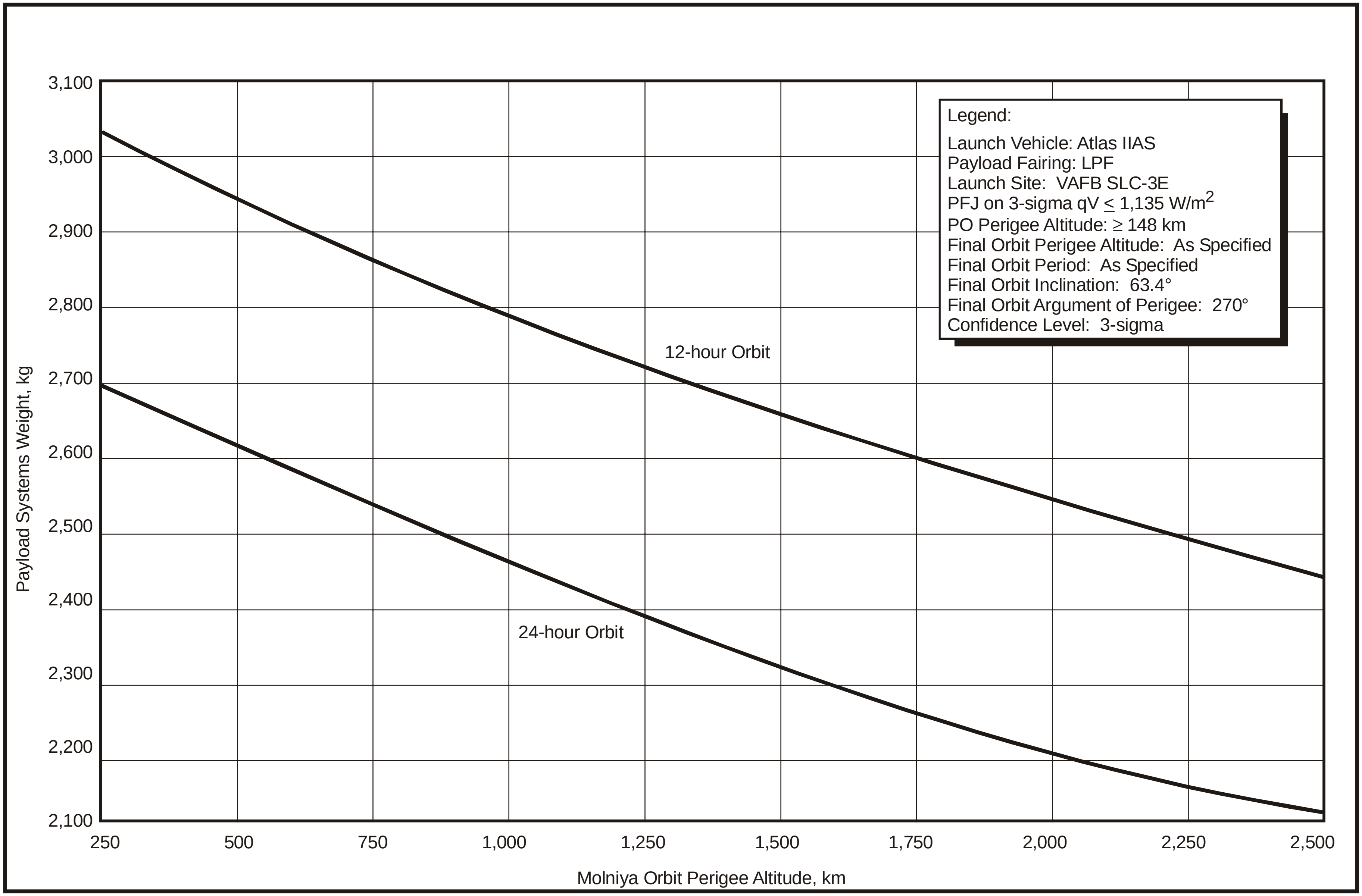


Figure 2.7-11 Atlas IIAS VAFB High-Inclination, High-Eccentricity Orbit Performance

3.0 ENVIRONMENTS

This section describes the environments to which the spacecraft is exposed. Detailed environmental data for Atlas IIAS and Atlas III vehicles are provided for Launch Complex (LC)-36 at Cape Canaveral Air Force Station (CCAFS) and for Space Launch Complex (SLC)-3 facility at Vandenberg Air Force Base (VAFB). Detailed spacecraft environmental data for Atlas V vehicles are provided for Launch Complex 41 (LC-41) at CCAFS.

Prelaunch environments are described in Section 3.1; flight environments are described in Section 3.2; and spacecraft test requirements are described in Section 3.3.

3.1 PRELAUNCH ENVIRONMENTS

3.1.1 Thermal

The spacecraft thermal environment is controlled during prelaunch activity, maintained during ground transport and hoist, and controlled after mate to the launch vehicle as follows:

Payload Processing Facility (PPF)—Environments in the spacecraft processing areas at Astrotech/CCAFS are controlled at 21-27°C (70-80°F) and 50 ±5% relative humidity. Portable air conditioning units are available to further cool test equipment or spacecraft components as required.

Ground Transport From PPF to Launch Site—During ground transport from the PPF to the LC-36 launch site, the temperature within the Atlas II/III 4-m payload fairing (PLF) remains between 4-30°C (40-86°F), with conditioning provided by an air environmental control system (ECS) or a gaseous nitrogen (GN_2) purge. If required, the air ECS can maintain air temperatures between 10-25°C (50-77°F). The maximum dewpoint temperature is -37°C (-35°F) for GN_2 and 4.4°C (40°F) for air. At VAFB SLC-3, the transporter environmental control unit maintains temperature between 10-27°C (50-80°F) at a maximum dewpoint temperature of 4.4°C (40°F). A GN_2 backup system is available at VAFB.

The air temperature within the Atlas V 5-m payload fairings is maintained between 4-30°C (40-86°F) during transport from the PPF to the vertical integration facility (VIF). This is accomplished with a mobile environmental control unit that guarantees a maximum dewpoint temperature of 4.4°C (40°F).

Hoisting Operations—During hoisting operations at LC-36A, SLC-3E, and LC-41, the encapsulated spacecraft is purged with dry GN_2 with a maximum dewpoint of -37°C (-35°F). A high-flow-rate conditioned air system with a maximum dewpoint of 4.4°C (40°F) is used during hoisting operations at LC-36B.

Post-Spacecraft Mate To Centaur—After spacecraft mate to Centaur, gas conditioning is provided to the PLF at the required temperature, humidity, and flow rate. Air with a maximum dewpoint of 4.4°C (40°F) is used until approximately 2 hours before launch for LC-36 and SLC-3E and approximately 4 hours before launch at LC-41, after which GN_2 with a maximum dewpoint of -37°C (-35°F) is used. Table 3.1.1-1 summarizes prelaunch gas conditioning temperature capabilities for the Atlas IIAS

Table 3.1.1-1 Atlas IIAS and Atlas III Gas Conditioning Capabilities

			Temperature Range Inside Payload Fairing**—LPF & EPF		
	Inlet Temp Capability*	**Flow Rate Capability**	**Atlas IIAS Baseline (Bare)**	**Atlas IIAS & Atlas III With Acoustic Blankets**	**Atlas IIAS & Atlas III with Thermal Shield**
Inside Tower	10-29°C (50-85°F)	50-72.7 kg/min (110-160 lb/min)	9-25°C (48-77°F)	9-19°C (48-67°F)	9-17°C (48-63°F)
After Tower Removal	10-29°C (50-85°F)	50-72.7 kg/min (110-160 lb/min)	7-29°C (45-85°F)	9-21°C (48-70°F)	9-18°C (48-65°F)

Note: * Inlet Temperature Is Adjustable (within System Capability) According to Spacecraft Requirements
** Temperature Ranges Are Based on Worst-Case Minimum & Maximum External Heating Environments
Above Ranges Assume a 72.7 kg/min (160-lb/min) PLF ECS Flow Rate

and Atlas III PLF configurations at LC-36 and SLC-3E. The optional thermal shield for the large payload fairing (LPF) and extended payload fairing (EPF) allows greater control over PLF internal temperatures during prelaunch gas conditioning. The shield consists of a noncontaminating membrane attached to the inboard surfaces of the PLF frames (Fig. 3.1.1-1).

Table 3.1.1-2 summarizes prelaunch gas conditioning temperature capabilities for the nominal Atlas V 400 series and 500 series configurations at LC-41. Mission-peculiar arrangements for dedicated purges of specific components can be provided.

The ECS flow to the payload compartment is supplied through a ground/airborne disconnect on the PLF and is controlled by prime and backup environmental control units. These units provide conditioned air or GN_2 to the specifications in Table 3.1.1-3. Figure 3.1.1-2 gives the general PLF gas-conditioning layout for the 4-m configuration. Figure 3.1.1-3 shows the general PLF gas-conditioning layout for 5-m configuration.

For Atlas II/III, internal ducting in the PLF directs the gas upward to prevent direct impingement on the spacecraft. The conditioning gas is vented to the atmosphere through one-way flapper doors in the aft end of the PLF. This baseline design ensures that average gas velocities across spacecraft components for the LPF and EPF are less than 6.1 m/s (20

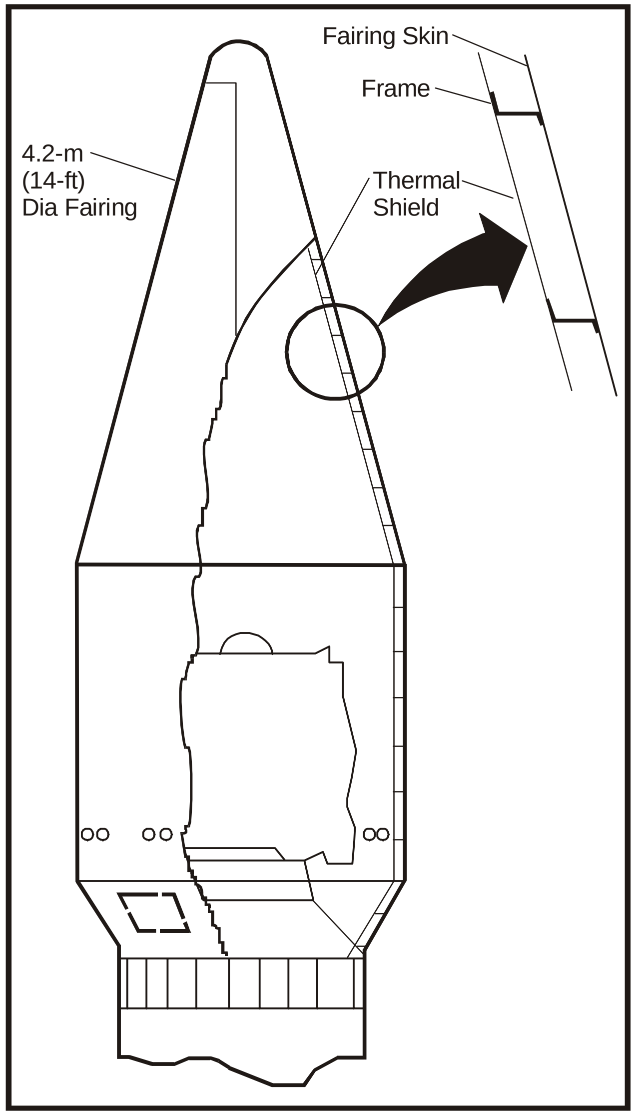


Figure 3.1.1-1 Thermal Shield Option

Table 3.1.1-2 Atlas V Gas Conditioning Capabilities

			Temperature Range Inside Payload Fairing**		
			Atlas V 400 Series	Atlas V 500 Series	
Location	Inlet Temperature Capability*	Inlet Flow Rate Capability, kg/min (lb/min)	EPF & LPF, °C(°F)	5-m Short, °C(°F)	5-m Medium, °C(°F)
Post-LV Mate Through Move to Launch Configuration	10-29°C (50-85°F)	Atlas V 400 Series 22.7-72.6 (50-160) Atlas V 500 Series 22.7-136.2 (50-300)	6-20°C (43-68°F)	6-20°C (43-68°F)	6-20°C (43-68°F)
Post-Move to Launch Configuration	10-29°C (50-85°F)	Atlas V 400 Series 22.7-72.6 (50-160) Atlas V 500 Series 22.7-136.2 (50-300)	6-21°C (43-70°F)	6-21°C (43-70°F)	6-21°C (43-70°F)

Notes: * Inlet Temperature Is Adjustable (Within System Capability) According to Spacecraft Requirements
** Temperature Ranges Are Based on Worst-Case Minimum & Maximum External Heating Environments
Ranges Shown Assume a 72.6 kg/min (160 lb/min) for EPF/LPF ECS Flow Rate & 136.2 kg/min (300 lb/min) for 5-m Short/Medium ECS Flow Rate

Table 3.1.1-3 Conditioned Air/GN_2 Characteristics

Parameter	Description	
	Atlas IIAS/III	Atlas V (400/500 Series)
Cleanliness	• Class 5,000 per FED-STD-209B at LC-36 • Class 5,000 per FED-STD-209D at VAFB	• Class 5,000 per FED-STD-209D
Inlet Temperature	• Setpoint from 10-29°C (50-85°F)	• Setpoint from 10-29°C (50-85°F) • 10-21°C (50-70°F) for Sensitive Operations
Inlet Temperature Control	• ±2°C (±3°F) at SLC-36 • ±1.1°C (±2°F) at VAFB	• ±3°C (±5°F)
Filtration	• 99.97% HEPA Not Dioctyl Phthalate (DOP) Tested	• 99.97% HEPA Not Dioctyl Phthalate (DOP) Tested
Flow Rate	• 50-72.7 kg/min (110-160 lb/min) at SLC 36 • 22.7-72.7 kg/min (50-160 lb/min) at VAFB	• **Atlas V 400 Series**: 22.7-72.6 kg/min ± 2.3 kg/min (50–160 lb/min ±5 lb/min) • **Atlas V 500 Series**: 22.7-136.2 kg/min ± 5.7 kg/min (50–300 lb/min ±12.5 lb/min)
Dewpoint (Maximum)	• 4.4°C (40°F) Air • -37.2°C (-35°F) GN_2	• 20-50% RH Air • 35-50% RH (Sensitive Operation) Air • -37.2°C (-35°F) GN_2

ft/s) with medium fill-factor spacecraft that extend into the lower conical section of the payload fairing. For large fill-factor spacecraft, gas impingement velocities within the LPF and EPF could conservatively approach 9.75 m/s (32 ft/s).

For Atlas V, the PLF air distribution system will provide a maximum airflow velocity in all directions of no more than 9.75 mps (32 fps) for the Atlas V 400 series and 10.67 mps (35 fps) for the Atlas V 500 series. There will be localized areas of higher flow velocity at, near, or associated with the air conditioning outlet. Maximum airflow velocities correspond to maximum inlet mass flow rates. Reduced flow velocities are achievable using lower inlet mass flow rates. If required, a computational fluid dynamics analysis can be performed to verify mission-unique gas impingement velocity limits.

The conditioned air is typically delivered near the top of the PLF. For Atlas V, the flow can be divided so up to 40% of the gas flow is directed to the base of the payload compartment when required for spacecraft battery cooling or other operations.

Mission-specific arrangements for dedicated grade B or C GN_2 purges with a maximum dewpoint of -37.2°C (-35°F) of specific satellite components can be provided at a rate of up to 14.2 standard m^3/hr (500 standard ft^3/hr).

3.1.2 Radiation and Electromagnetics

To ensure that electromagnetic compatibility (EMC) is achieved for each launch, the electromagnetic (EM) environment is thoroughly evaluated. The launch services customer will be required to provide all spacecraft data necessary to support EMC analyses (See Appendix C) used for this purpose.

3.1.2.1 Launch Vehicle-Generated Radio Environment—Launch vehicle intentional transmissions are limited to the S-band telemetry transmitters at 10.8 dBW or 15.45 dBW depending on LV antenna configuration (for operation with the Tracking and Data Relay Satellite System [TDRSS]) and the C-band beacon transponder at 28.5 dBW (peak) or 26 dBm (average).

Figure 3.1.2.1-1 shows the theoretical worst-case antenna radiation environment generated by the launch vehicle. The curve is based on transmitter fundamental and “spurious output” requirements and assumes (1) maximum transmit output power, (2) maximum antenna gain and minimum passive line loss (i.e., measured values at transmit frequency applied across the entire frequency spectrum), and (3) straight-line direct radiation. Actual levels encountered by the satellite (influenced by many factors) can

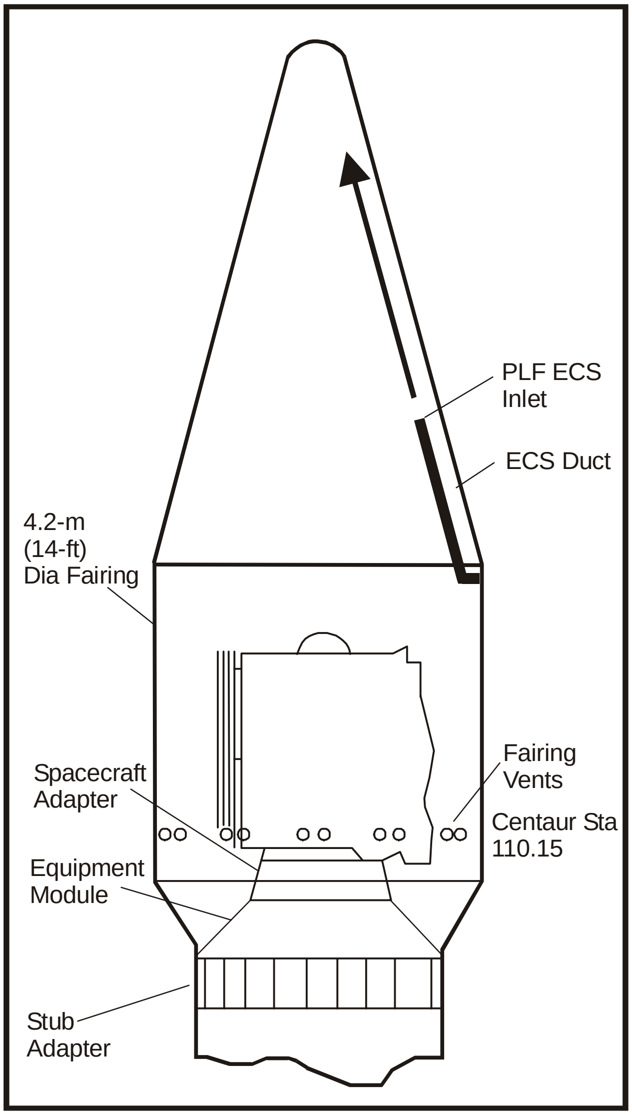


Figure 3.1.1-2 The PLF air conditioning system provides a controlled thermal environment during ground checkout and prelaunch.

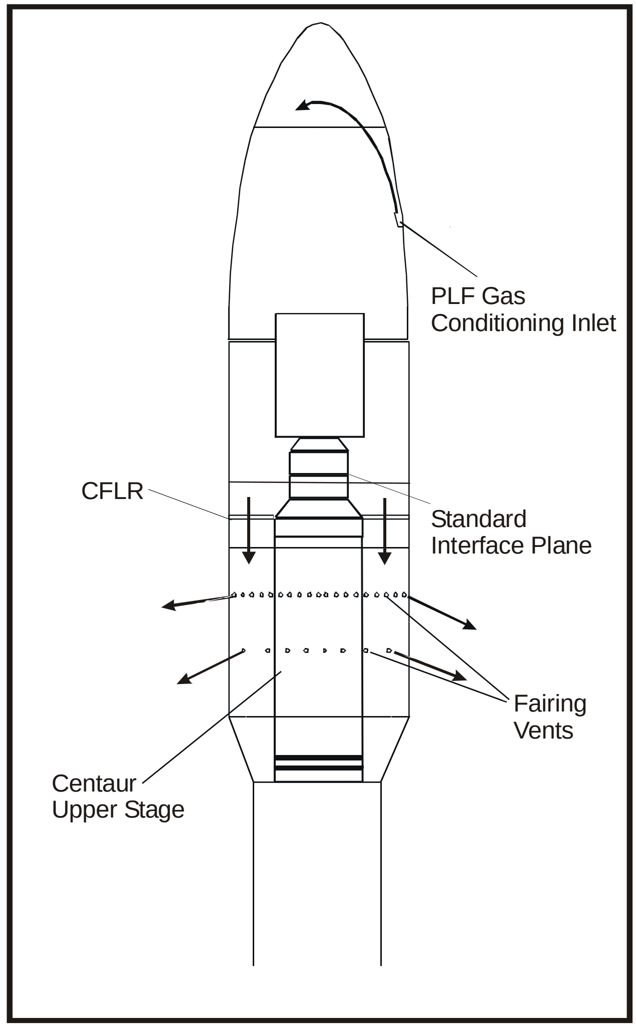


Figure 3.1.1-3 5-m PLF Environmental Conditioning System Provides a Controlled Thermal Environment During Ground Checkout and Prelaunch

only be less than the levels depicted in the figure. Initial reductions are provided to the user on determination of which launch vehicle and payload adapter will be used.

3.1.2.2 Launch Vehicle-Generated Electromagnetic Environment—The unintentional EM environments generated by the launch vehicle at the satellite location are depicted in Figures 3.1.2.2-1, 3.1.2.2-2, and 3.1.2.2-3. Actual levels to be encountered by the satellite will approach the typical levels depicted in the figures.

3.1.2.3 Launch Range Electromagnetic Environment—An EMC analysis will be performed to ensure EMC of the spacecraft/launch vehicle with the range environment, including updates as available.

Table 3.1.2.3-1 documents EM emitters in the vicinity of CCAFS. The EM environment of the launch range is based on information in TOR-2001 (1663)-1 “Cape Canaveral Spaceport Radio Frequency Environment**.**” Field intensities at Astrotech, LC-36 and LC-41 are provided. Data can be provided for other locations on request.

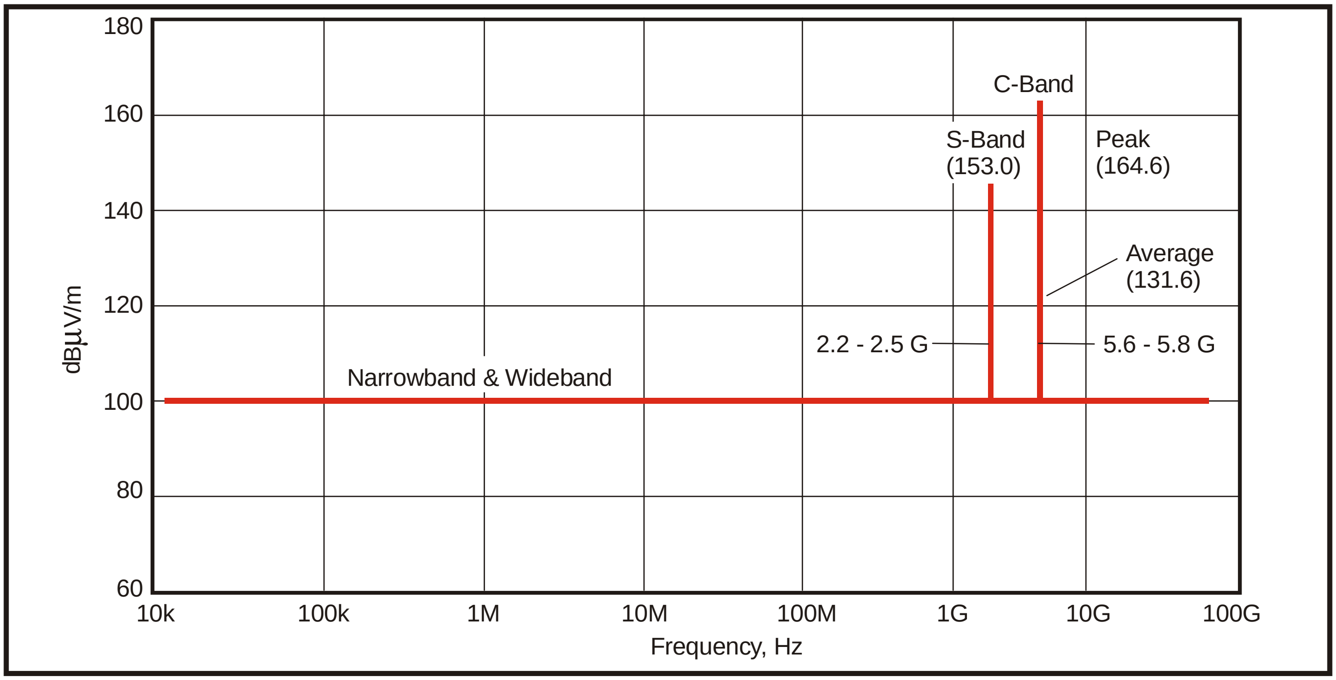


Figure 3.1.2.1-1 Launch Vehicle Field Radiation from Antennas

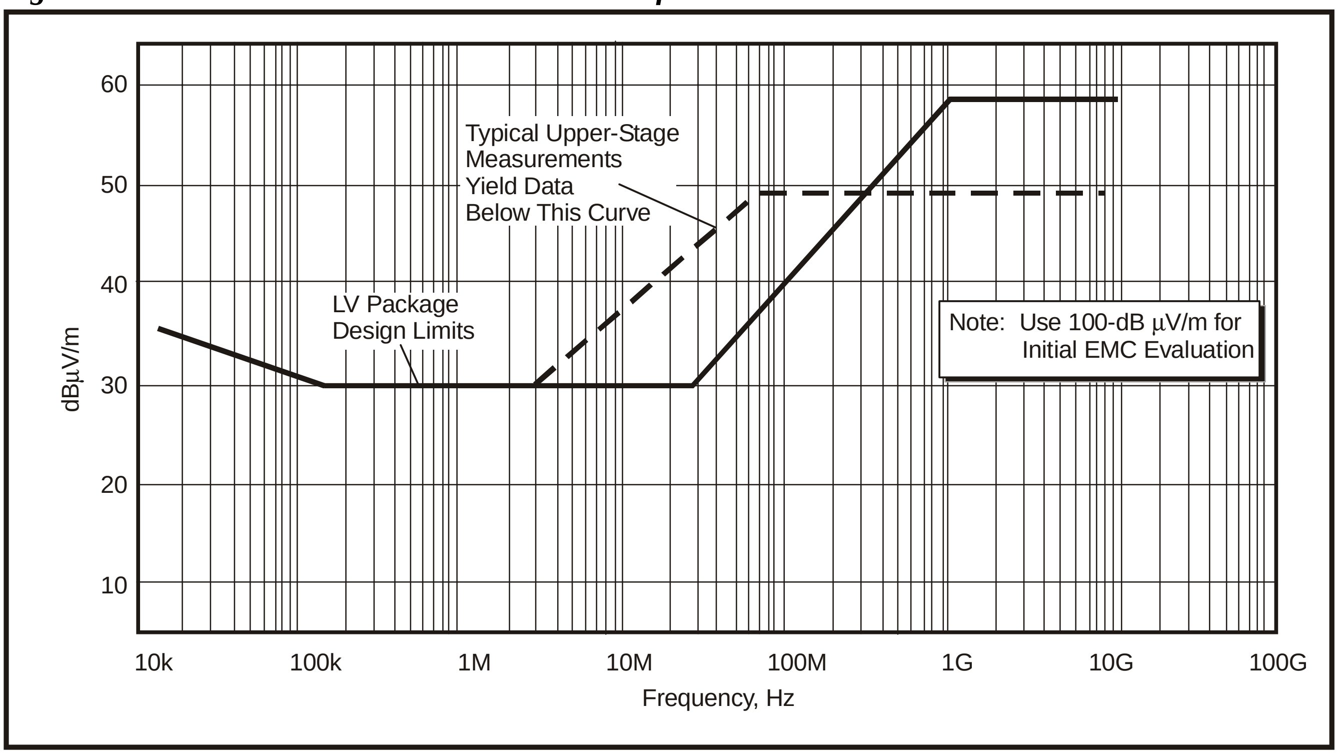


Figure 3.1.2.2-1 Launch Vehicle Electrical Field Radiation (Narrowband)

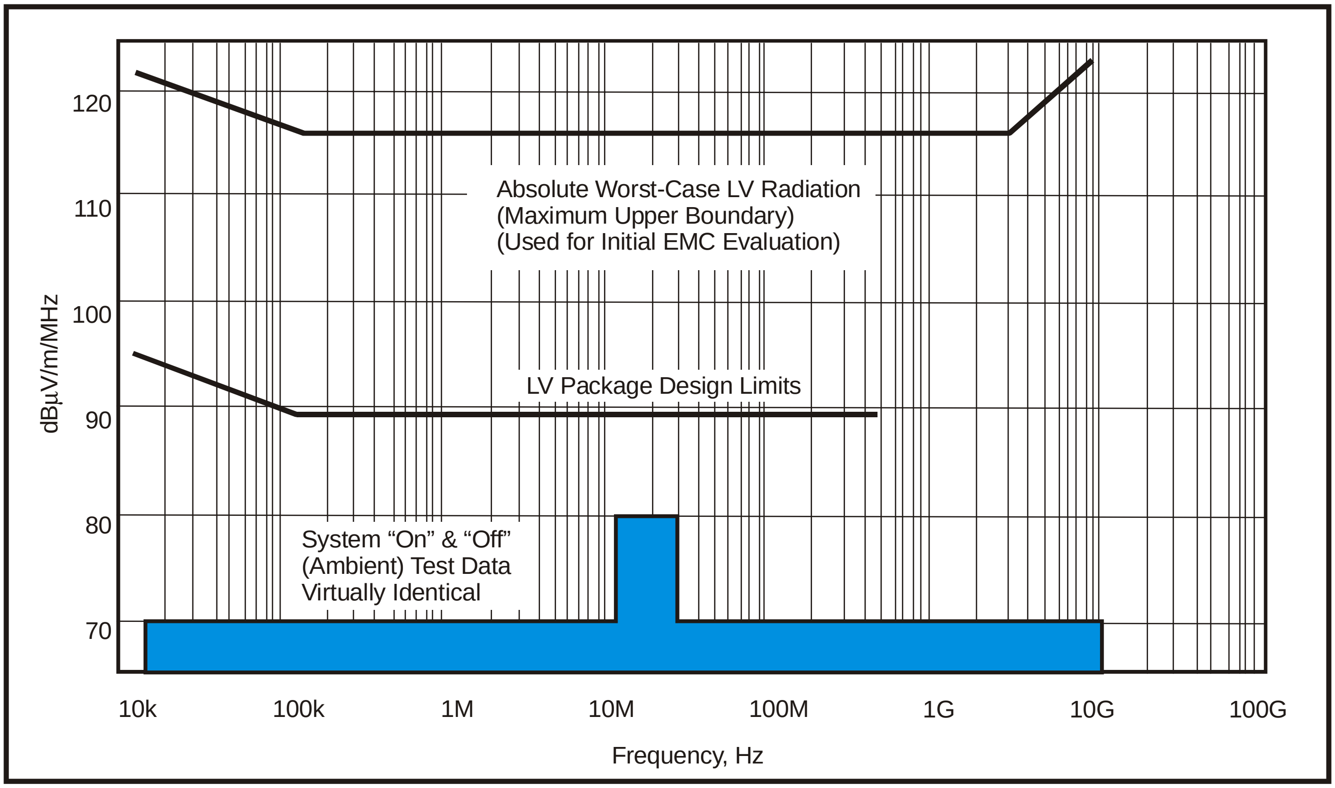


Figure 3.1.2.2-2 Launch Vehicle Electrical Field Radiation (Wideband)

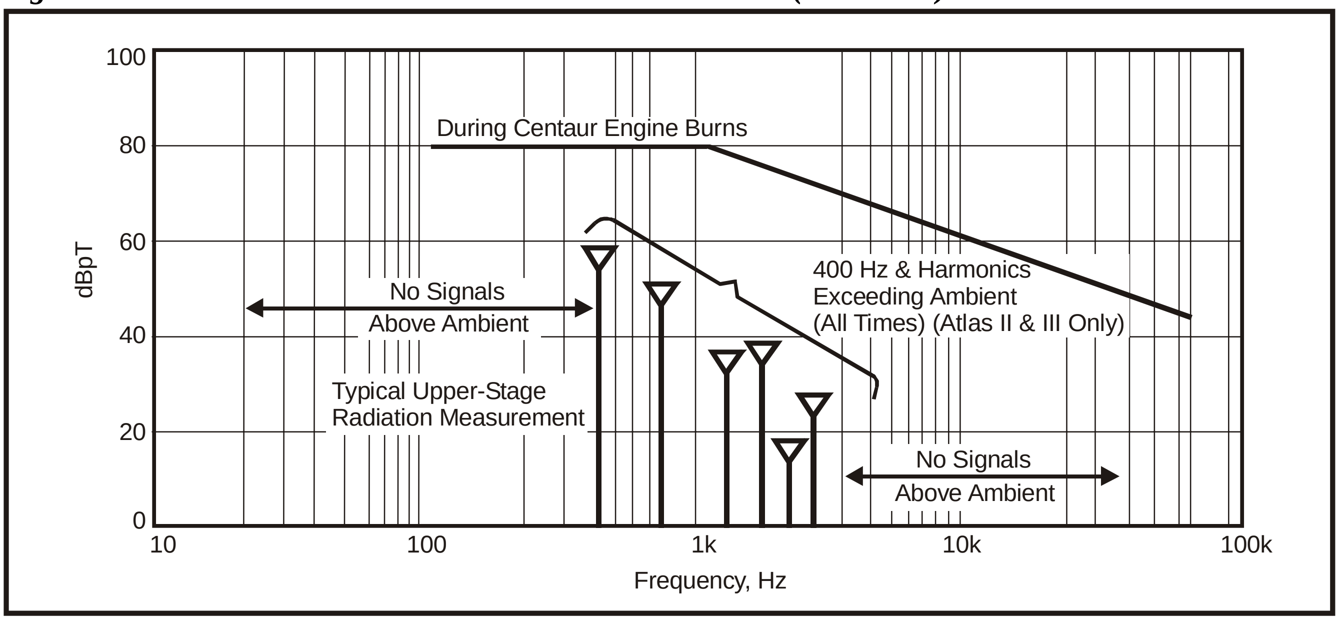


Figure 3.1.2.2-3 Launch Vehicle Magnetic Field Spurious Radiation (Narrowband)

Table 3.1.2.3-1 Worst-Case RF Environment for CCAFS

Emitter Name	Frequency, MHz	Theoretical Intensity, V/m	Duty Cycle	Mitigation
Combined LC-36				
Emitter Name	**Frequency, MHz**	**Theoretical Intensity, V/m**	**Duty Cycle**	**Mitigation**
Radar 0.14	5,690	103.6	0.0016	Procedure Mask
Radar 1.16	5,690	158.2	0.00064	Procedure Mask
Radar 1.39	5,690 & 5,800	77.5	0.005	Procedure Mask
Radar 19.14	5,690	158.6	0.0016	Procedure Mask
Radar 19.17	5,690	26.8	0.0008	Procedure Mask
Radar 28.14	5,690	16.7	0.0016	Topography
Radar 1.8	9,410	4.8	0.0012	No (OD10040)
Radar ARSR-4	1,244.06 & 1,326.92	1.6	0.0006	None
Radar GPN-20	2,750 & 2,840	7.5	0.0008	None
WSR-74C	5,625	15.8	0.0064	None
WSR-88D	2,879	16.3	0.006	None
GPS Gnd Station	1,784	7.1	CW	Ops Min 3°
NASA STDN	2,025 & 2,120	1.0	CW	None
Astrotech				
Emitter Name	**Frequency, MHz**	**Theoretical Intensity, V/m**	**Duty Cycle**	**Mitigation**
Radar 0.14	5,690	71.8	0.0016	Procedure Mask
Radar 1.16	5,690	24.5	0.00064	Procedure Mask[1]
Radar 1.39	5,690 & 5,800	17.4	0.005	Procedure Mask
Radar 19.14	5,690	108.9	0.0016	Procedure Mask[1]
Radar 19.17	5,690	34.2	0.0008	Procedure Mask[1]
Radar 28.14	5,690	15.4	0.0016	Topography
Radar 1.8	9,410	0.8	0.0012	Procedure Mask
Radar ARSR-4	1,244.06 & 1,326.92	1.7	0.0006	None
Radar GPN-20	2,750 & 2,840	5.2	0.0008	None
WSR-74C	5,625	10.6	0.0064	None
WSR-88D	2,879	13.8	0.006	None
GPS Gnd Station	1,784	0.9	CW	Ops Min 3°
NASA STDN	2,025 & 2,120	1.0	CW	None
LC-41				
Emitter Name	**Frequency, MHz**	**Theoretical Intensity, V/m**	**Duty Cycle**	**Mitigation**
Radar 0.14	5690	71.7	0.0016	Procedure Mask
Radar 1.16	5690	52.6	0.00064	Procedure Mask
Radar 1.39	5690 & 5800	57.5	0.005	Procedure Mask
Radar 19.14	5690	106.5	0.0016	Procedure Mask
Radar 19.17	5690	55.1	0.0008	Procedure Mask
Radar 28.14	5690	15.5	0.0016	Topography
Radar 1.8	9410	2.0	0.0012	Procedure Mask
Radar ARSR-4	1,244.06 & 1,326.92	1.4	0.0006	None
Radar GPN-20	2750 & 2840	5.2	0.0008	None
WSR-74C	5625	10.6	0.0064	None
WSR-88D	2879	12.7	0.006	None
GPS Gnd Station	1784	2.3	CW	Ops Min 3°
NASA STDN	2025 & 2120	1.0	CW	None

Note: Sources Taken from Aerospace Report TOR-2001 (1663)-1, "Cape Canaveral Spaceport Radio Frequency Environment," October 2000
[1]See TOR-2001 (1663)-1, Sect. 2.2.1 Astrotech
Avg V/m = Pk, V/m*sqrt (Duty Cycle); CW = Continuous Wave
Shaded Blocks Indicate Emitters Without Specific Mechanical or Software Mitigation Measures
In-Flight Levels for Tracking Radars (0.14, 1.16, 1.39, 19.14, & 19.17) Are 20 V/m

Table 3.1.2.3-2 lists EM emitters in the vicinity of VAFB. These data are based on an EM site survey conducted at SLC-3E in June 1997.

Launch trajectory and uncontrolled emitters in the area, such as nearby cruise ships or Navy vessels, may cause RF environments to exceed the levels shown in these tables. Emitters less than 1 V/m are not recorded.

3.1.2.4 Spacecraft-Generated EMC Environment Limitation—During ground and launch operation timeframes through spacecraft separation, any spacecraft electromagnetic interference (EMI) radiated emissions (including antenna radiation) should not exceed values depicted in Figure 3.1.2.4-1. Launch vehicle/spacecraft external interfaces (EMI-conducted emissions) must be examined individually.

Each spacecraft will be treated on a mission-peculiar basis. Assurance of the launch vehicle/spacecraft EMC with respect to payload emissions will be a shared responsibility between Lockheed Martin and the spacecraft contractor.

3.1.3 Prelaunch Contamination Control and Cleanliness

Launch vehicle hardware that comes into contact with the spacecraft environment has been designed and manufactured according to strict contamination control guidelines. This hardware is defined as "contamination critical." Contamination-critical surfaces include the Centaur equipment module, PLF

Table 3.1.2.3-2 Worst-Case RF Environment for VAFB

SLC-3				
Emitter Name	**Frequency, MHz**	**Measured Intensity, V/m**	**Duty Cycle**	**Mitigation**
CT-2	416.5	2.54	CW	None
ARSR-4	1,262/1,345	1.4	0.0006	None
AN/GPN-12	2,800	8.07	0.000835	None
FPS-16-1	5,725	150.66	0.001	Procedure Mask
FPS-77	5,450 & 5,650	11.4 (Est)	0.0064	None
HAIR	5,400-5,900	279.25	0.002	Procedure Mask
MOTR	5,400-5,900	31.62	0.005	Procedure Mask
TPQ-18	5,840	874.0	0.0016	Procedure Mask
TPQ-39	9,180	50.47	0.00024	Procedure Mask
NEXRAD	2,890	30.27 (Est)	0.002	None
DRWP	404.37	NSO	0.067	Vertical Emitter
Astrotech				
Emitter Name	**Frequency, MHz**	**Theoretical Intensity, V/m**	**Duty Cycle**	**Mitigation**
CT-2	416.5	1.14	CW	None
ARSR-4	4,262/1,345	1/5	0.0006	Procedure Mask
AN/GPN-12	2,800	100+	0.000835	None
FPS-16-1	5,725	65	0.001	Procedure Mask
FPS-77	5,450 & 5,650	27	0.0064	None
HAIR	5,400-5,900	670	0.002	Procedure Mask
MOTR	5,400-5,900	14	0.005	Procedure Mask
TPQ-18	5,840	416	0.0016	Procedure Mask
TPQ-39	9,180	24	0.00024	Procedure Mask
NEXRAD	2,890	30	0.002	None
DRWP	404.37	NSO	0.0067	None

Note: Theoretical intensity data taken from the "Field Strength Measurement Data Summary at the Western Range" March 7, 1994, prepared for the 30th Space Wing.
CT: Command Transmitter; CW: Continuous Wave
Estimated (Est) values are theoretical and were not operating when measurement were taken.

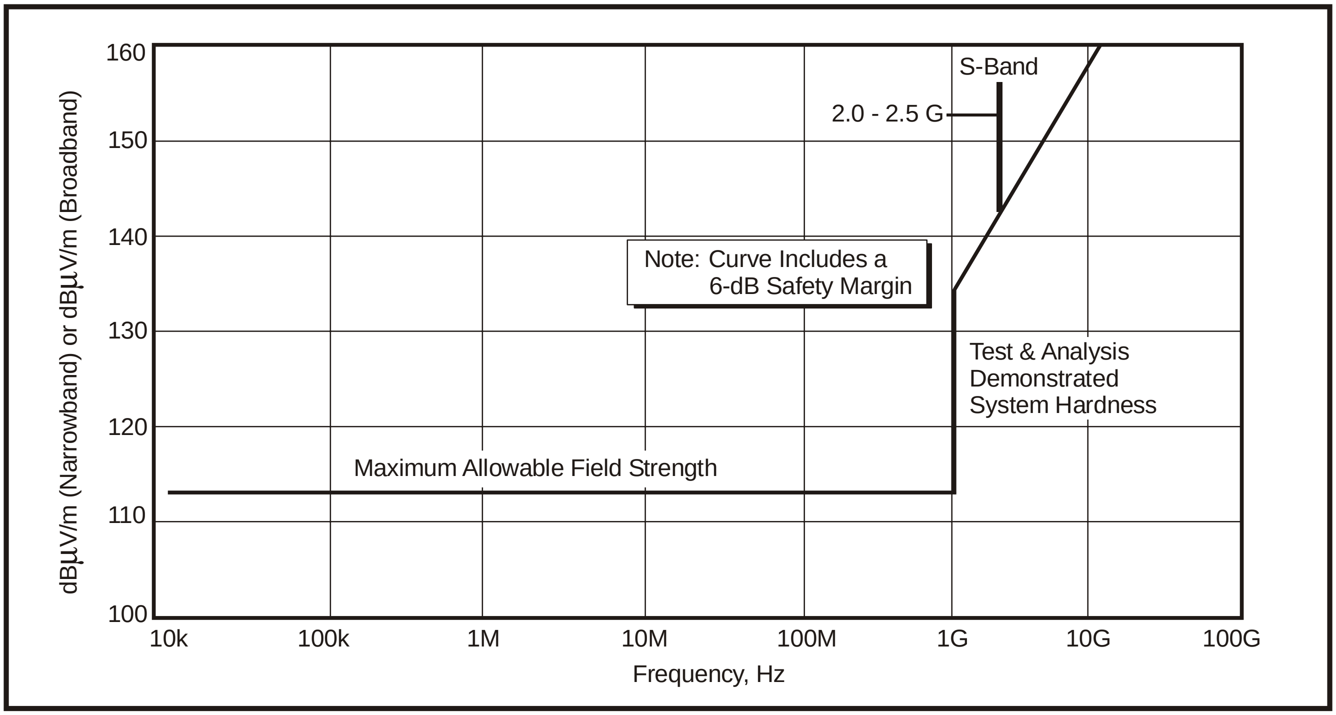


Figure 3.1.2.4-1 Spacecraft Electric Field Radiation Impingement on Launch Vehicle

interior, boattail interior, payload adapter, and for the 5-m PLFs, the top of the Centaur forward load reactor (CFLR) (See Section A.2.3).

In addition, ground operations at the launch site have been designed to ensure a clean environment for the spacecraft. A comprehensive Contamination Control Plan has been written to identify these requirements and procedures. A mission-unique appendix will be written to supplement the contamination control plan if the mission-specific requirements identified in the ICD are more stringent than those baselined in the control plan. Some guidelines and practices used in the plan are found in the following paragraphs.

Analysis of launch vehicle contamination of the spacecraft is discussed in Section 5.2.11.

3.1.3.1 Contamination Control Before Launch Site Delivery

Design and Assembly—Contamination control principles are used in design and manufacturing processes to limit the amount of contamination from launch vehicle components. Interior surfaces include maintainability features to facilitate the removal of manufacturing contaminants. Contamination-critical hardware is entered into a controlled production phase in which the hardware is cleaned and maintained clean to prevent contaminants in difficult-to-clean places at the end of production. To support this effort, final assembly of payload adapters, the PLF, and the Centaur vehicle is performed in a Class 100,000 facility to ensure that hardware surfaces and, in particular, any entrapment areas, are maintained at an acceptable level of cleanliness before shipment to the launch site. Inspection points are provided to verify cleanliness throughout the assembly process. Plastic wrapping is used to protect critical surfaces during contaminant-generating activities.

Materials Selection—In general, materials are selected for contamination-critical hardware inside the PLF that will not become a source of contamination to the spacecraft. Metallic or nonmetallic materials that are known to chip, flake, or peel are prohibited. Materials that are cadmium-plated, zinc-plated, or made of unfused electrodeposited tin cannot be used inside the PLF volume. Corrosion-resistant materials are selected wherever possible and dissimilar materials are avoided or protected according to MIL-STD-889B. Because most nonmetallic materials are known to exhibit some outgassing, these materials are evaluated against criteria that were developed using National Aeronautics and Space Administration (NASA) SP-R-0022 as a starting point.

3.1.3.2 Contamination Control Before Spacecraft Encapsulation

Cleanliness Levels—Contamination-critical hardware surfaces are cleaned and inspected to Visibly Clean Level 2. These checks confirm the absence of all particulate and molecular contaminants visible to the unaided eye at a distance of 15.2-45.7 cm (6-18 in.) with a minimum illumination of 1,076 lumen/m2 (100 fc). Hardware that is cleaned to this criterion at the assembly plant is protected to maintain this level of cleanliness through shipping and encapsulation.

Contingency cleaning may also be required to restore this level of cleanliness if the hardware becomes contaminated. Contingency cleaning procedures outside of the encapsulation facility before encapsulation are subject to Lockheed Martin engineering approval. The cognizant spacecraft engineer must approve any required cleaning of launch vehicle hardware in the vicinity of the spacecraft.

Certain spacecraft may require that contamination-critical hardware surfaces be cleaned to a level of cleanliness other than Visibly Clean Level 2. Because additional cleaning and verification may be necessary, these requirements are implemented on a mission peculiar basis.

PLF Cleaning Techniques—Lockheed Martin recognizes that cleaning of large, interior PLF surfaces depends on implementation of well-planned cleaning procedures. To achieve customer requirements, all cleaning procedures are verified by test and reviewed and approved by Material and Processes Engineering. Final PLF cleaning and encapsulation is performed in a Class 100,000 facility.

Cleanliness Verification—All contamination-critical hardware surfaces are visually inspected to verify Visibly Clean Level 2 criteria as described above. For Atlas V, contamination-critical surfaces are also verified to have less than 1 mg/ft^2 of nonvolatile residue (NVR). The additional verification techniques shown below can be provided on a mission-unique basis:

1) Particulate Obscuration—Tape lift sampling,
2) Particulate Obscuration—Ultraviolet light inspection,
3) Nonvolatile Residue (NVR)—Solvent wipe sampling,
4) Particulate and Molecular Fallout—Witness plates.

3.1.3.3 Contamination Control After Encapsulation

Contamination Diaphragm—Contamination barriers are provided (for both 4-m and 5-m PLFs) to protect the spacecraft from possible contamination during transport and hoist. After the two halves of the PLF are joined, the encapsulation is completed by closing the aft opening with a ground support equipment (GSE) Kevlar-reinforced, Teflon-coated diaphragm. The toriodal diaphragm stretches from the payload adapter to the aft end of the PLF cylinder and creates a protected environment for the spacecraft from mating to the Centaur through final LV closeouts prior to launch, typically one day before launch.

For the 5-m payload fairings, the entire Centaur is contained within the PLF. In this case, the launch vehicle is designed so the direction of the conditioned airflow is always in the aft direction (through the CFLR deck) to minimize spacecraft exposure from sources aft of the CFLR.

PLF Purge—After encapsulation, the PLF environment is continuously purged with filtered nitrogen or high-efficiency particulate air (HEPA)-filtered air to ensure the cleanliness of the environment.

Personnel Controls—Personnel controls are used to limit access to the PLF to maintain spacecraft cleanliness. Contamination control training is provided to all launch vehicle personnel working in or around the encapsulated PLF. Lockheed Martin provides similar training to spacecraft personnel working on the spacecraft while at the launch complex to ensure that they are familiar with the procedures.

Complexes 36A and 36B—Standard access to the encapsulated spacecraft is performed from workstands situated on PLF Access Level 14 for LC-36A and Level 29 for LC-36B. Clean work procedures and personnel control are established to maintain the spacecraft environment within the PLF to Class 100,000 standards. Garments are provided to personnel working inside the PLF to provide optimum cleanliness control as dictated by spacecraft requirements.

Complex 3E Controlled Work Area—The encapsulated spacecraft is contained within an environmentally controlled area (ECA) on Mobile Service Tower (MST) Levels 8 through 15. The conditioned air supply to this facility is 90% filtered (removing 90% of particles 0.7 microns and larger).

The Vertical Integration Facility (VIF) at LC-41—Access to the encapsulated spacecraft is performed from work stands situated on Levels 5, 6, and 7. Work procedures and personnel control are established to maintain the spacecraft environment within the PLF to Class 100,000 standards. Garments are provided to personnel making PLF entry at the Centaur Equipment Module (CEM) station to provide optimum cleanliness control as dictated by spacecraft requirements. A portable clean room tent is available for entry through mission-peculiar access doors as required (portable between VIF levels 5, 6, 6.5, and 7).

3.1.3.4 Payload Fairing Helium Environment in Prelaunch Operations—The volume between the Centaur LH_2 tank and the equipment module is purged with helium while the vehicle is on the pad with the LH_2 tank loaded, to prevent condensation on the hydrogen tank forward bulkhead. Although the CEM is sealed, some helium may leak into the payload compartment. The helium mixes with the GN_2 being provided by the PLF and equipment module environmental control systems. At T-8 seconds, a pyroactivated helium vent door opens, venting the equipment module helium into the payload compartment. Environmental control and helium purge systems are terminated at T-0 in a normal launch. During ascent, the payload compartment vents to negligible pressure approximately 3 minutes after launch. In case of an aborted launch attempt with the helium vent door open, the helium flow is shut off shortly after abort. The flow is allowed to continue during detanking if the vent door remains closed.

Measured helium concentrations above the Centaur equipment module are variable and no specific exposure level can be guaranteed until further testing is performed. Some solutions, such as GN_2 purge of sensitive components, are available to helium sensitive spacecraft on a mission-unique basis.

The interstage adapter (ISA) is filled with helium for launch of the Atlas V 500 and HLV configurations to attenuate the acoustic environment within the ISA. To accomplish this, the ISA is purged with helium for the last 5 minutes before launch. The ISA is vented through 5-m PLF the base module compartment to the PLF vents just below the CFLR deck. Flow of this helium through the CFLR into the payload compartment is minimized by the flow of GN_2 from the payload module ECS, which enters into the payload compartment at a high, flow rate and is vented down through the CFLR and out the vents in the PLF base module compartment. However, no specific helium exposure level can be guaranteed at this time.

3.2 LAUNCH AND FLIGHT ENVIRONMENTS

This section describes general environmental conditions that may be encountered by a spacecraft during launch and flight of the Atlas launch vehicle. A thorough description of each environment is presented (Section 3.2), as well as an outline of necessary spacecraft compatibility testing (Section 3.3). All flight environments defined in this section are maximum expected levels and do not include margins typically associated with qualification tests. Verification analysis necessary to assure spacecraft compatibility with Atlas environments is performed during the Mission Integration process as described in Section 5.2.

3.2.1 Spacecraft Design Load Factors (DLF)

Design Load Factors are provided in Table 3.2.1-1 and Figure 3.2.1-1 for use in preliminary design of primary structure and/or evaluation of compatibility of existing spacecraft with Atlas launch vehicles. Factors are provided for each transient event that produces significant spacecraft loading, including any engine-induced thrust oscillation effects (CLE—Centaur Longitudinal Event and BETC—Booster Engine Thrust Coupling). The total DLF for any direction can be determined by addition of the quasi-steady-state and oscillatory dynamic components provided. No uncertainty factors related to spacecraft design maturity are included in the DLF definitions.

The load factors were derived for application to the center of gravity (cg) of a rigid spacecraft to generate a conservative estimate of interface loading. The actual responses of a spacecraft due to launch vehicle transients will depend on its specific static and structural dynamic characteristics; however, the values provided have generally proven conservative for spacecraft in the weight range of 1,800 to 4,500 kg for Atlas IIAS, Atlas IIIA, and IIIB vehicles, 1,800 to 9,100 kg for Atlas V 400 vehicles, and 4,500 to 19,000 kg for the Atlas V 500 vehicles. The spacecraft cantilevered fundamental mode frequencies are assumed to be a minimum of 8 Hz lateral and 15 Hz axial to ensure applicability of the design load factors. Spacecraft that do not meet these criteria will require configuration specific analyses for assessing compatibility with Atlas launch vehicles.

Table 3.2.1-1 Spacecraft Design Limit Load Factors

Load Condition	Direction	Atlas IIAS		Atlas III		Atlas V 40Z, 50Z		Atlas V 4YZ, 5YZ	
		Steady State, g	Dynamic, g	Steady State, g	Dynamic, g	Steady State, g	Dynamic, g	Steady State, g	Dynamic, g
Launch	Axial	1.2	±1.1	1.2	±1.1	1.2	±0.5	1.8	±2.0
	Lateral	0.0	±1.3	0.0	±1.3	0.0	±1.0	0.0	±2.0
Flight Winds	Axial	1.7–2.7	±0.8	1.3–2.7	±0.3	1.0–2.8	±0.5	1.0–2.8	±0.5
	Lateral	0.4	±1.6	0.4	±1.6	±0.4	±1.6	±0.4	±1.6
Strap–On Separation	Axial	—	—	—	—	—	—	3.3	±0.5
	Lateral	—	—	—	—	—	—	0.0	±0.5
BECO/BETC (Max Axial)	Axial	5.0	±0.5	5.5	±0.5	5.5	+0.5	5.5	+0.5
	Lateral	0.0	±0.5	0.0	±0.5	0.0	±0.5	0.0	±0.5
(Max Lateral)	Axial	2.5-1.0	±1.0	2.5-0.0	±1.0	3.0-0.0	±1.0	3.0-0.0	±1.0
	Lateral	0.0	±2.0	0.0	± 1.5	0.0	± 1.5	0.0	± 1.5
SECO	Axial	2.0-0.0*	±0.4	—	—	—	—	—	—
	Lateral	0.0	±0.3	—	—	—	—	—	—
MECO/CLE (Max Axial)	Axial	4.5-0.0*	±1.0	4.5-0.0*	±1.0	4.5-0.0*	±1.0	4.5-0.0*	±1.0
	Lateral	0.0	±0.3	0.0	±0.3	0.0	±0.3	0.0	±0.3
(Max Lateral)	Axial	0.0	±2.0	0.0	±2.0	0.0	±2.0	0.0	±2.0
	Lateral	0.0	±0.6	0.0	±0.6	0.0	±0.6	0.0	±0.6

Sign Convention
Longitudinal Axis: + (Positive) = Compression
– (Negative) = Tension
± May Act in Either Direction

Lateral & Longitudinal Loading May Act Simultaneously During Any Flight Event
Loading Is Applied to the Spacecraft cg

"Y" in vehicle designator is number of SRBs & ranges from 1 to 3 (400 Series) or 1 to 5 (500 Series)
"Z" in vehicle designator is number of Centaur engines & is 1 or 2

* Decaying to Zero

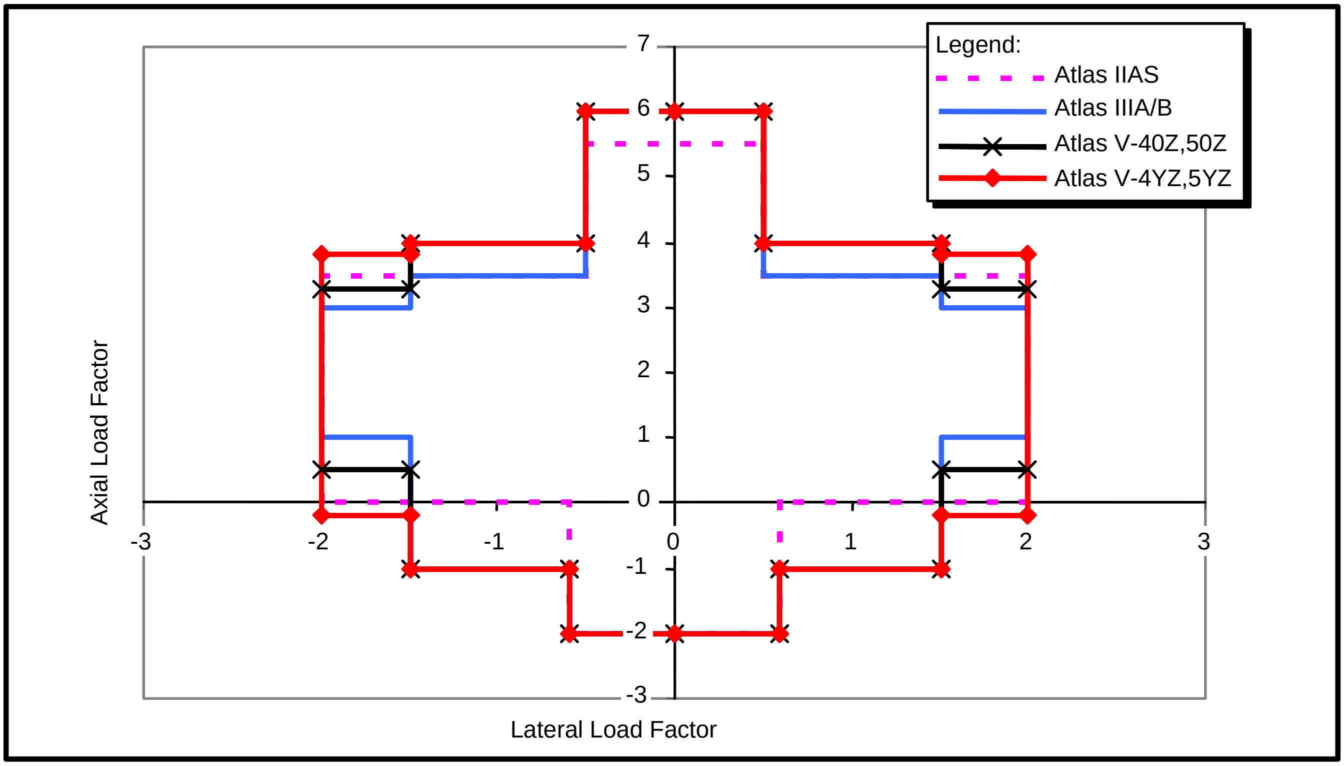


Figure 3.2.1-1 Spacecraft Design Limit Load Factors

Coupled loads analyses (CLA) are conducted as part of the mission integration activity to provide actual loads, accelerations, and deflections for both primary and secondary structure of the space vehicle for use in design, test planning, and verification of minimum margins of safety.

3.2.2 Acoustics

The spacecraft is exposed to an acoustic environment throughout the boost phase of flight until the vehicle is out of the sensible atmosphere. Two portions of flight have significantly higher acoustic levels than any other. The highest acoustic level occurs for approximately 10 seconds during liftoff, when the acoustic energy of the engine exhaust is being reflected by the launch pad. The other significant level occurs for approximately 20 seconds during the transonic portion of flight and is due to transonic aerodynamic shock waves and a highly turbulent boundary layer. The acoustic level inside the PLF will vary slightly with different spacecraft due to acoustic absorption that varies with spacecraft size, shape, and surface material properties. Acoustic sound pressure levels for Atlas IIAS 4-m, Atlas III 4-m, Atlas V 4-m, and Atlas V 5-m fairing are provided in Figures 3.2.2-1, 3.2.2-2, 3.2.2-3, and 3.2.2-4, respectively. These figures represent the maximum expected environment based on a 95% probability and 50% confidence (limit level). The levels presented are for spacecraft of square cross-sectional shape with typical cross-sectional fill ratios of 50-60% for the 4-m fairing and 40-50% for the 5-m fairing. A mission-peculiar acoustic analysis is required for spacecraft with other fill factors. Spacecraft should be capable of functioning properly after 1-minute exposure to these levels.

An optional mission-peculiar acoustic panel design is available for the Atlas IIAS 4-m fairings (Fig. 3.2.2-5) to reduce high-frequency acoustic energy within the payload envelope. The acoustic panel configuration is standard for Atlas III and Atlas V.

For 4-m fairings using acoustic panels, special consideration should be given to components within 76 cm (30 in.) of fairing vents. Sound pressure levels for components near the vents are provided in Figure 3.2.2-6.

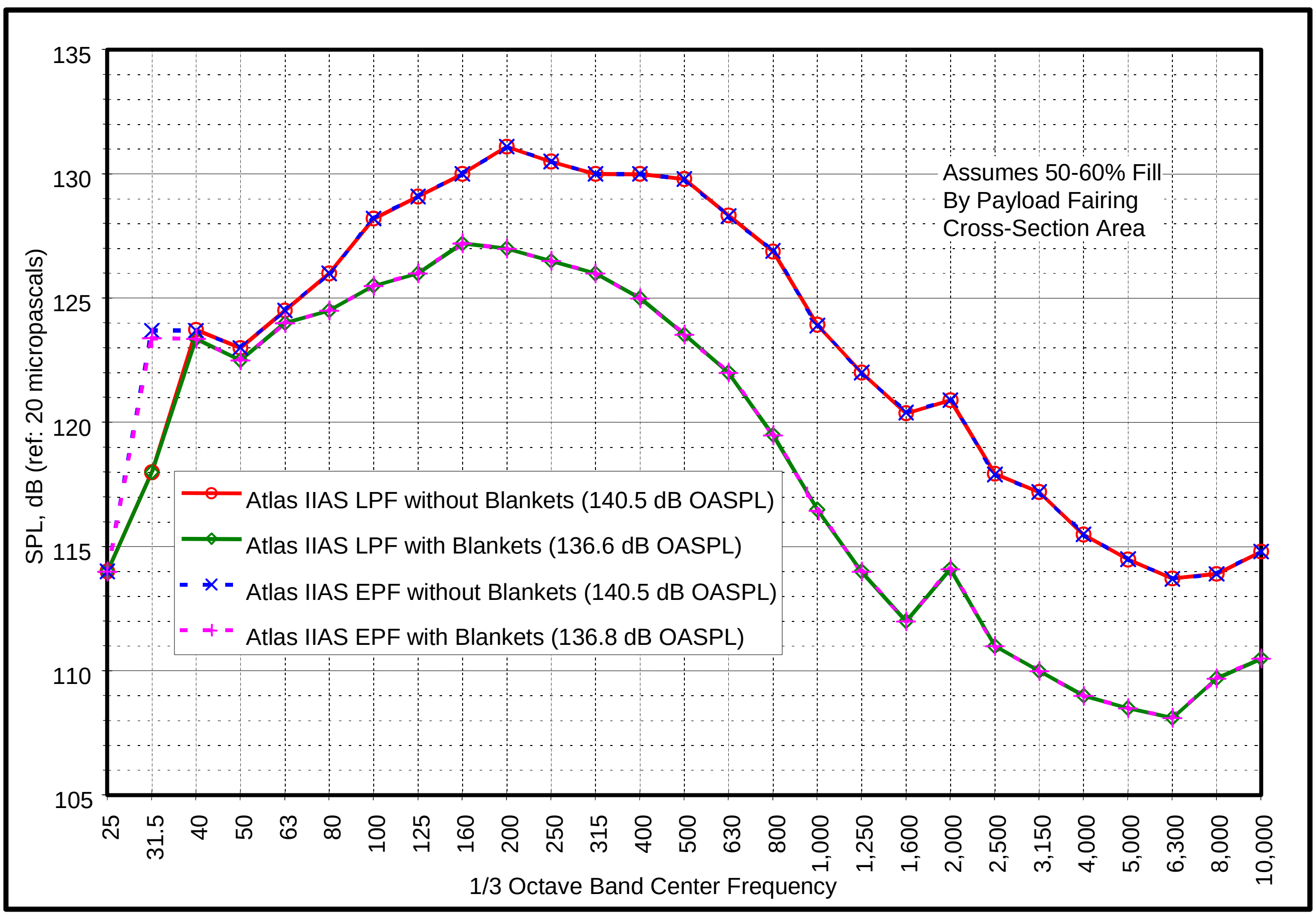


Figure 3.2.2-1 Acoustic Levels for Atlas IIAS with Large or Extended-Length Payload Fairings

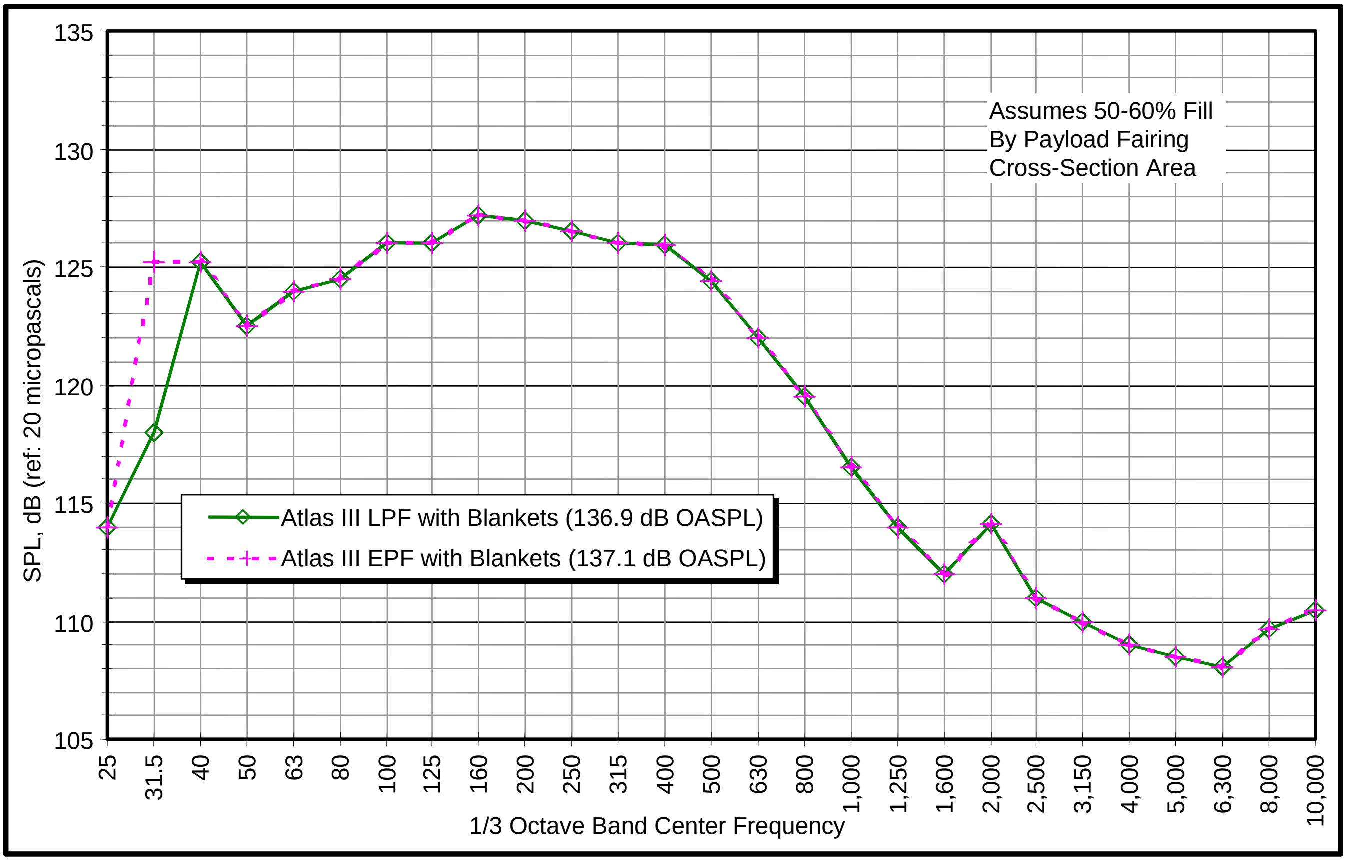


Figure 3.2.2-2 Acoustic Levels for Atlas III with Large or Extended-Length Payload Fairings

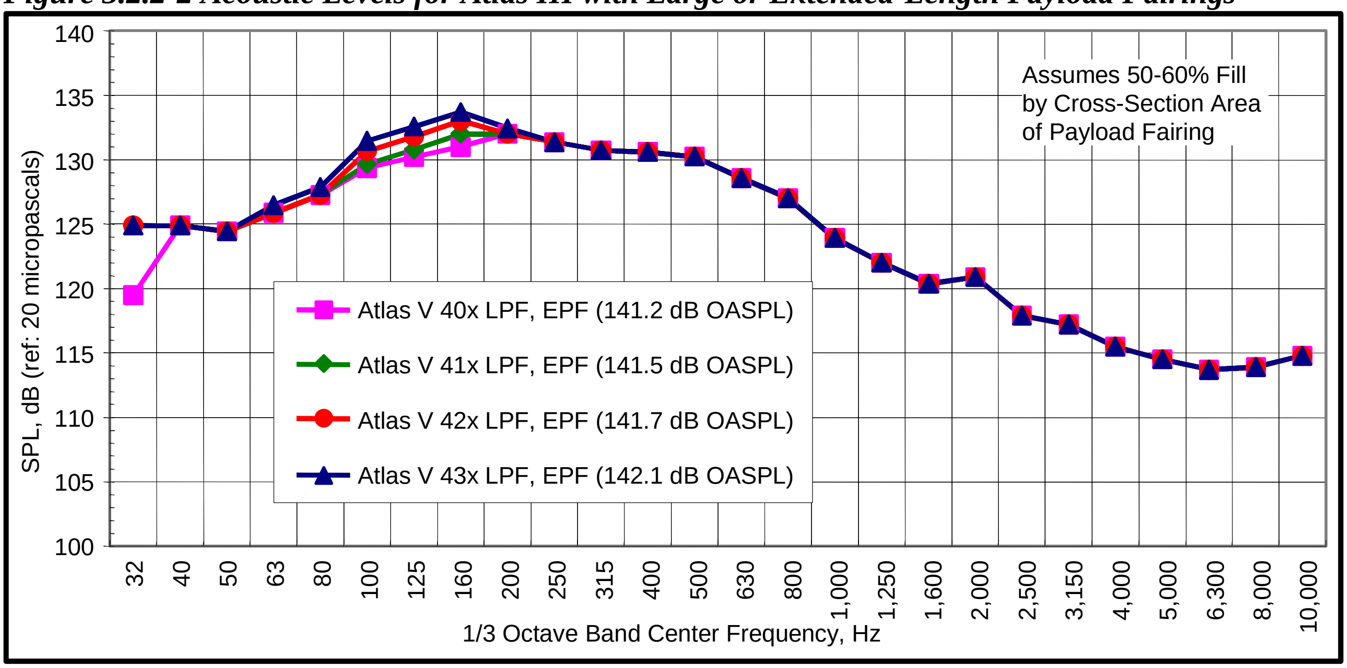


Figure 3.2.2-3 Acoustic Levels for Atlas V 400 Series with Large and Extended-Length Payload Fairings (with Blankets)

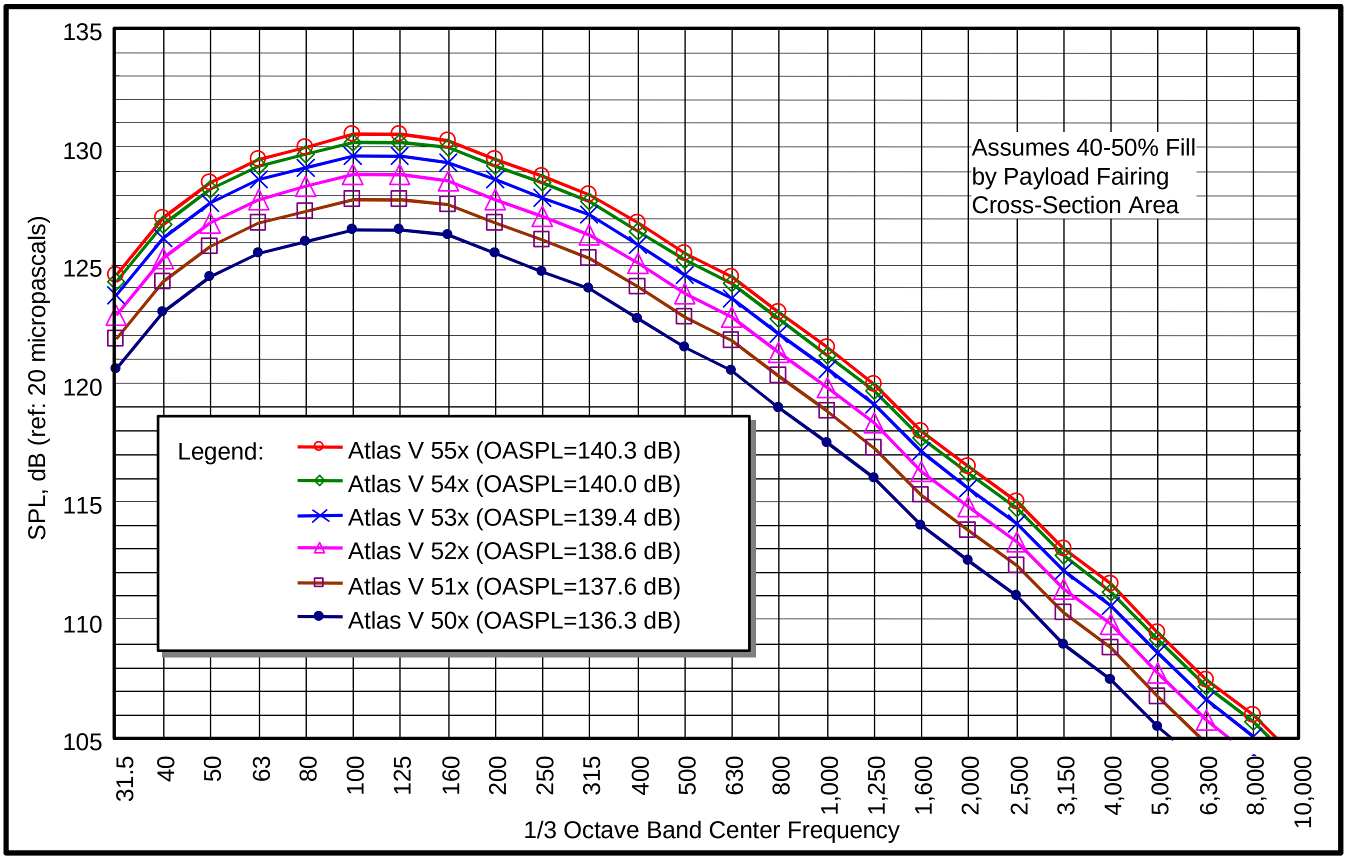


Figure 3.2.2-4 Acoustic Levels for Atlas V 500 Series with 5-m Payload Fairing

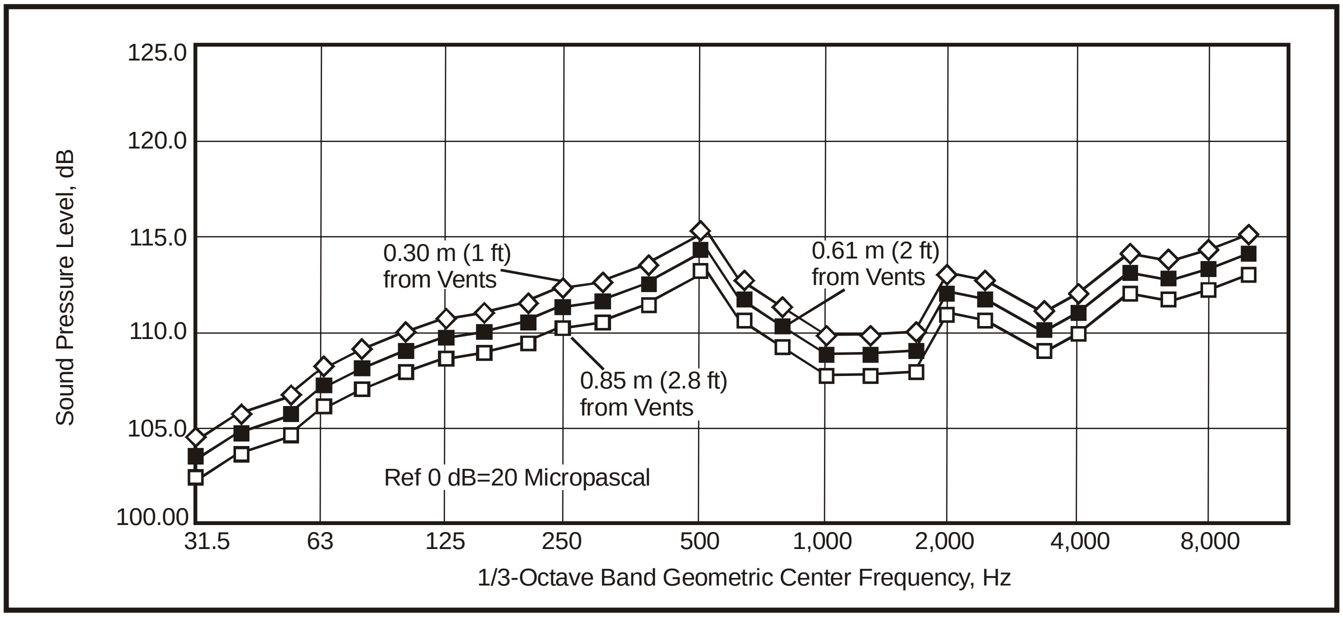


Figure 3.2.2-5 Acoustic Levels Near the Vents with the Large or Extended-Length Large Fairings

3.2.3 Vibration

The spacecraft is exposed to a vibration environment that may be divided into two general frequency ranges: (1) low-frequency quasi-sinusoidal vibration, and (2) high-frequency broadband random vibration.

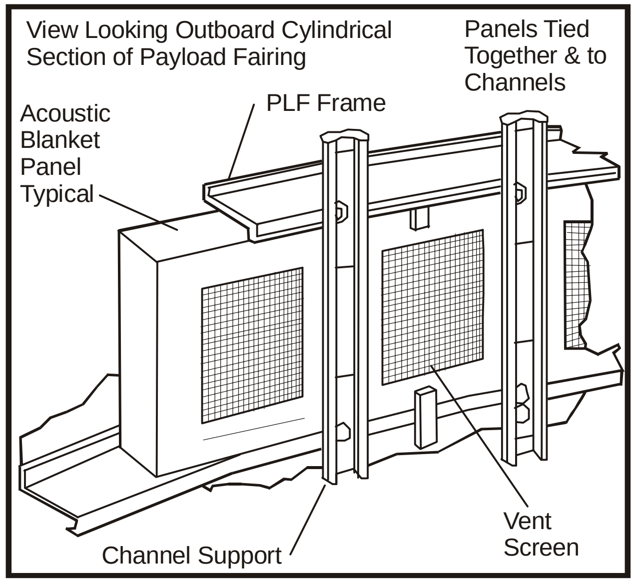


Figure 3.2.2-6 Acoustic Blanket Panels

The low-frequency vibration tends to be the design driver for spacecraft structure. The flight measured equivalent sine vibration near the spacecraft interface for Atlas IIAS, Atlas IIIA, and Atlas IIIB vehicle configurations is shown in Figure 3.2.3-1. The envelope shown is based on a 99% probability and 90% confidence statistical envelope. The maximum expected levels for the Atlas V 400 and 500 series vehicles are shown in Figure 3.2.3-2. Most peak responses occur for a few cycles during transient events, such as launch, gusts, booster engine cutoff (BECO), jettison events, and Centaur main engine cutoff (MECO). Other flight events produce multicycle responses such as buffet, Centaur Longitudinal Event (CLE), and Booster Engine Thrust Coupling (BETC).

The low-frequency transient response environment during Atlas/Centaur flight is characterized by a combination of the equivalent sinusoidal vibration specified at the spacecraft interface and coupled dynamic response analysis (CLA). Verification of minimum factors of safety is required by performance of a system-level vibration test (See Section 3.3). CLA is used to notch the test input in the 0- to 50-Hz

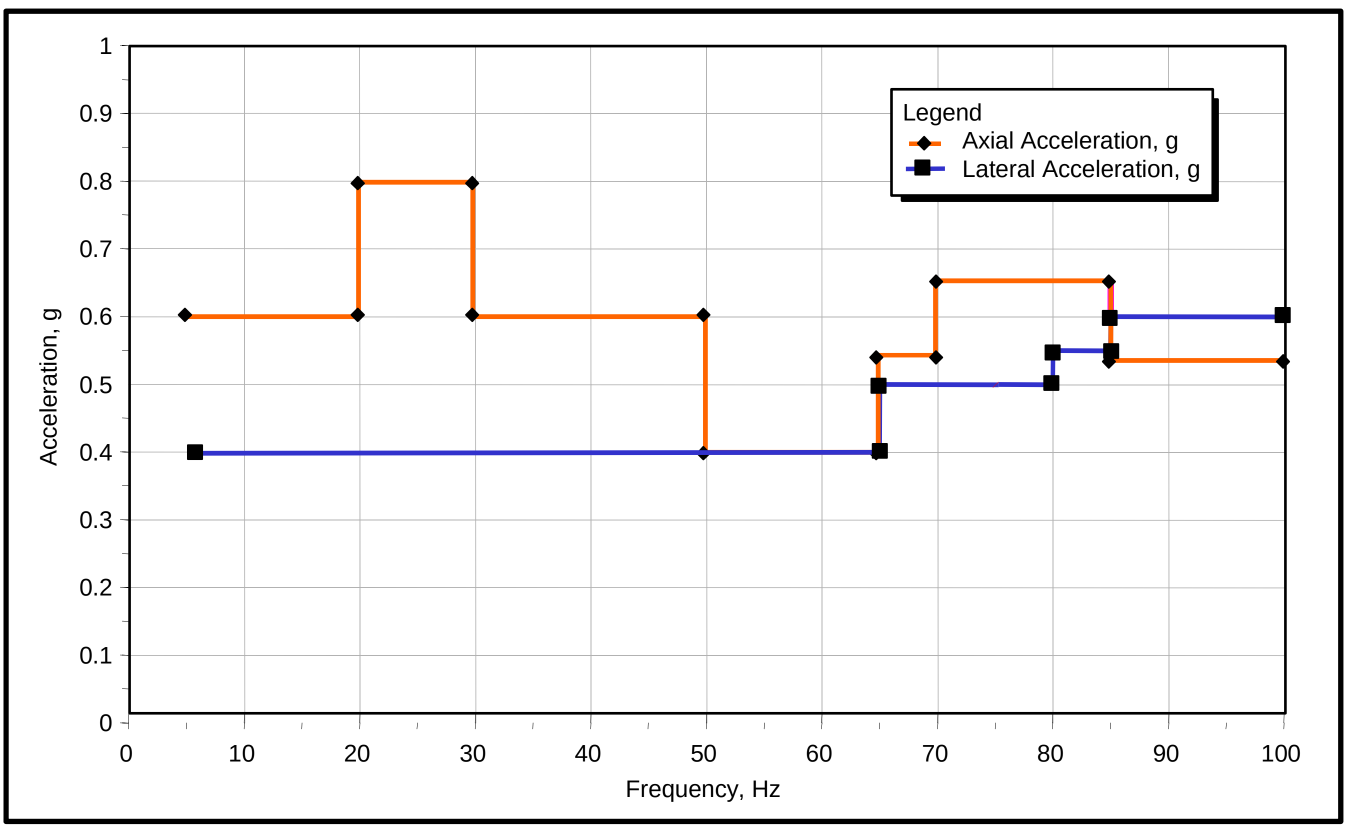


Figure 3.2.3-1 Quasi-Sinusoidal Vibration Levels for Atlas IIAS, IIIA, and IIIB

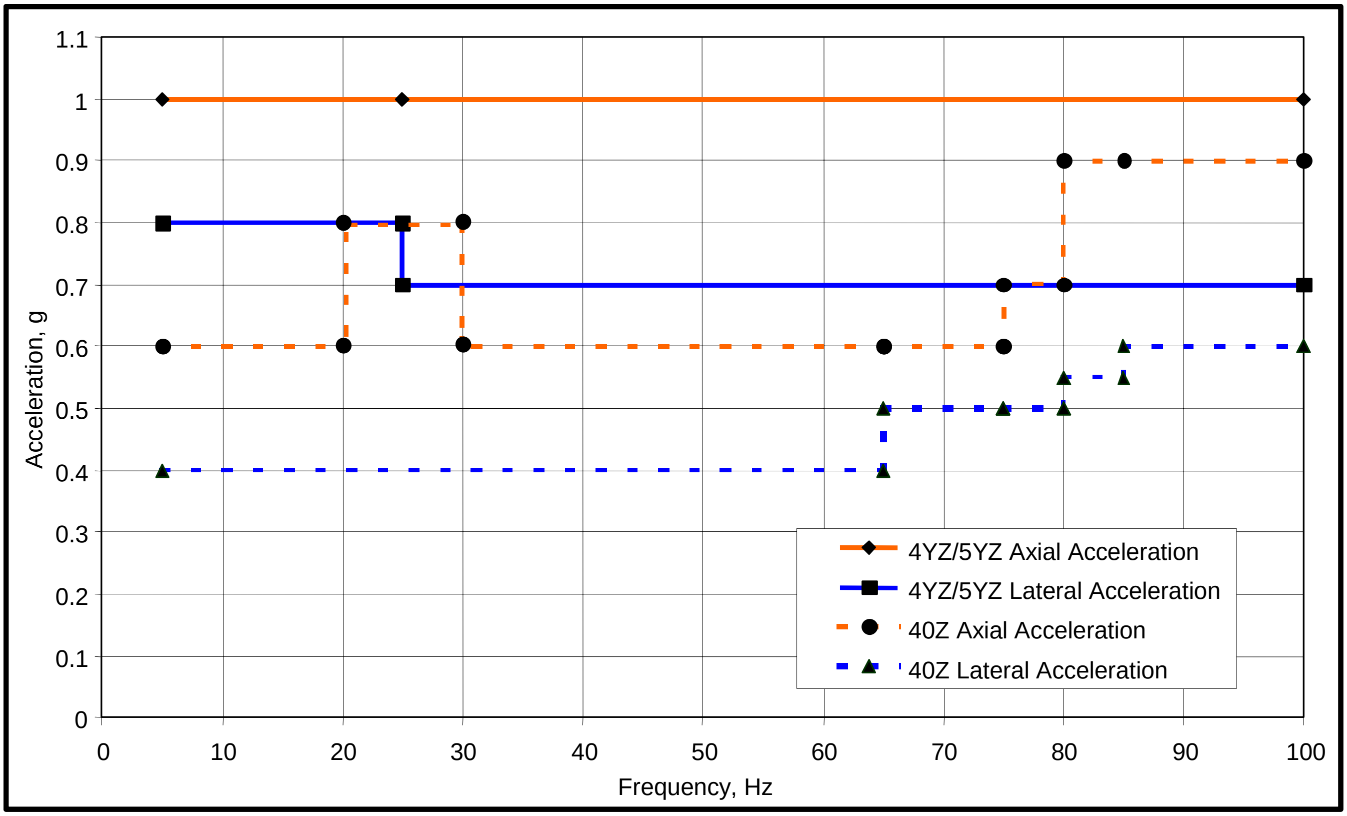


Figure 3.2.3-2 Quasi-Sinusoidal Vibration Levels for Atlas V 400 Series and Atlas V 500 Series

range to maintain component responses below design levels. Notching in the 50- to 100-Hz frequency range is not recommended without technical discussion concerning flight equivalent levels as related to actual spacecraft damping.

High-frequency random vibration that the spacecraft experiences is primarily due to the acoustic noise field, with a very small portion being mechanically transmitted from through the spacecraft interface. The acoustically excited random vibration environment tends to be the design driver for lightweight components and small structural supports. The high-frequency vibration level will vary from one location to another depending on physical properties of each area of the spacecraft. Because the vibration level at the payload interface depends on the adjacent structure above and below the interface, the exact interface levels depend on the structural characteristics of the lower portion of the spacecraft, the particular payload adapter, and the influence of the acoustic field for the particular spacecraft. Acoustic test of the spacecraft will therefore be the most accurate simulation of the high-frequency environment experienced in flight and is preferable to base input random vibration test. If the spacecraft is mounted to a test fixture that has structural characteristics similar to the PLA, then vibration levels at the interface will be similar to flight levels. To accurately reflect the flight environment, it is not recommended to attach the spacecraft to a rigid fixture during acoustic testing.

3.2.4 Shock

Four pyrotechnic shock events occur during flight on the Atlas family of vehicles; payload fairing jettison (PFJ), Centaur separation from the Atlas, spacecraft separation, and separation of the Centaur Forward Load Reactor (CFLR) for the Atlas V 500 series vehicles. The CFLR provides a structural connection between the top of the Centaur and the 5-m payload fairing for reduced loss of clearance within the payload compartment. In general, the maximum shock environment for the spacecraft is generated by the spacecraft separation device. While the other events do produce noticeable shock, the levels are significantly lower than separation system shock due to distance attenuation.

Figure 3.2.4-1 shows the maximum expected shock levels for spacecraft separation for a typical spacecraft at the separation plane for the Type A, B1, and D adapters. Figure 3.2.4-2 provides the shock specification for a hard-point connection between the launch vehicle and spacecraft for the Type E (or similar) nut-fired adapter. All shock environments are defined on the Launch Vehicle side of the

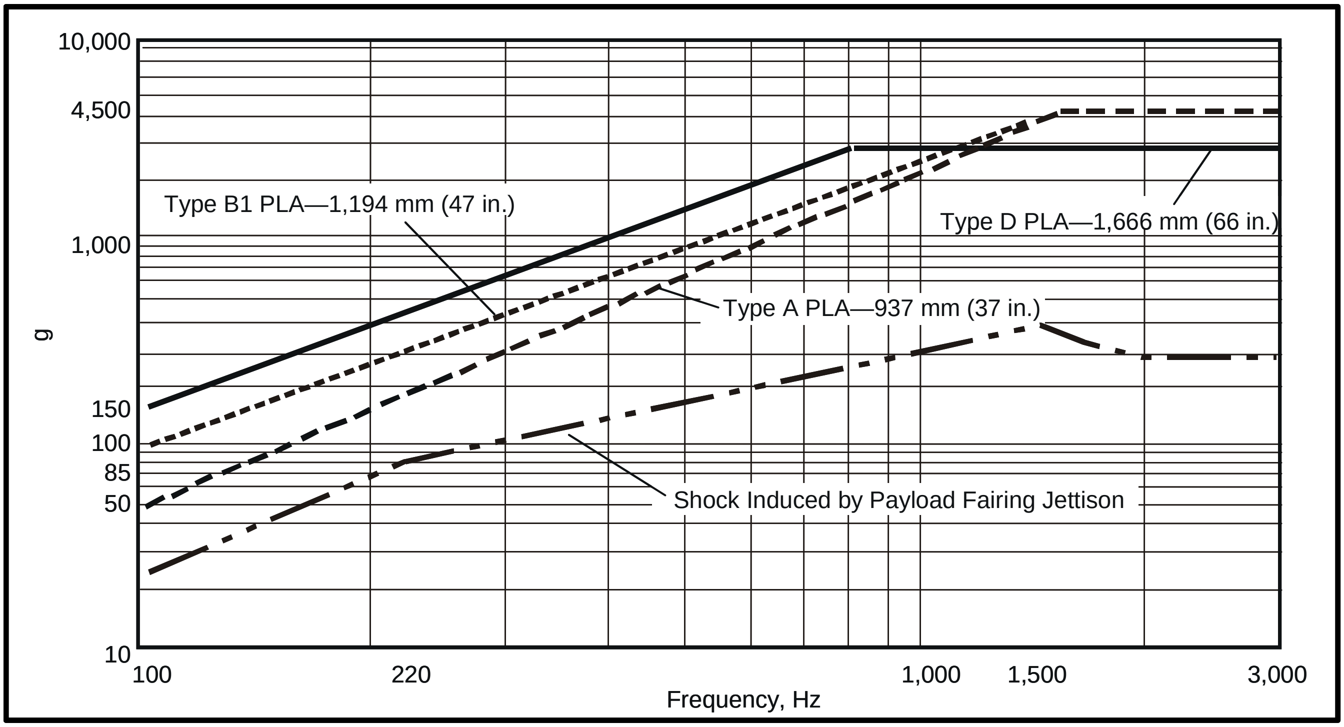


Figure 3.2.4-1 Typical Maximum Atlas Shock Levels—Type A, B1, and D Adapters

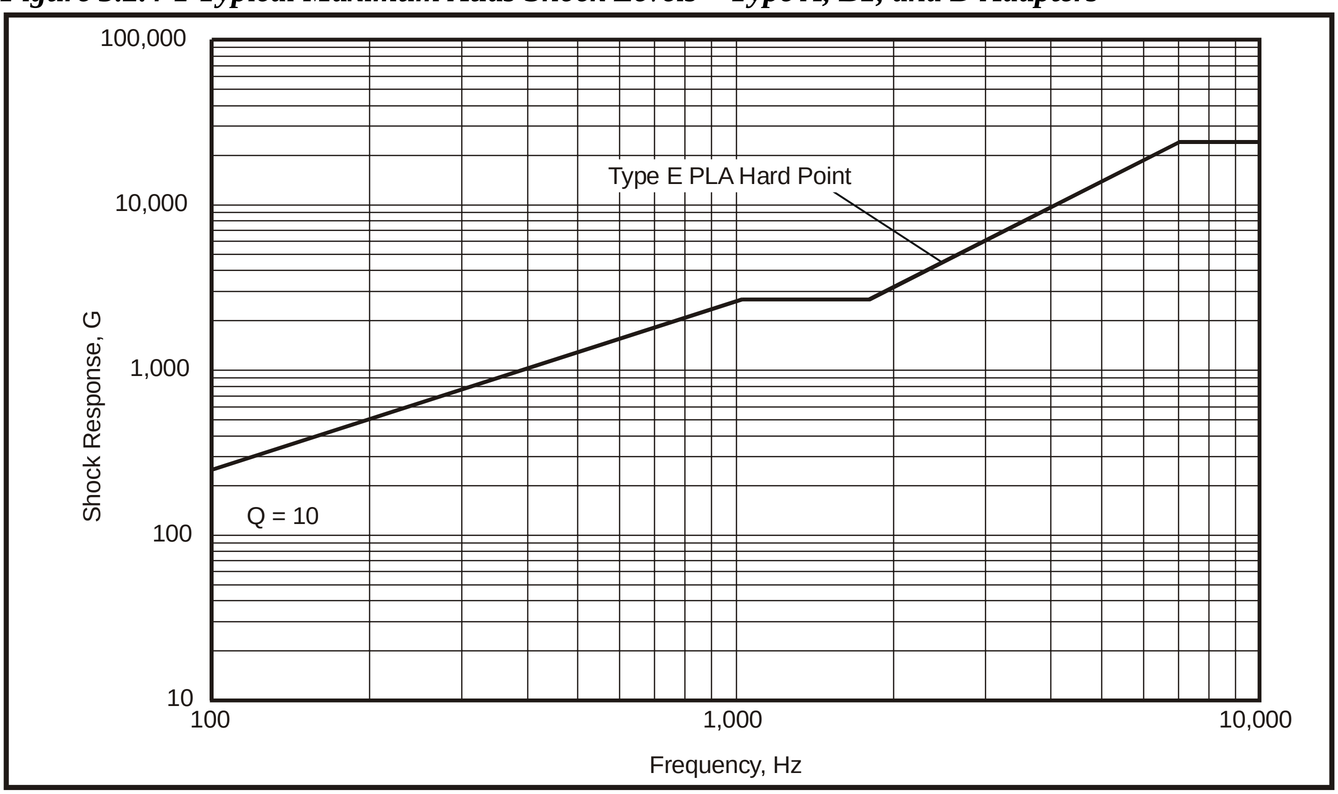


Figure 3.2.4-2 Maximum Atlas Shock Levels for the Type E System

interface and represent a 95% probability and 50% confidence envelope with a resonant amplification factor (Q) of 10. Response on the spacecraft side will be dependent on the unique characteristics of the spacecraft's interface structure. Spacecraft adapter systems incorporating low shock separation devices are under development (Section 4.1.2). Shock information for these systems is available upon request.

Figure 3.2.4-3 shows the maximum acceptable shock level at the equipment module interface for a customer-provided separation system. This requirement is necessary to assure Atlas Launch Vehicle component compatibility with spacecraft induced shock.

3.2.5 Thermal

Within Fairing—The PLF protects the spacecraft during ascent to an altitude of approximately 113,000 m (370,000 ft). Aerodynamic heating on the fairing results in a time-dependent radiant heating environment around the spacecraft before fairing jettison. The PLF uses cork on the external surface to minimize fairing skin temperatures. For the 4-m PLF, the inner PLF bare skin surfaces of the cone and cylinder have a low-emittance finish (ε <0.1) that minimizes heat transfer to the spacecraft. The PLF boattail and inner acoustic blanket surfaces facing the spacecraft for the 4-m PLF have an emittance of 0.9 (ε 0.9). The peak heat flux radiated by the 4-m PLF bare cone and cylinder inner surfaces is less than 400 W/m^2 (125 Btu/hr-ft^2) and peak PLF skin temperatures remain below 212°C (414°F) at the warmest location. Inner acoustic blanket temperatures remain below 49°C (120°F).

For the Atlas V 5-m PLF, the spacecraft thermal environment is attenuated by the acoustic suppression system that is baselined for the PLF cylinder and the lower portion of the ogive nose section. The inner surfaces of the bare composite 5-m PLF nose and cylinder have an emittance of 0.9. The peak heat flux radiated by the inner surfaces of the bare cone and cylinder of the 5-m PLF is less than 914 W/m^2 (290 Btu/hr-ft^2), and peak temperatures remain below 93°C (200°F), at the warmest location. The inner acoustic suppression system surface temperatures for the 5-m PLF also remain below 93°C (200°F).

After Fairing Jettison—Fairing jettison typically occurs when the 3-sigma maximum free molecular heat flux decreases to 1,135 W/m^2 (360 Btu/hr-ft^2). Jettison timing can be adjusted to meet specific

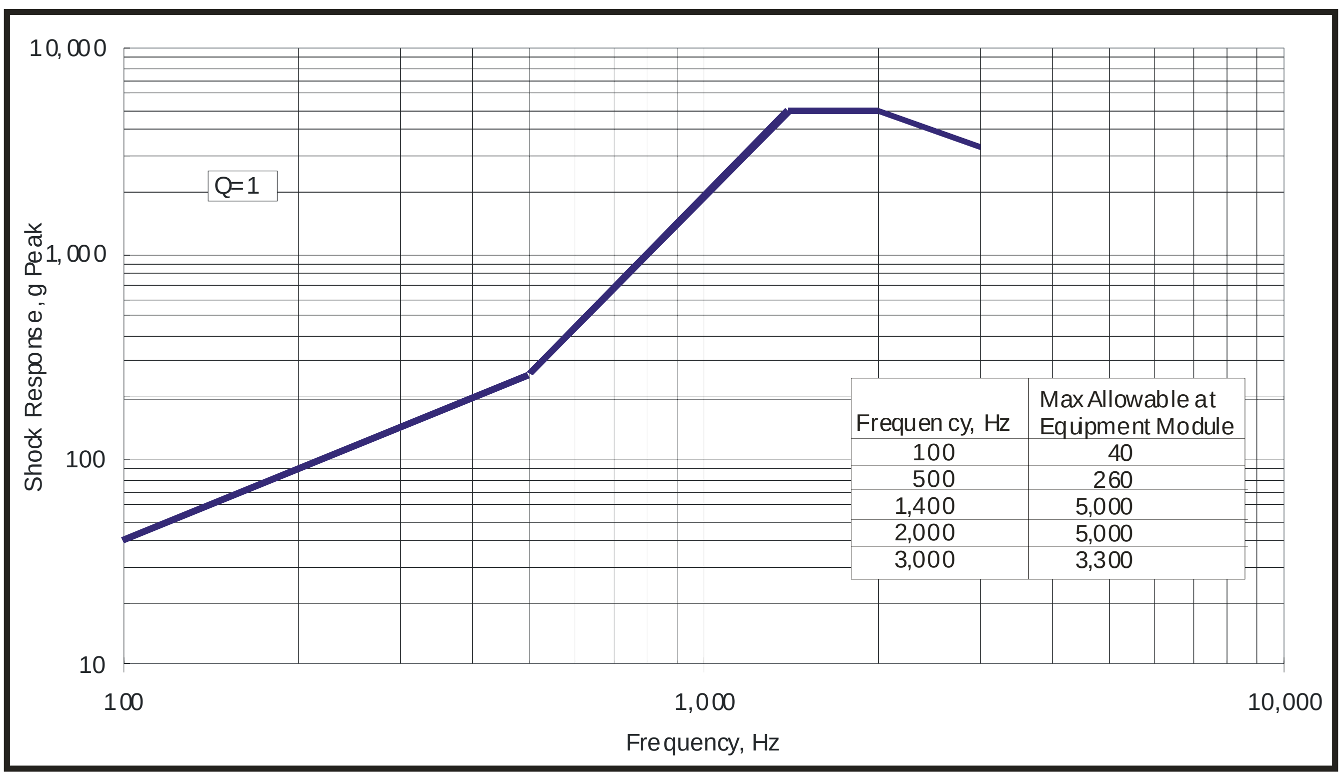


Figure 3.2.4-3 Maximum Allowable Spacecraft-Produced Shock at Forward Adapter Interface

mission requirements. Free molecular heating (FMH) profiles are highly dependent on the trajectory flown. Typical FMH profiles for standard parking orbit coast length missions (15 minutes) are shown in Figure 3.2.5-1. Because actual profiles are highly dependent on the trajectory flown, these data should not be used for design. Raising the parking orbit perigee altitude can reduce peak FMH levels (Fig. 3.2.5-1), however it will have a minor negative effect on delivered launch vehicle performance.

The spacecraft thermal environment following fairing jettison includes free molecular heating, solar heating, Earth albedo heating, Earth thermal heating, and radiation to the upper stage and to deep space. The spacecraft also is conductively coupled to the forward end of the Centaur upper stage through the spacecraft adapter. Solar, albedo, and Earth thermal heating can be controlled as required by the spacecraft by specification of launch times, vehicle orientation (including rolls), and proper mission design.

For a typical 30-minute GTO mission, the Centaur nominally provides a benign thermal influence to the spacecraft, with radiation environments ranging from -45 to 52°C (-50 to 125°F) and interface temperatures ranging from 4 to 49°C (40 to 120°F) at the forward end of the spacecraft adapter. Neither upper stage main engine plumes nor reaction control system (RCS) engine plumes provide any significant heating to the spacecraft. The main engine plumes are nonluminous due to the high purity of LH_2 and LO_2 reactants.

3.2.6 Static Pressure (PLF Venting)

The payload compartment is vented during the ascent phase through one-way vent doors. Payload compartment pressure and depressurization rates are a function of the fairing design and trajectory.

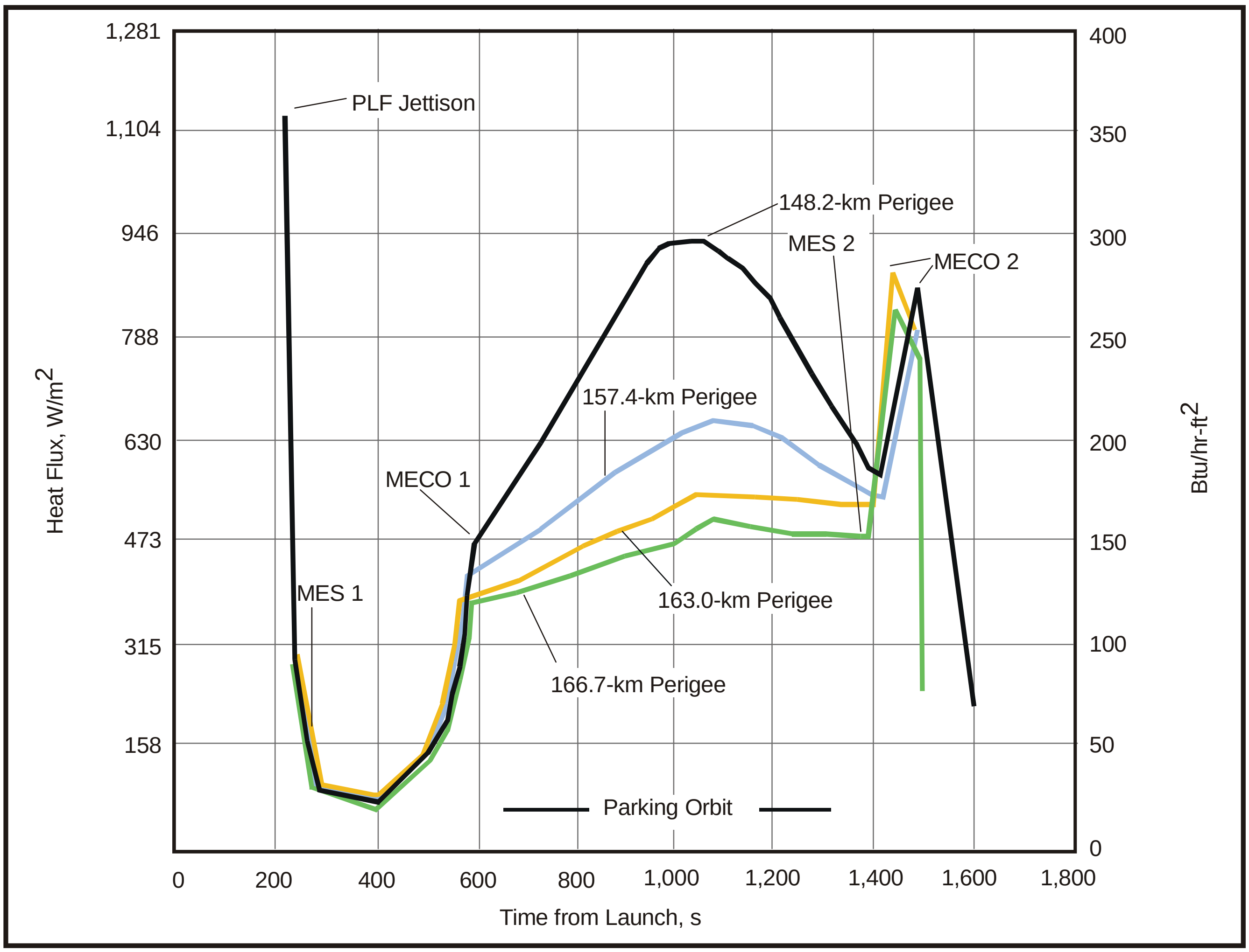


Figure 3.2.5-1 Typical FMH Flux Profiles

The 4-m large payload fairing (LPF) and extended payload fairing (EPF), and the 5-m payload fairing, were designed to have a depressurization rate of no more than 6.20 kPa/s (0.9 psi/s). The pressure decay rate will always be less than 2.5 kPa/s (0.36 psi/s), except for a short period around transonic flight when the decay rate will not exceed 5.0 kPa/s (0.73 psi/s). Typical depressurization rates are less than these values. Figures 3.2.6-1 through 3.2.6-4 illustrate typical bounding pressure profiles and

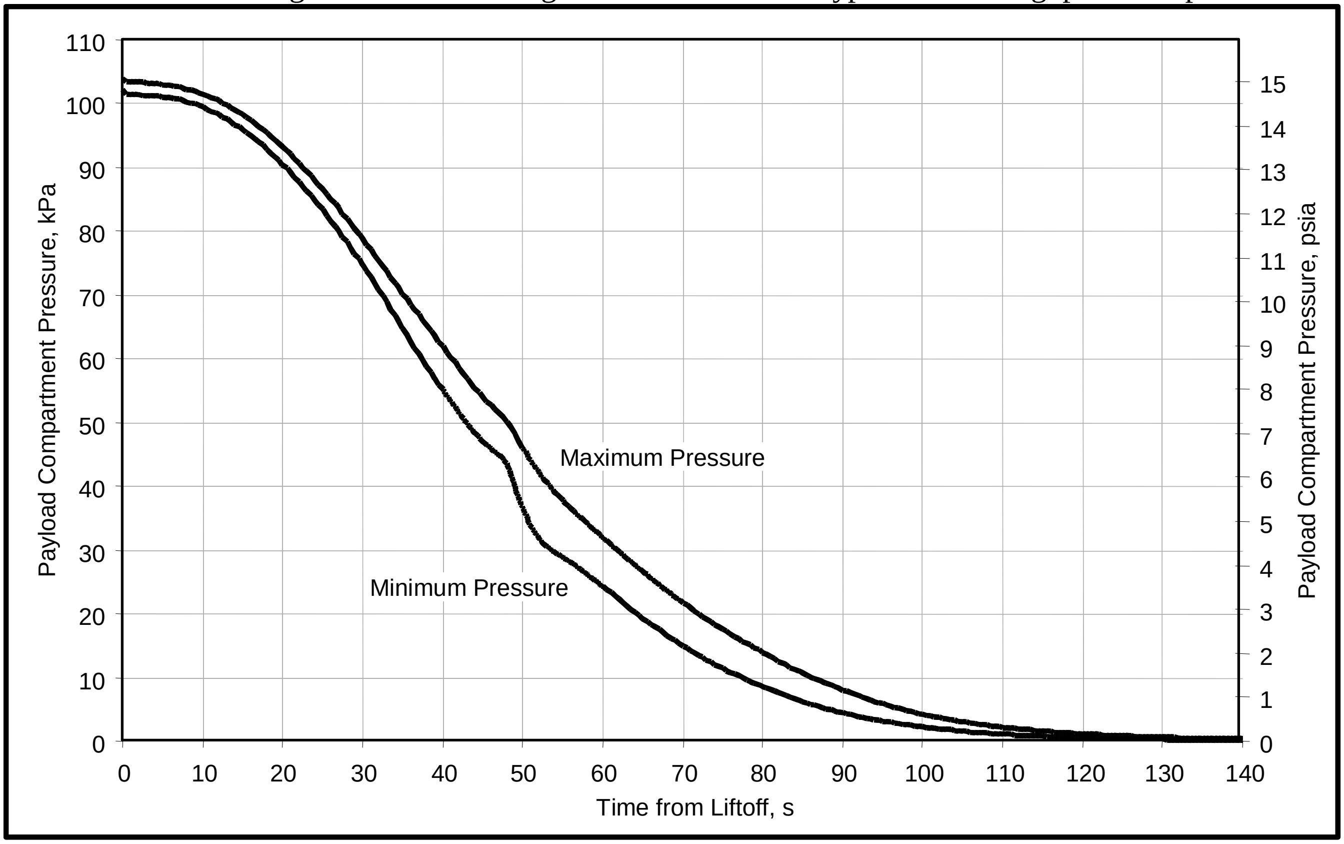


Figure 3.2.6-1 Typical Static Pressure Profiles Inside the Large Payload Fairing for Atlas IIIB Vehicle

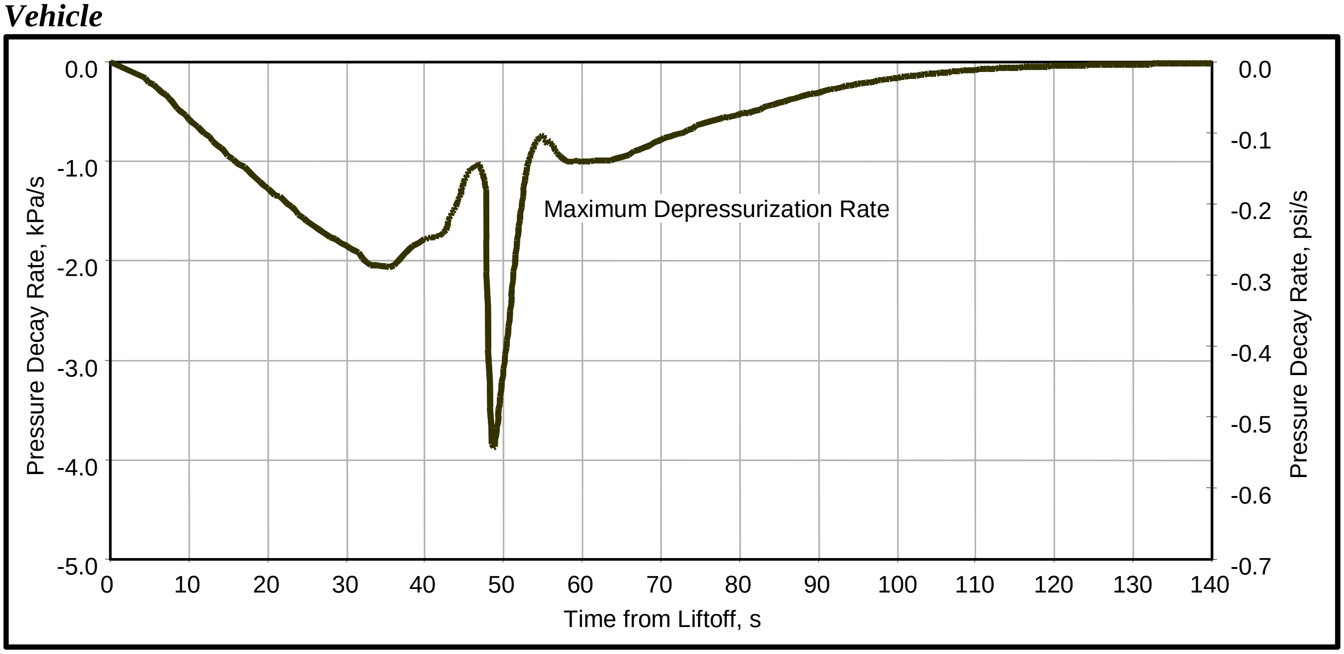


Figure 3.2.6-2 Typical Payload Compartment Pressure Decay Rate Inside the LPF for Atlas IIIB Vehicle

depressurization rates for the 4-m fairing, Figures 3.2.6-5 and 3.2.6-6 illustrate typical pressure profiles and depressurization rates for the 5-m fairing.

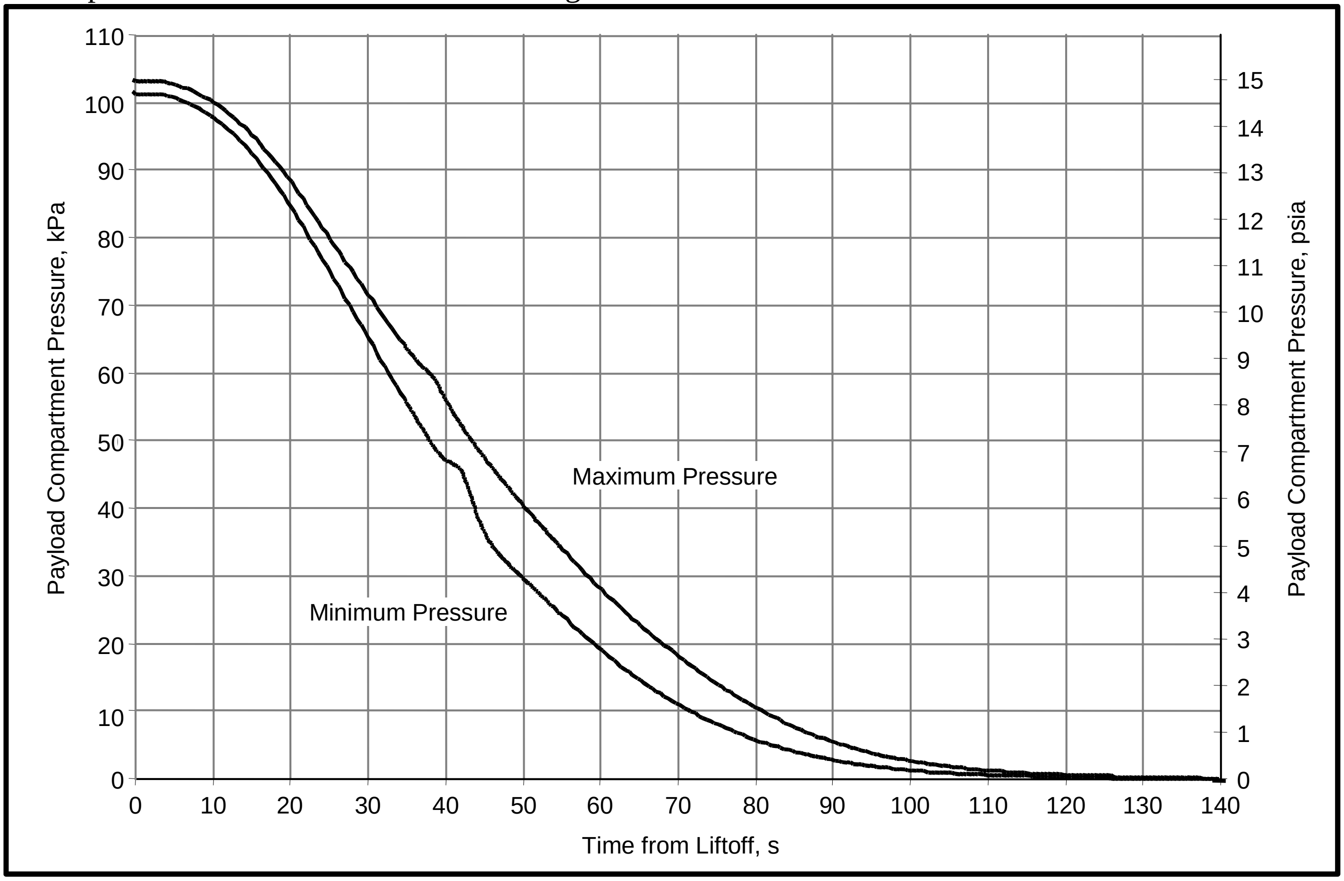


Figure 3.2.6-3 Typical Static Pressure Profiles Inside the Extended Payload Fairing for Atlas V 431 Vehicle

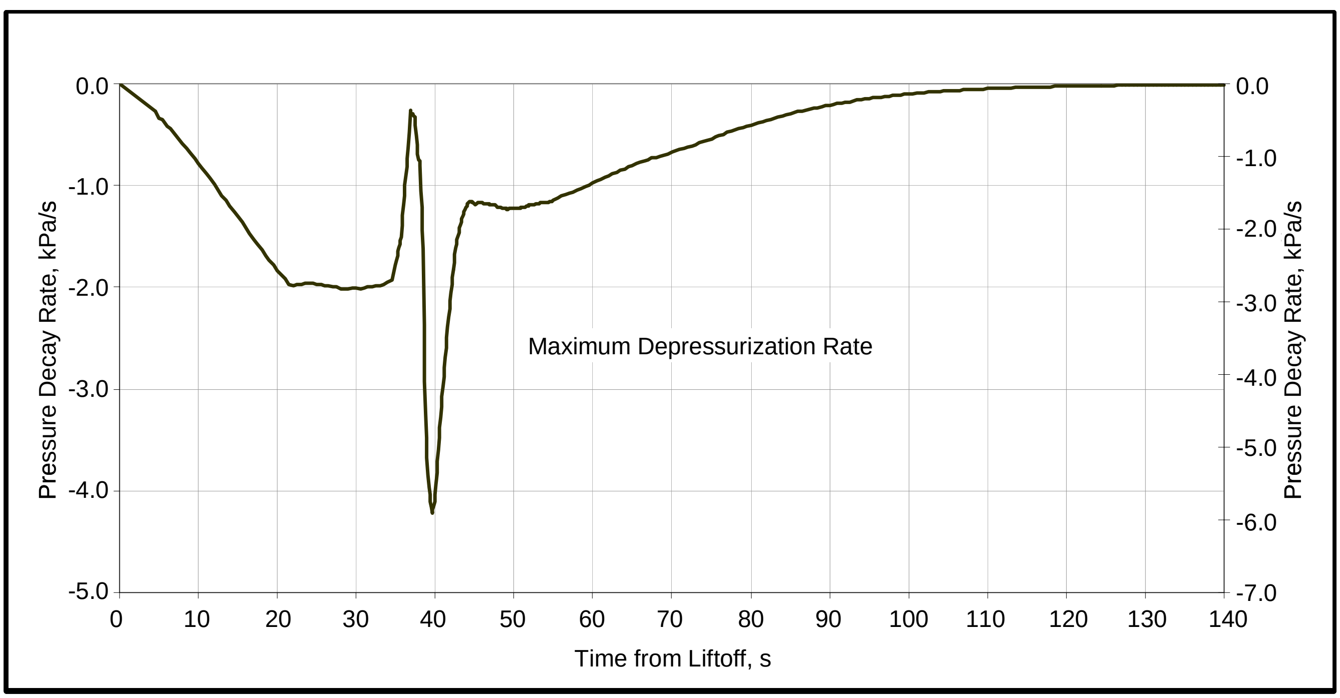


Figure 3.2.6-4 Typical Payload Compartment Pressure Decay Rate Inside the EPF for Atlas V 431 Vehicle

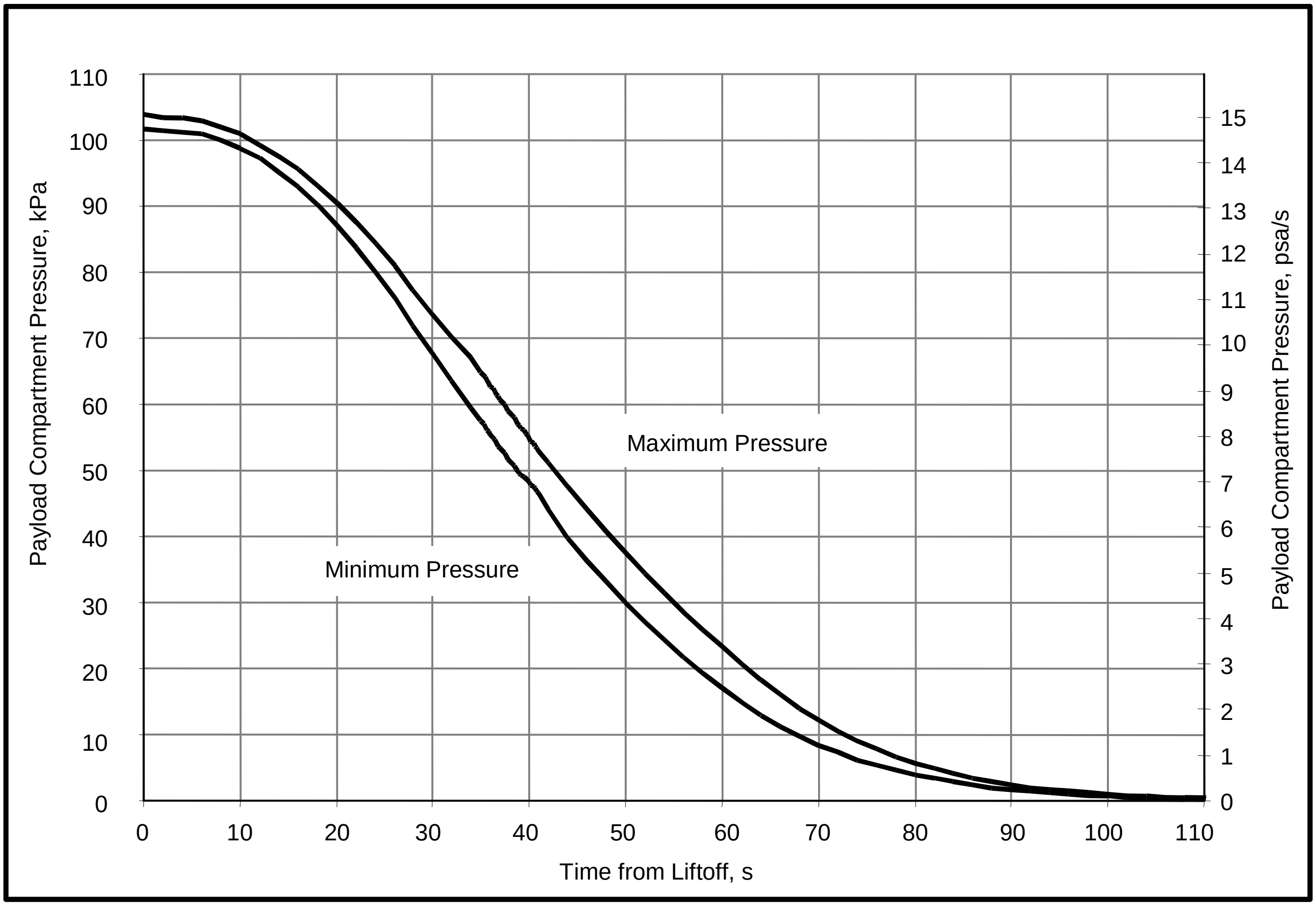


Figure 3.2.6-5 Typical Static Pressure Profiles Inside the 5-m Fairing

3.2.7 Inflight Contamination Control

Launch system ground contamination sources were addressed in Paragraphs 3.1.3.2 and 3.1.3.3. Launch system ascent contamination sources are discussed below.

3.2.7.1 Molecular Outgassing—Nonmetallic materials outgas molecules that can deposit on spacecraft surfaces. This source is limited by choosing low outgassing materials where possible and by limiting, encapsulating, or vacuum-baking higher outgassing materials.

3.2.7.2 Nonvolatile Residue Redistribution—PLF and other surfaces in the PLF volume will have small amounts of adsorbed molecules that desorb when these surfaces are warmed. They can deposit on spacecraft surfaces that are cooler than the condensation temperature of these molecules. This source is limited on Atlas II and III by removing adsorbed molecules with demonstrated cleaning techniques. Atlas V hardware is cleaned and tested to less than 1 mg/ft^2 of nonvolatile residue (NVR).

3.2.7.3 Particle Redistribution—Particles on surfaces within the PLF volume can shake loose and redistribute to spacecraft surfaces during launch and flight. This source is limited by cleaning hardware to Visibly Clean Level 2 criteria before encapsulation. The PLF is also vibrated as part of the cleaning process. Additional launch vehicle hardware cleanliness levels may be specified to meet mission-unique requirements.

3.2.7.4 PLF Separation—Separation of the 4-m PLF is accomplished using pyrotechnic separation bolts and jettison springs. The pyrotechnic bolts on the PLF are located in individual cavities that isolate them from the spacecraft. Particle production from PLF jettison springs has been tested and is negligible. The 5-m PLF halves are separated by a linear pyrotechnic separation system that is fully contained in an expandable bellows. The bellows expands forcing the shearing of a rivet line. The sheared rivets

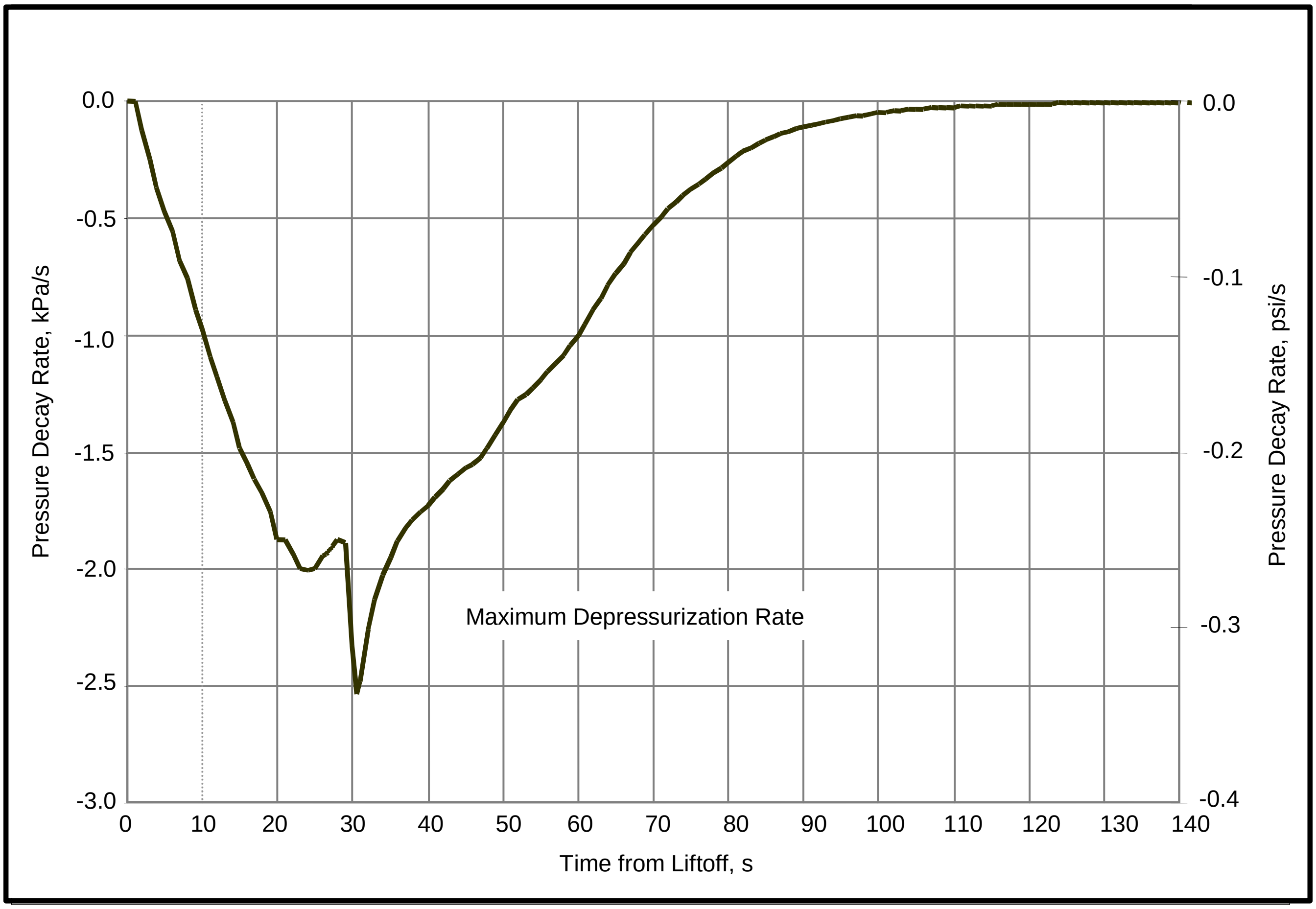


Figure 3.2.6-6 Typical Payload Compartment Pressure Decay Rate for the 5-m Fairing

are also retained by tape. Satellite contamination from this system is expected to be negligible based on the results of analyses and tests.

3.2.7.5 Booster Separation—The booster is separated from the Centaur by a linear charge and propelled away from it by retrorockets. These two systems are discussed for Atlas II, III, and V vehicles separately below:

3.2.7.5.1 Atlas IIAS—Approximately 270 seconds from launch, the Atlas sustainer is separated from Centaur by the linear charge. The linear charge produces debris which travels in a primary direction 90° from the direction to the spacecraft. The Centaur aft tank ring prevents movement directly in the forward direction. After separation, eight retrorockets near the aft end of the Atlas (Station 1133) are fired to ensure the expended Atlas stage moves away from Centaur. These eight retrorockets use solid propellants; plume products will consist of small solid particles and very low-density gases.

Retrorocket nozzles are canted outboard 40° from the vehicle axis. This cant angle ensures that virtually no solid particles will impact the spacecraft. Plume gases that impinge on the spacecraft are rarefied and only a small fraction are condensable because spacecraft surfaces are still relatively warm from prelaunch payload compartment gas conditioning. Moderate molecular depositions can be expected on aft facing outlying surfaces (> 152-cm [60-in.] radius).

3.2.7.5.2 Atlas IIIA, IIIB, and V 400—The Atlas IIIA, IIIB, and Atlas V 400 mission designs have the PLF in place on the vehicle during detonation of the linear charge and firing of the retrorockets. PLF leak areas are small and entrance of gases through vent areas is controlled by one-way flapper doors on all PLF designs. This virtually eliminates the separation systems as a contamination source since there is no credible transport path from the retrorockets to the payload.

3.2.7.5.3 Atlas V 500**—**The booster is separated from Centaur with a linear-shaped charge located just aft of the Centaur aft tank ring. This event occurs inside the PLF boattail that remains even though the PLF has been jettisoned before the booster separation event. Testing of debris transport in this configuration is in work for Atlas V.

After separation, eight retrorockets located in the booster intertank compartment are fired to ensure the expended booster moves away from Centaur. These eight retrorockets use solid propellants and are canted outboard; the resulting plume products consist of small solid particles and very low-density gases. The boattail shields the spacecraft from particles in the retrorocket plume. Retrorocket plume gases that expand around the boattail and impinge on the spacecraft are rarefied and only a small fraction are condensable because spacecraft surfaces are still relatively warm from prelaunch payload compartment gas conditioning. Moderate molecular depositions can be expected on aft-facing surfaces.

3.2.7.6 Spacecraft Separation Event—Atlas uses a spacecraft separation system similar to those used throughout the launch vehicle industry. This system uses two pyrotechnically initiated bolt cutters to release two clamp halves that hold the spacecraft to the launch vehicle payload adapter. Upon actuation, small particles can be generated by this system. The kinetic energy of particles observed during separation system testing was approximately two orders of magnitude less than the micrometeoroid design criteria specified in NASA SP 8013. The 99.9% largest expected particle has a mass of 0.008 grams and a longest dimension of 0.43 cm (0.17 in.).

3.2.7.7 Collision and Contamination Avoidance Maneuver (CCAM)—The upper-stage RCS consists of 12 hydrazine (N_2H_4) thrusters for Centaur propellant settling and attitude-control requirements. Four thrusters provide axial thrust and eight provide roll, pitch, and yaw control. Thrusters are located slightly inboard on the upper-stage aft bulkhead. Thrusters produce a plume that has an extremely low contaminant content.

Before upper-stage/spacecraft separation, the spacecraft will not be exposed to RCS exhaust plumes. The RCS thruster's inboard location on the aft bulkhead precludes direct line of access between the spacecraft and thrusters.

After separation, some minor spacecraft impingement from thruster exhaust plumes may occur during the CCAM. CCAM is designed to prevent recontact of the Centaur with the spacecraft while minimizing contamination of the spacecraft.

A typical CCAM sequence is shown in Figure 3.2.7.7-1. This figure shows typical spacecraft motion after the separation event as longitudinal and lateral distance from the upper stage. Included are contour lines of constant flux density for the plumes of the aft-firing RCS settling motors during operation. The plumes indicate the relative rate of hydrazine exhaust product impingement on the spacecraft during the 2S-ON (two settling motors on) phase before blowdown and during hydrazine depletion. There is no impingement during the CCAM 4S-ON phase because the spacecraft is forward of the settling motors. Analysis of spacecraft contamination from this event predicts a maximum deposition of 6.03 x 10-9 g/cm^2 (0.6Å) on spacecraft surfaces from the Atlas II and III vehicles. The Atlas V has a similar design philosophy, but the use of a different grade of hydrazine combined with worst-case separation attitudes and vehicle weights results in a maximum deposition of 3Å.

3.2.7.8 Upper-Stage Main Engine Blowdown—As part of the CCAM, hydrogen and oxygen are expelled through the engine system to safe the vehicle and to further increase upper-stage/spacecraft separation distance. Hydrogen is expelled out the engine cooldown ducts and oxygen is expelled out the main engine bells. The expelled products are hydrogen, oxygen, and trace amounts of helium, which are noncontaminating to the spacecraft. Main engine blowdown does not begin until a separation distance

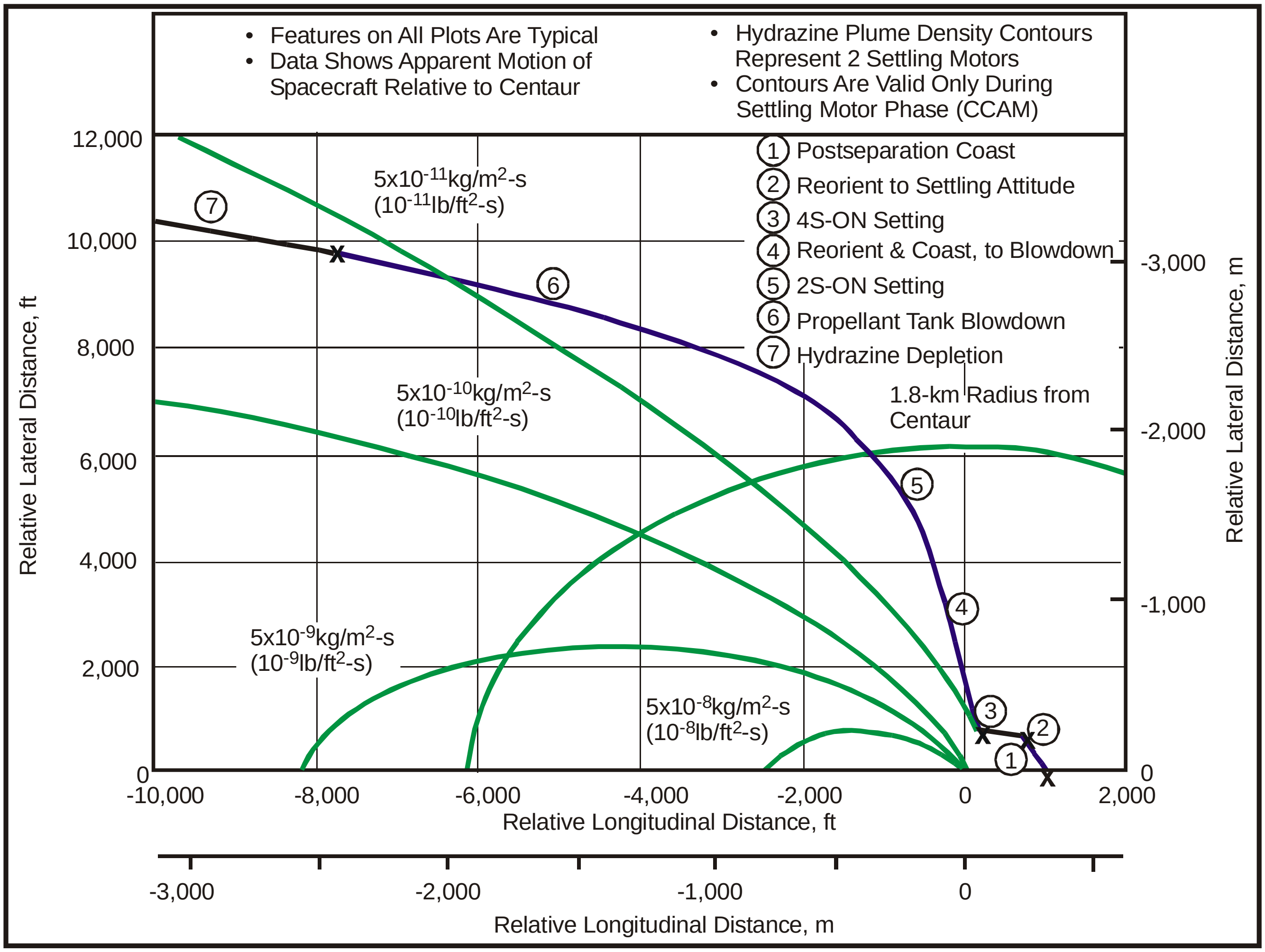


Figure 3.2.7.7-1 Typical Spacecraft Motion Relative to Centaur Upper Stage

≥1 nautical mile has been attained. Figure 3.2.7.8-1 identifies typical main engine blowdown exhaust product impingement rates on the spacecraft.

3.2.8 Radiation and EMC

The description of environments in Section 3.1.2 encompasses worst-case flight environments with some individual exceptions, which are dependent upon launch trajectory. Any individual exceedances will be addressed for specific launches.

3.3 SPACECRAFT COMPATIBILITY TEST REQUIREMENTS

Lockheed Martin requires that the spacecraft be capable of experiencing maximum expected flight environments multiplied by minimum factors of safety to preclude loss of critical function. An environmental test report is required to summarize the testing performed and to document the compatibility of the spacecraft with flight environments.

The spacecraft testing required for demonstration of compatibility is listed in Table 3.3-1. Table 3.3-2 describes structural environment test margins and durations appropriate as a minimum for programs in three phases of development. The structural test model (STM) is considered a test-dedicated qualification unit with mass simulation of components to be tested in unit qualification programs. Data acquired during STM tests may be used to establish qualification levels for each component. The system level vibration test shall include wet propellant tanks for new designs if acceptance testing will not include this important effect.

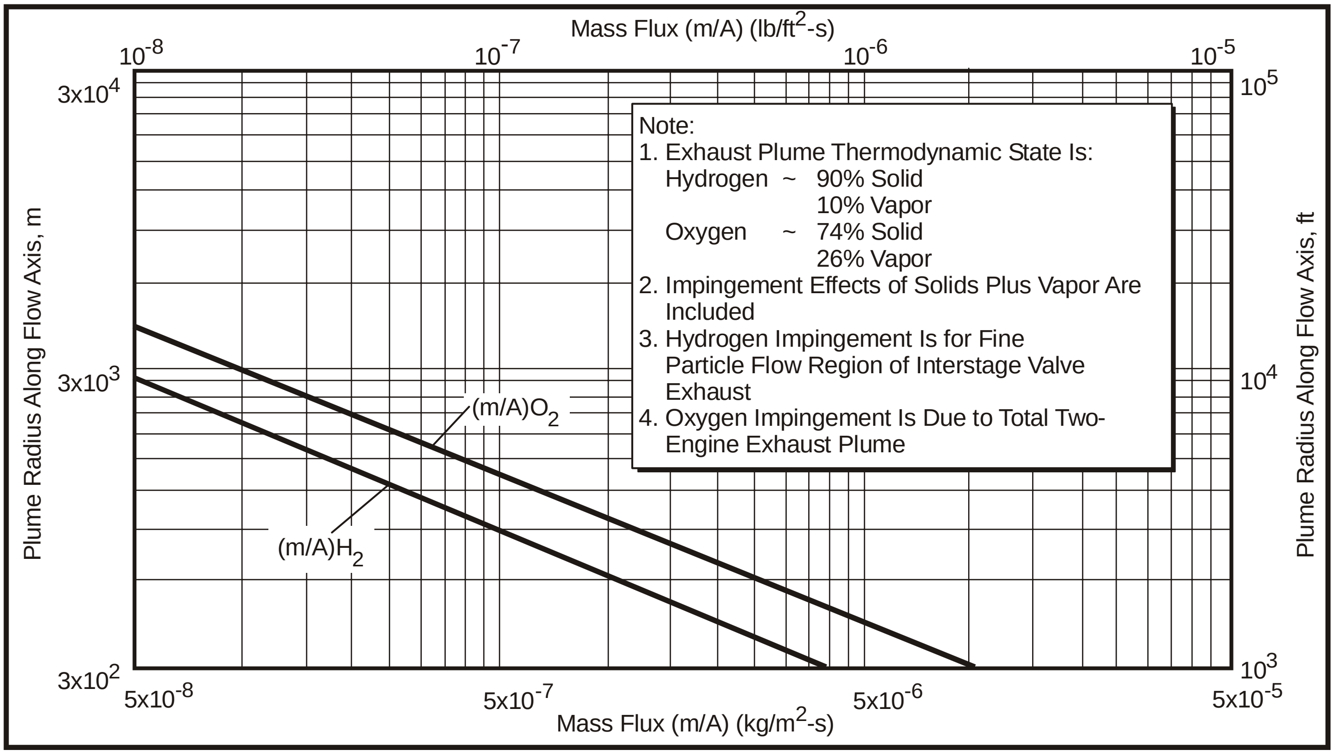


Figure 3.2.7.8-1 Typical Spacecraft Impingement Fluxes During Propellant Tank Blowdown

The protoflight model (PFM) is the first flight article produced without benefit of a qualification or STM program. The flight-configured spacecraft is exposed to qualification levels for acceptance durations. The flight model (FM) is defined as each flight article produced after the qualification or protoflight article. Tests required for each FM are intended as proof-of-manufacturing only and are performed at maximum expected flight levels.

Flight hardware fit checks are performed to verify mating interfaces and envelopes. Verification of spacecraft compatibility with shock produced by separation systems provided by Lockheed Martin is typically demonstrated by firing of a flight-configured system following the FM fit check. This test may also be performed during STM or PFM testing to establish a mapping of shock levels for component locations near the interface. Component unit qualification testing must envelop the mapped environment. For user-supplied adapters and separation systems, firing of the actual separation device on a representative payload adapter and spacecraft to measure the actual level and/or qualify the spacecraft is recommended.

Lockheed Martin also suggests that the spacecraft contractor demonstrate the spacecraft capability to withstand thermal and EMI/EMC environments.

Table 3.3-1 Spacecraft Qualification and Acceptance Test Requirements

	Acoustic	**Shock**	**Sine Vibration**	**EMI/EMC**	**Modal Survey**	**Static Loads**	**Fit Check**
Qual	X	X	X	X	X	X	
Accept	X		X				X

Table 3.3-2 Spacecraft Structural Tests, Margins, and Durations

Test	**STM (Qual)**	**PFM* (Protoflight)**	**FM (Flight)**
Static • Level • Analyses	 1.25 x Limit (DLF or CLA)	 1.25 x Limit (CLA)	 1.1 x Limit (Proof Tests)
Acoustic • Level • Duration	 Limit + 3 dB 2 Min	 Limit + 3 dB 1 Min	 Limit Level 1 Min
Sine Vib • Level • Sweep Rate	 1.25 x Limit 2 Oct/Min	 1.25 x Limit 4 Oct/Min	 Limit Level 4 Oct/Min
Shock	1 Firing	1 Firing	1 Firing

*Note: The Protoflight test levels are also used for validation of ICD dynamic environments when supplemental FM measurements (Mission Satisfaction Option) are made for a specific mission.

4.0 SPACECRAFT INTERFACES

4.1 SPACECRAFT-TO-LAUNCH VEHICLE INTERFACES

The Atlas launch system is designed to meet requirements of currently defined spacecraft and offers the flexibility to adapt to mission-specific needs. Primary interfaces between the Atlas launch vehicle and the spacecraft consist of a payload fairing (PLF) that encloses and protects the spacecraft and a payload adapter (PLA) that supports the spacecraft on top of the launch vehicle. These components are designed to provide mechanical and electrical interfaces required by the spacecraft and to provide a suitable environment during integration and launch activities. Payload adapters and payload fairings that are offered by the Atlas program are shown in Figure 4.1-1.

The static payload envelope defines the usable volume for a spacecraft. This envelope represents the maximum allowable spacecraft static dimensions (including manufacturing tolerances) relative to the spacecraft and payload adapter interface. For clearances between the spacecraft and payload fairing, primary clearance concerns are for dynamic deflections of the spacecraft and payload fairing and the resulting relative loss of clearance between these components. For clearances between the spacecraft and payload adapter, primary envelope concerns are for access to the mating components and payload

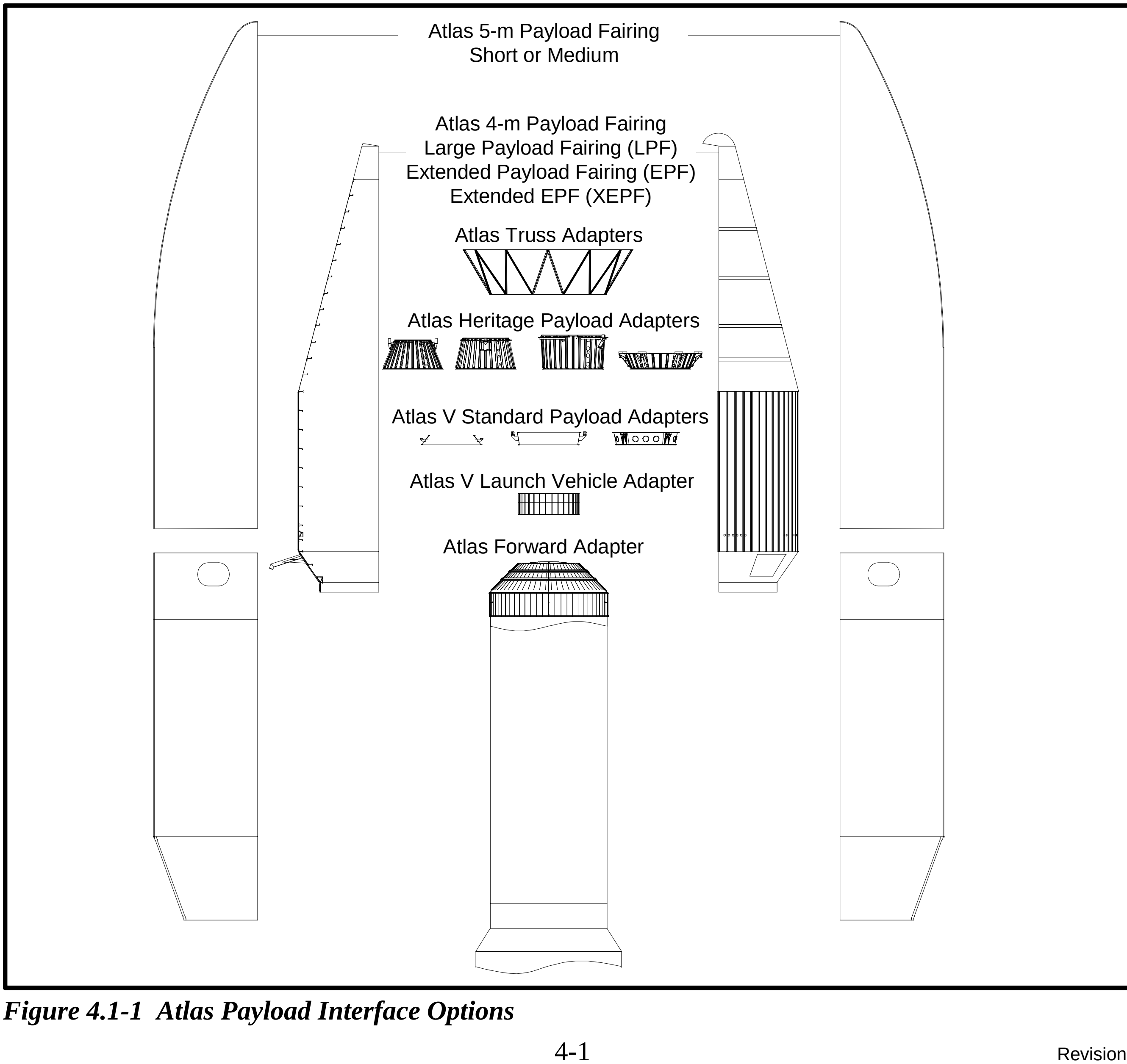


Figure 4.1-1 Atlas Payload Interface Options

separation system for integration and installation operations, motion of the payload separation system during its operation, and movement of the spacecraft and launch vehicle after separation of the spacecraft and payload. For all areas of the static payload envelope, clearance layouts and analyses are performed for each spacecraft configuration, and if necessary, critical clearance locations are measured during spacecraft and launch vehicle integration operations to ensure a positive clearance is maintained. Overall views of static payload envelopes for different Atlas vehicle configurations are shown in Figure 4.1-2.

The following sections summarize these interface options. Detail descriptions of payload fairings are included in Appendix D, and of payload adapters in Appendix E. The interface information in these sections should be used only as a guideline. Modifications to these systems may be accommodated on a mission-specific basis. Ultimate control of interface information for a given mission is governed through the interface control document (ICD) developed and maintained during the mission integration process.

4.1.1 Atlas Payload Fairings

The payload fairing encloses and protects the spacecraft during ground operations and launch vehicle ascent. The payload fairing also incorporates hardware to control thermal, acoustic, electromagnetic, and cleanliness environments for the spacecraft and may be tailored to provide access and radio frequency (RF) communications to the encapsulated spacecraft. The Atlas user has a choice between the 4-m and 5-m diameter payload fairings (Fig. 4.1.1-1), both of which are available in different lengths.

4.1.1.1 Atlas 4-m Payload Fairing (4-m LPF, 4-m EPF, and 4-m XEPF)—The Atlas large payload fairing (LPF), extended payload fairing (EPF), and extended EPF (XEPF) have a 4-m diameter cylindrical section topped by a conical section. Major sections of these fairings are the boattail, the cylindrical section, and the nose cone that is topped by a spherical cap (Fig. 4.1.1.1-1). The EPF was

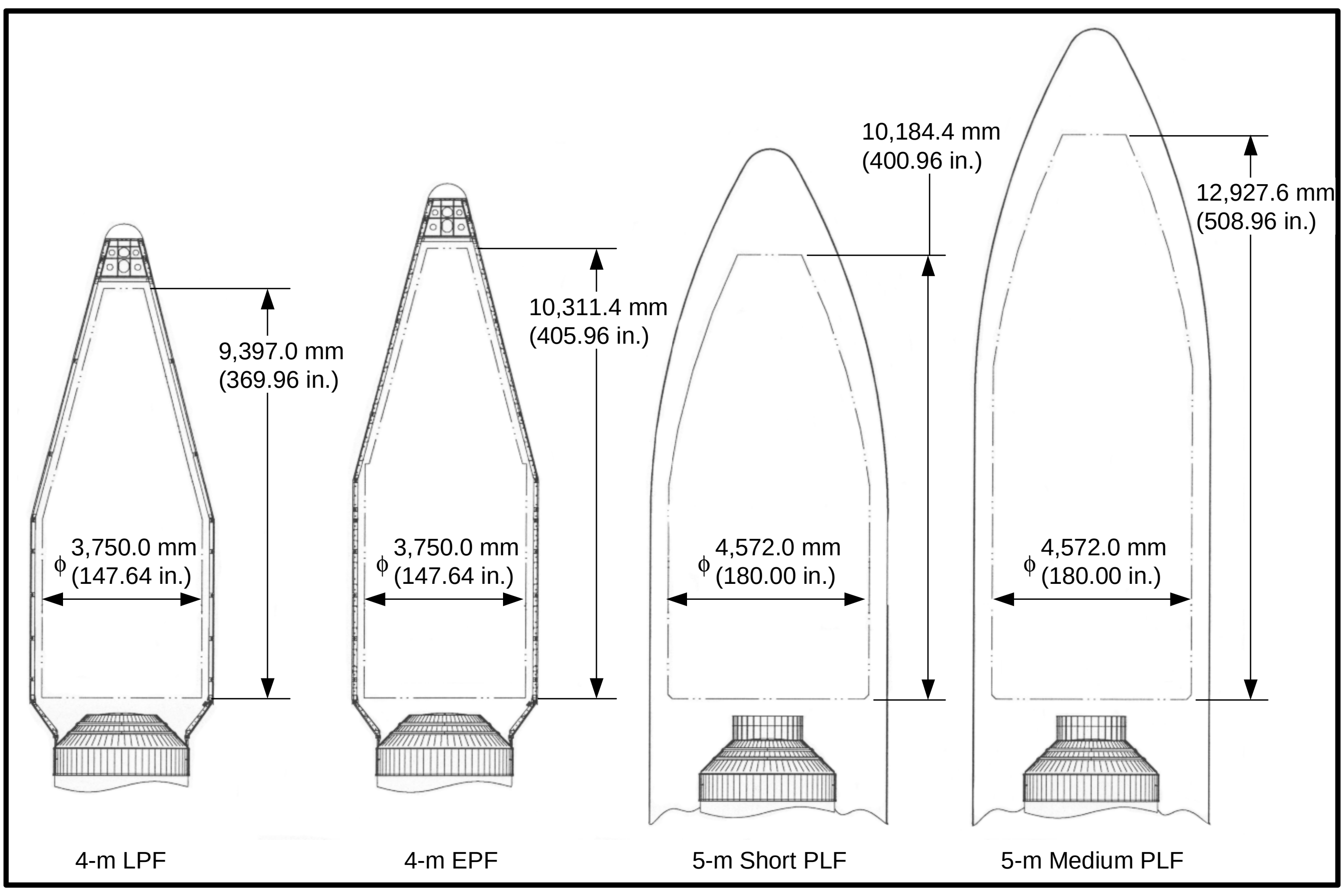


Figure 4.1-2 Atlas Static Payload Envelope

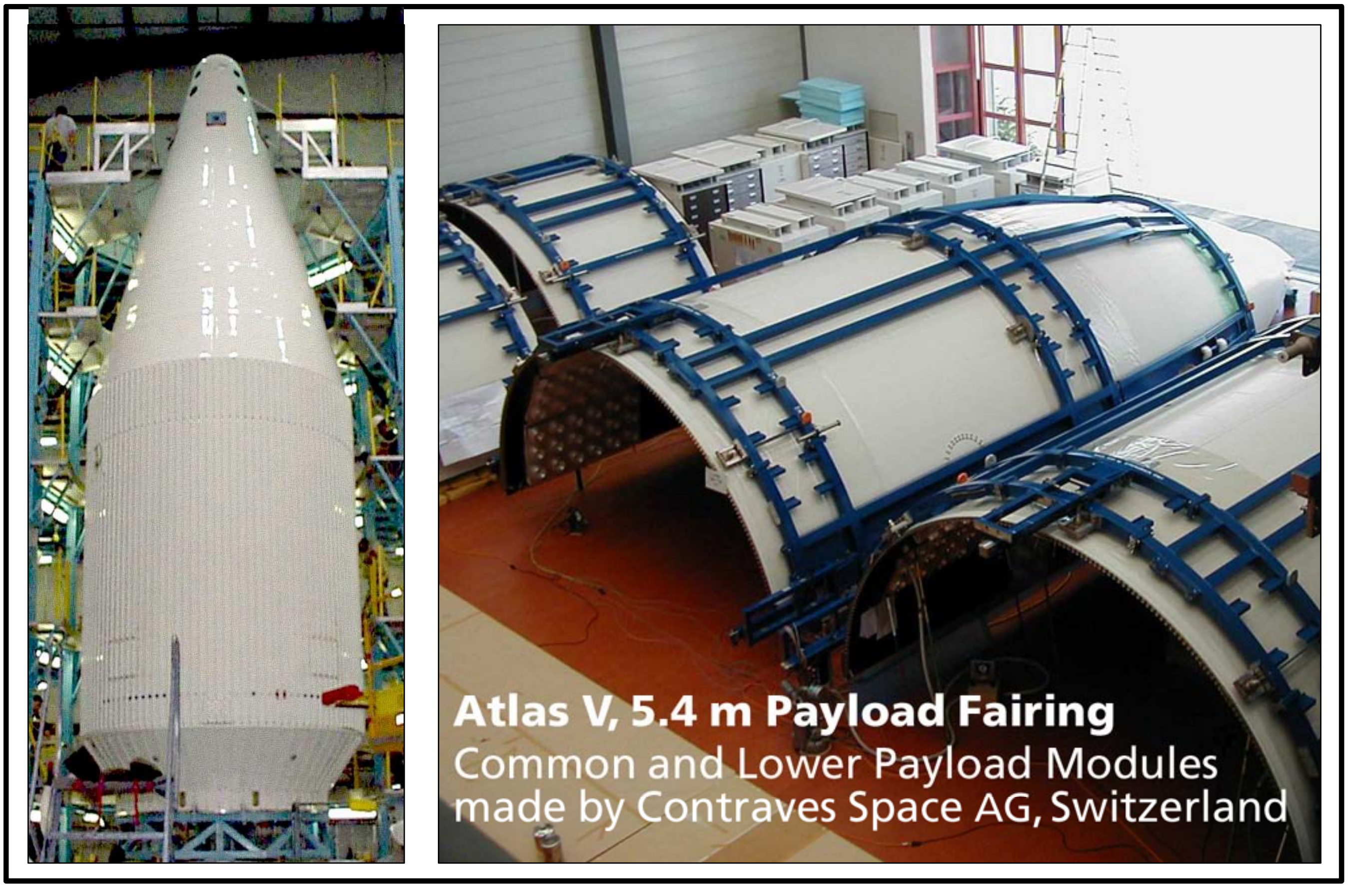


Figure 4.1.1-1 Atlas Payload Fairings

developed to support launch of larger volume spacecraft by adding a 0.9-m (36-in.) high cylindrical plug to the top of the cylindrical section of the LPF. The XEPF is a modified version of the EPF that incorporates an additional 0.9-m (36-in.) high cylindrical plug to increase the available payload volume. The XEPF is under development and information on the XEPF design and payload envelope capabilities is available upon request.

The fairing is designed to provide a controlled environment for spacecraft. On the Atlas III and Atlas V vehicles, acoustic panels are provided in the cylindrical section of the fairing to attenuate the sound pressure levels to acceptable limits. These panels may also be added to the Atlas IIAS vehicle as a mission-specific option. For thermal control, the external surface of the conical section fairing is insulated with cork to limit temperatures to acceptable values. Thermal shields may be added in the conical section of the fairing to provide additional thermal control. During prelaunch activities, conditioned air is provided through the air-conditioning duct that is located in the cylindrical and nose cone portion of the fairing. Vent holes and housings are mounted on the lower part of the cylindrical section for the LPF and EPF to allow air from the air-conditioning system to exit the fairing and to allow depressurization during ascent. A secondary environmental control system may be added as a mission-specific option to provide additional cooling or to direct cooling air to specific points on the payload.

The four large doors in the boattail section of the 4-m PLFs provide primary access to Centaur forward adapter packages and the encapsulated spacecraft. Work platforms can be inserted through these doors into the payload compartment to allow access to spacecraft hardware near the aft end of the payload compartment. If access to other portions of the spacecraft is required, additional doors can be provided on a mission-specific basis on the cylindrical section of each payload fairing.

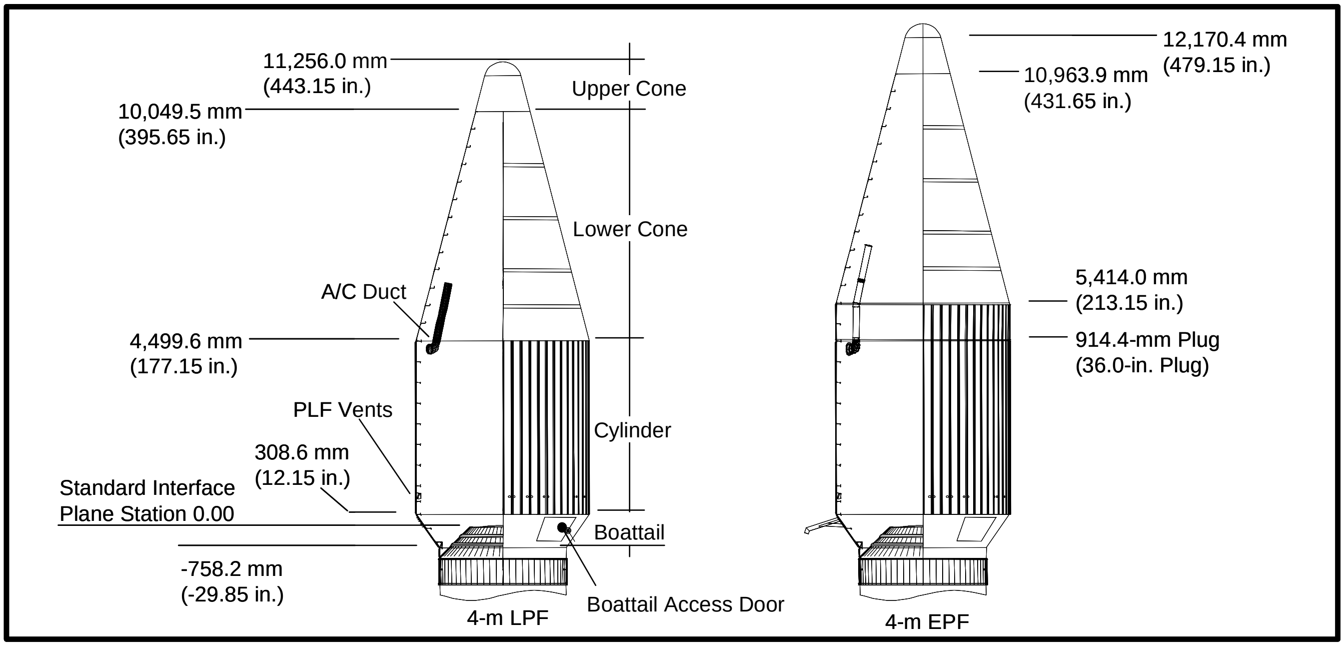


Figure 4.1.1.1-1 Atlas 4-m Payload Fairings

A customer-specified logo may be placed on the cylindrical section of the payload fairing. The Atlas program will work with the customer and provide layouts of the logo on the launch vehicle to assist in determining the proper size and location for these logos.

Additional mission-specific items that can be mounted on the payload fairing to provide support services for the payload include an RF reradiating antenna. The reradiating antenna allows spacecraft RF communications after the spacecraft is encapsulated inside the payload fairing. Standard and mission-specific features of the payload fairing are shown in Figure 4.1.1.1-2. Detailed information on the Atlas 4-m payload fairings may be found in Appendix D.

4.1.1.2 Atlas V 5-m Payload Fairing (5-m Short PLF and 5-m Medium PLF)—The Atlas V 5-m PLF was developed along with the increased launch vehicle performance to accommodate evolving spacecraft needs. This fairing is a bisector fairing with a composite structure made from sandwich panels with carbon fiber facesheets and a vented aluminum honeycomb core. There are two major components of the fairing. The lower section is the base module that encapsulates the Centaur upper stage. The upper section is the common payload module (CPM) that encapsulates the spacecraft. The CPM consists of a cylindrical section that transitions into a constant radius ogive nose section topped by a spherical nose cap (Fig. 4.1.1.2-1). For the 5-m medium payload fairing, a 2,743-mm (108-in.) lower payload module section is added to the base of the common payload module to increase the available payload volume. The fairing interfaces with the launch vehicle at the fixed conical boattail that is attached to the launch vehicle first stage. Clearance losses for payloads are minimized by the Centaur forward-load reactor (CFLR) system that stabilizes the top of the Centaur and thereby reduces the relative motion between the PLF, Centaur, and payload. PLF sections provide mounting provisions for various secondary systems.

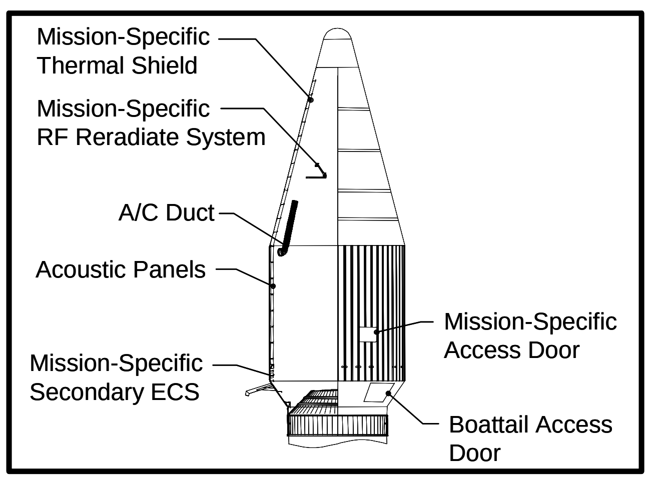


Figure 4.1.1.1-2 Atlas 4-m Payload Fairing Interface Features

Payload compartment cooling system provisions are in the ogive-shaped portion of the fairing. Electrical packages required for the fairing separation system are mounted on the internal surface of the fairing.

The fairing and boattail provide a protective enclosure for the spacecraft and Centaur upper stage during prelaunch and ascent. Fairing acoustic protection (FAP) is provided as a standard service to attenuate acoustic sound pressure levels to acceptable limits. For thermal control, external fairing surfaces are insulated with cork and painted white to limit temperatures to acceptable values. During prelaunch activities, conditioned air is provided through the air-conditioning inlet that is located in the ogive section of the fairing. This inlet directs conditioned air to provide thermal and humidity control for the payload compartments and prevents direct impingement of this flow on the spacecraft. Vent ports and vent port assemblies are mounted in the mid-section of the base module for air from the air-conditioning system to exit the fairing and to allow depressurization during ascent. A secondary environmental control system may be added as a mission-specific option to provide additional cooling or to direct cooling air to specific points on the payload.

The 5-m payload fairings have four large doors in the base module portion of the PLF to provide primary access to the Centaur upper stage equipment module. These doors also allow access to the spacecraft hardware near the aft end of the payload module. If access to other portions of the spacecraft is required, additional doors can be provided on a mission-specific basis.

A customer-specified logo may be placed on the cylindrical section of the payload fairing. The Atlas program will work with the customer and provide layouts of the logo on the launch vehicle to assist in determining the proper size and location for these logos.

Additional mission-specific items that can be mounted on the payload fairing to provide support services for the payload include an RF reradiating antenna. The reradiating antenna allows spacecraft RF

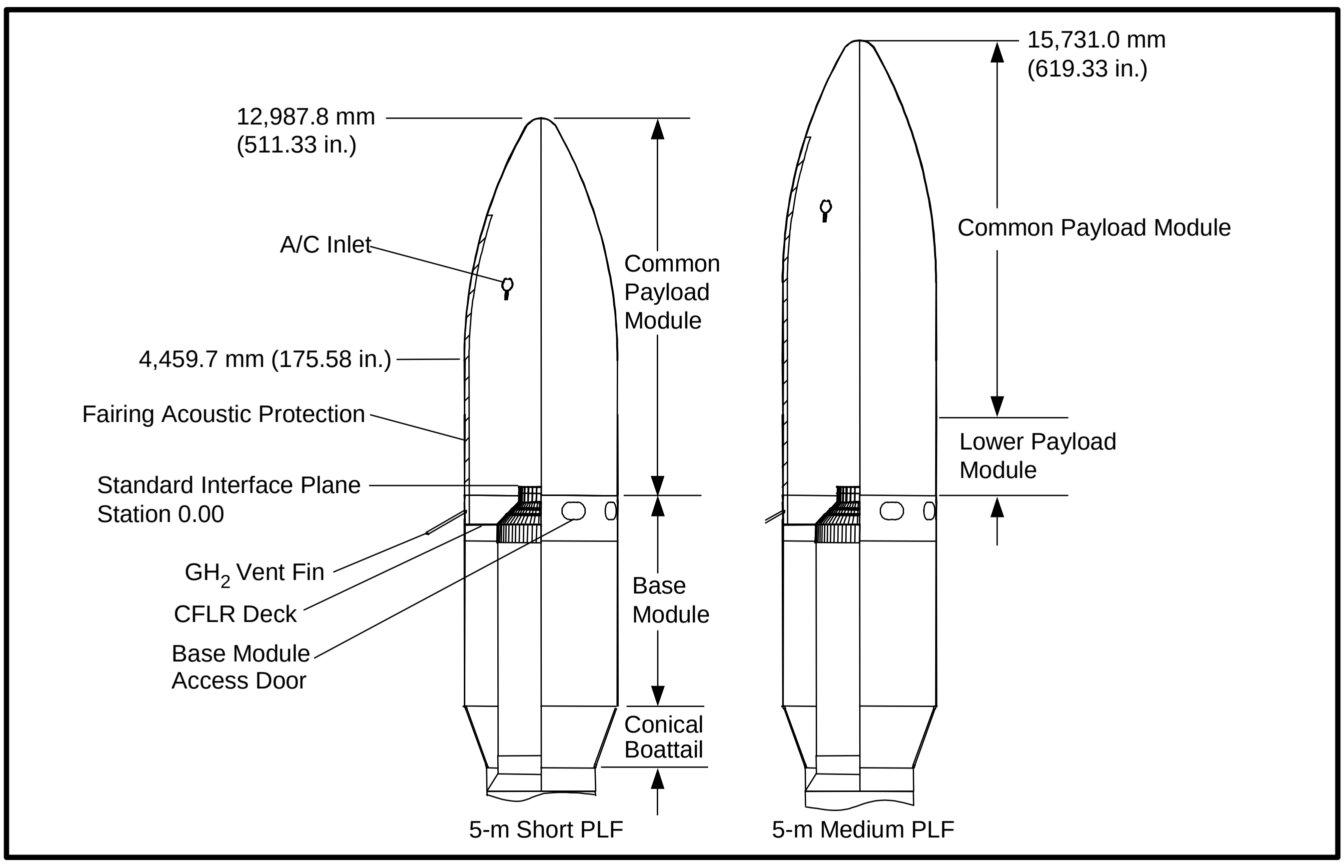


Figure 4.1.1.2-1 Atlas 5-m Payload Fairings

communications after the spacecraft is encapsulated inside the payload fairing. Detailed information on the Atlas V 5-m payload fairings may be found in Appendix D.

4.1.2 Mechanical Interface—Payload Adapters

The Atlas offers a range of mechanical interface options including standardized bolted interfaces, payload adapters, and trusses that mate the spacecraft to the launch vehicle. These options include provisions for mechanical and electrical interfaces between the launch vehicle and spacecraft and for mission-specific services. Payload interface options currently offered by the Atlas program include a bolted interface at the standard interface plane, Atlas heritage payload adapters that were developed for the Atlas I/II/III vehicles, Atlas V standard payload adapters that have been designed to handle heavier payloads that can take advantage of the higher performance of the Atlas V vehicle, and truss interfaces that allow the user to take advantage of the full volume and performance capabilities of the Atlas V 500 series. These payload interface options are summarized in Figure 4.1.2-1. Payload capabilities for these adapters are shown in Figure 4.1.2-2. These spacecraft mass and cg capabilities were determined using generic spacecraft interface ring geometry as shown in Appendix E, and quasi-static load factors shown in Section 3.2.1. Actual spacecraft design allowables may vary depending on interface ring stiffness and results of spacecraft mission-specific coupled loads analyses. Coordination with the Atlas program is required to define appropriate structural capabilities for spacecraft designs that exceed these generic allowables.

4.1.2.1 Atlas Standard Interface Plane—The reference plane for the spacecraft to launch vehicle interface is the Atlas standard interface plane (Fig. 4.1.2.1-1). For vehicles using a 4-m payload fairing, this plane is at the top of the Atlas forward adapter. For vehicles using a 5-m payload fairing, this plane occurs at the top of an Atlas program-provided C22 launch vehicle adapter that is mounted on top of the forward adapter. The C22 adapter is standard with the 5-m fairing to allow launch vehicle ground support equipment interfaces and is designed to provide an interface that is identical to that provided by the forward adapter. This C22 adapter is considered to be part of the basic launch vehicle and does not count against the payload systems weight (PSW). For vehicles using a 5-m payload fairing and a launch vehicle-provided truss adapter, the C22 adapter is not used and the standard interface plan occurs at the top of the truss.

For customers that provide their own adapter, the standard interface plane is the interface point between the spacecraft-provided hardware and the launch vehicle. If a customer-provided spacecraft adapter is used, it must provide interfaces for ground handling, encapsulation, and transportation equipment. In particular, there will need to be provisions for torus arm fittings and an encapsulation diaphragm unless a launch vehicle-supplied intermediate adapter is used. Detailed information on the standard interface plane and on interfaces to launch vehicle flight and ground support equipment (GSE) hardware may be found in Appendix E.

4.1.2.2 Atlas Launch Vehicle Adapters—The Atlas launch vehicle's adapters were developed to provide a common interface for launch vehicle-required ground support equipment that interfaces with payload adapter systems. The launch vehicle adapter is a machined aluminum component in the form of an integrally stiffened cylinder and is available in heights from 330.2 mm (13.00 in.) to 736.6 mm (29.00 in.). The standard configurations available are the C13 (330.2-mm/13.00-in. high), C15 (384.8-mm/15.15-in. high), and C22 (558.8-mm/22-in. high) adapters (Fig. 4.1.2.2-1).

On the Atlas V 500 vehicle, a C22 launch vehicle adapter is mounted to the top of the Atlas forward adapter and provides an interface surface and hole-pattern at its forward end that is compatible with standard interface plane requirements. The C22 adapter is standard with the 5-m payload fairing to

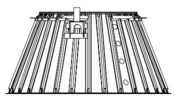
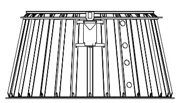
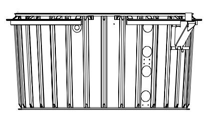
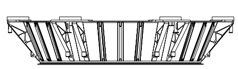
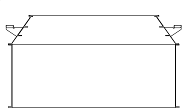
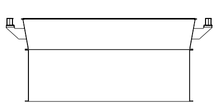
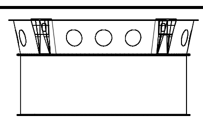
Standard Interface Plane		
	Bolted Interface, 121 Fasteners Bolt Circle Diameter: 1,575.0 mm/62.010 in.	Appendix E.1
Launch Vehicle Adapters		
C13 C15 C22 Cxx-(Mission Specific)	Integrally Machined Aluminum Forging Bolted Interface, 120 or 121 Fasteners Bolt Circle Diameter: 1,575.0 mm/62.010 in. Height: 330.2 mm/13 in. to 736.6 mm/29 in.	Appendix E.2
C13Light C15Light C22Light Cxx-(Mission Specific) 	Integrally Machined Aluminum Forging Bolted Interface, 120 or 121 Fasteners Bolt Circle Diameter: 1,575.0 mm/62.010 in. Height: 330.2 mm/13 in. to 736.6 mm/29 in.	In Development
Atlas Heritage Payload Adapters		
A 	Aluminum Skin/Stringer/Frame Construction PSS37 Marmon-Type Clampband Separation System Forward Ring Diameter: 945.3 mm/37.215 in. Height: 762.0 mm/30.00 in. Mass: 44 kg (97 lb)	Appendix E.3
B1 	Aluminum Skin/Stringer/Frame Construction PSS47 Marmon-Type Clampband Separation System Forward Ring Diameter: 1,215.0 mm/47.835 in. Height: 812.8 mm/32.00 in. Mass: 52 kg (116 lb)	Appendix E.4
D 	Aluminum Skin/Stringer/Frame Construction PSS66 Marmon-Type Clampband Separation System Forward Ring Diameter: 1,666.1 mm/65.594 in. Height: 889.0 mm/35.00 in. Mass: 54 kg (119 lb)	Appendix E.5
E 	Aluminum Skin/Stringer/Frame Construction 6 Separation Bolt Interface Bolt Circle Diameter: 1,969.6 mm/77.543 in. Height: 422.9 mm/16.65 in. Mass: 98 kg (215 lb)	Appendix E.6
Atlas V Standard Payload Adapters		
B1194VS 	Integrally Machined Aluminum Forging PSS47VS Low-Shock Marmon-Type Clampband Separation System Forward Ring Diameter: 1,215.0 mm/47.835 in. Height: 812.8 mm/32.00 in. Mass: 91 kg (200 lb)	Appendix E.7
D1666VS 	Integrally Machined Aluminum Forging PSS66VS Low-Shock Marmon-Type Clampband Separation System Forward Ring Diameter: 1,666.1 mm/65.594 in. Height: 889.0 mm/35.00 in. Mass: 91 kg (200 lb)	Appendix E.8
F1663 	Integrally Machined Aluminum Forging 4 Separation Bolt Interface, FASSN Low-Shock System Bolt Circle Diameter: 1,663.7 mm/65.50 in. Height: 889.0 mm/35.00 in.	Appendix E.9
Atlas V Truss Adapters		
T4394 (173 in.)	Bolted Interface, 18 Places Forward Ring Bolt Circle Diameter: 4,394.2 mm/173.0 in. Height: 1,168.4 mm/46.00 in.	Section 4.1.2.5
T3302 (130 in.)	Bolted Interface, 18 Places Forward Ring Bolt Circle Diameter: 3,302.0 mm/130.0 in. Height: 914.4 mm/36.00 in.	Section 8.4.1

Figure 4.1.2-1 Atlas Payload Support Interface Options

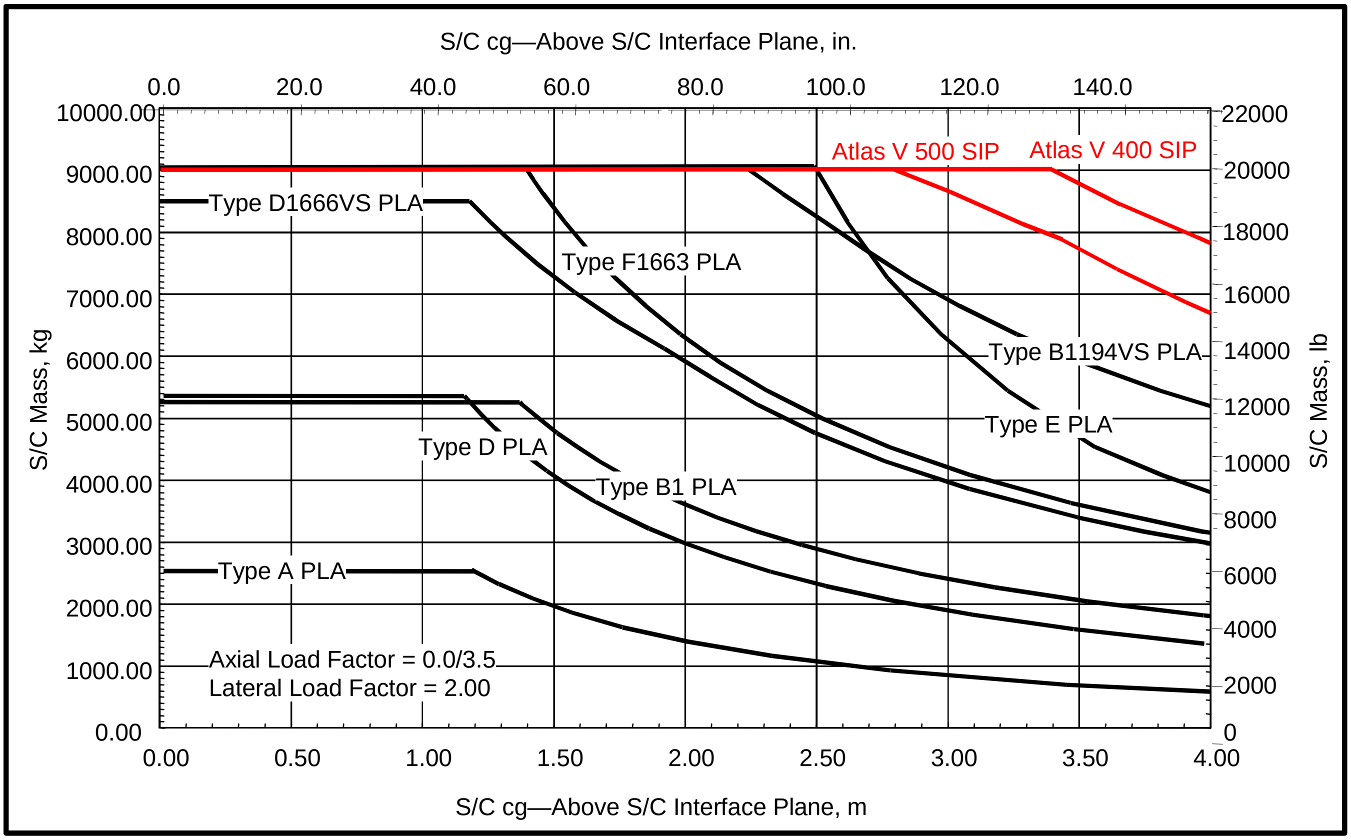


Figure 4.1.2-2 Atlas Payload Support System Structural Capabilities

allow clearance for launch vehicle ground support equipment. In this configuration, cost and performance impacts of the C22 launch vehicle adapter are considered to be a part of the basic launch vehicle service and are not counted against payload systems weight.

The Atlas program has adapted a modified version of the launch vehicle adapter as a component of Atlas V standard payload adapters. This allowed creation of a modular series of payload adapters that have common interfaces to the launch vehicle flight hardware and ground support equipment on a component that is separate from the mission specific requirements of the spacecraft.

For customers that provide their own payload adapter and payload separation system, launch vehicle adapters are available as a mission-specific option. This allows the customer to raise the position of the standard interface plane relative to the launch vehicle for additional clearance or to take advantage of standard GSE interfaces that are built into the launch vehicle adapter. The Atlas program is also developing lighter weight versions of these adapters, with reduced structural capability, for lighter weight spacecraft. Information on these adapters is available to customers upon request.

4.1.2.3 Atlas Heritage Payload Adapters—Atlas heritage payload adapters were developed for the Atlas I/II/III vehicles and are compatible with and available for use on the Atlas V 400 and 500 series. Their use on the Atlas V vehicles is only limited by their structural capability. Available systems include the Atlas A, B, D, and E payload adapters. These adapters use an aluminum skin, stringer, and frame construction with machined forward and aft rings that mate to the spacecraft and launch vehicle forward adapter (Fig. 4.1.2.3-1). These adapters include provisions for supporting all hardware that directly interfaces with spacecraft including the payload separation system, electrical connectors, and mission-specific options, and include all provisions for mating to the launch vehicle ground support equipment, including the torus arms and isolation diaphragm, used during ground processing operations.

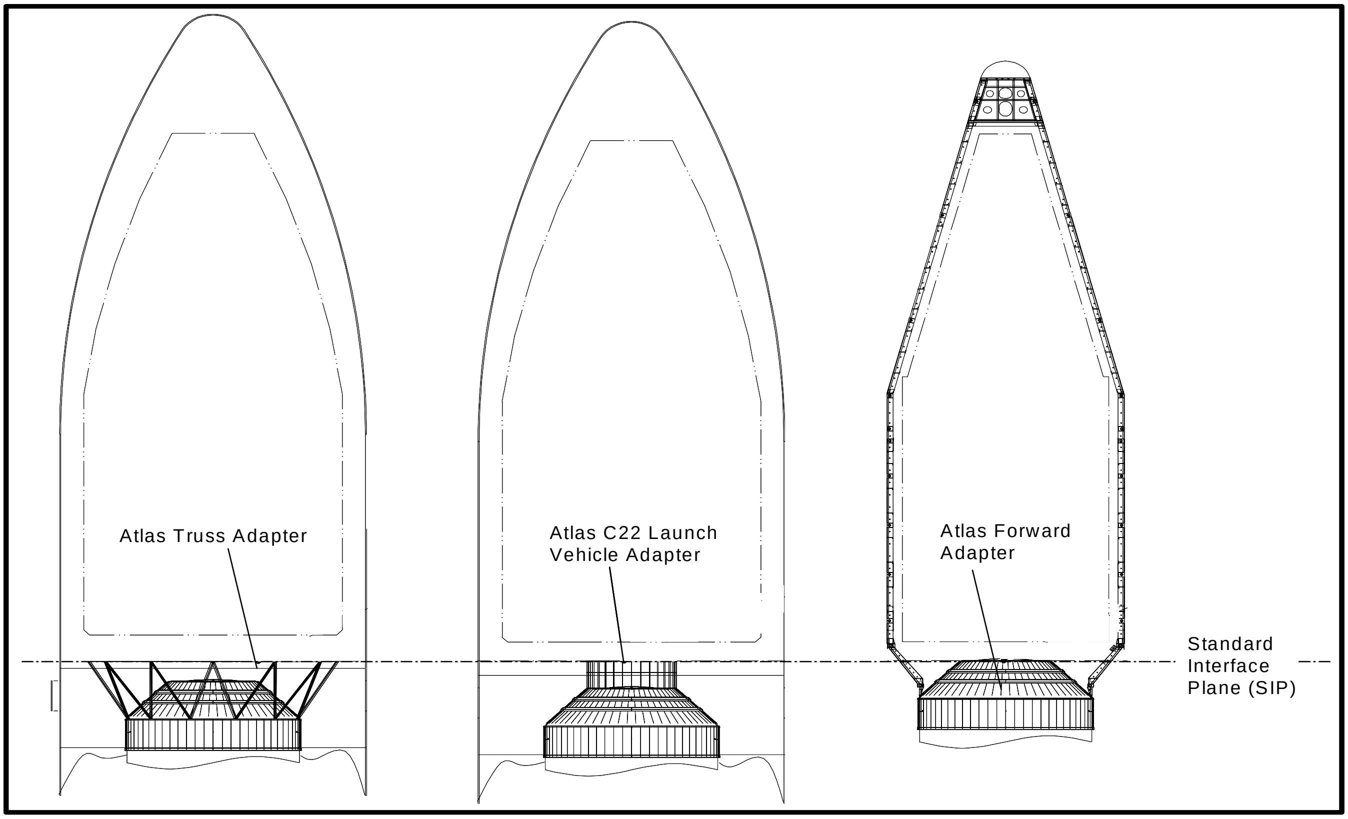


Figure 4.1.2.1-1 Atlas Standard Interface Plane

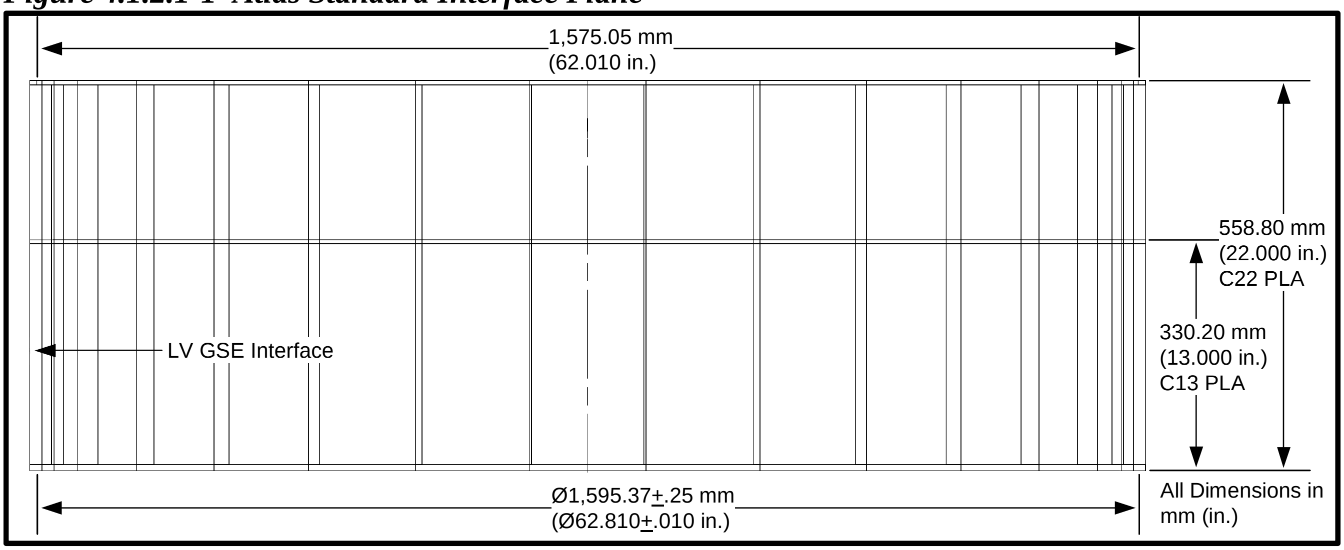


Figure 4.1.2.2-1 Atlas Launch Vehicle Adapter Configuration

The Atlas Type A, B1, and D adapters use a launch vehicle provided Marmon-type clampband payload separation system (Fig. 4.1.2.3-2). This separation system consists of a clampband set, release mechanism, and separation springs. The clampband set consists of a clampband for holding the spacecraft and adapter rings together plus devices to extract, catch, and retain the clampband on the adapter structure after separation. The release mechanism consists of a pyrotechnically activated bolt-cutter that severs the tension bolts allowing the end of the clampband segments to move apart and release the payload adapter and spacecraft mating rings. Separation spring assemblies provide the necessary separation energy after the clampband is released and are mounted to the payload adapter forward ring and bear on the spacecraft aft ring. Positive spacecraft separation is detected through

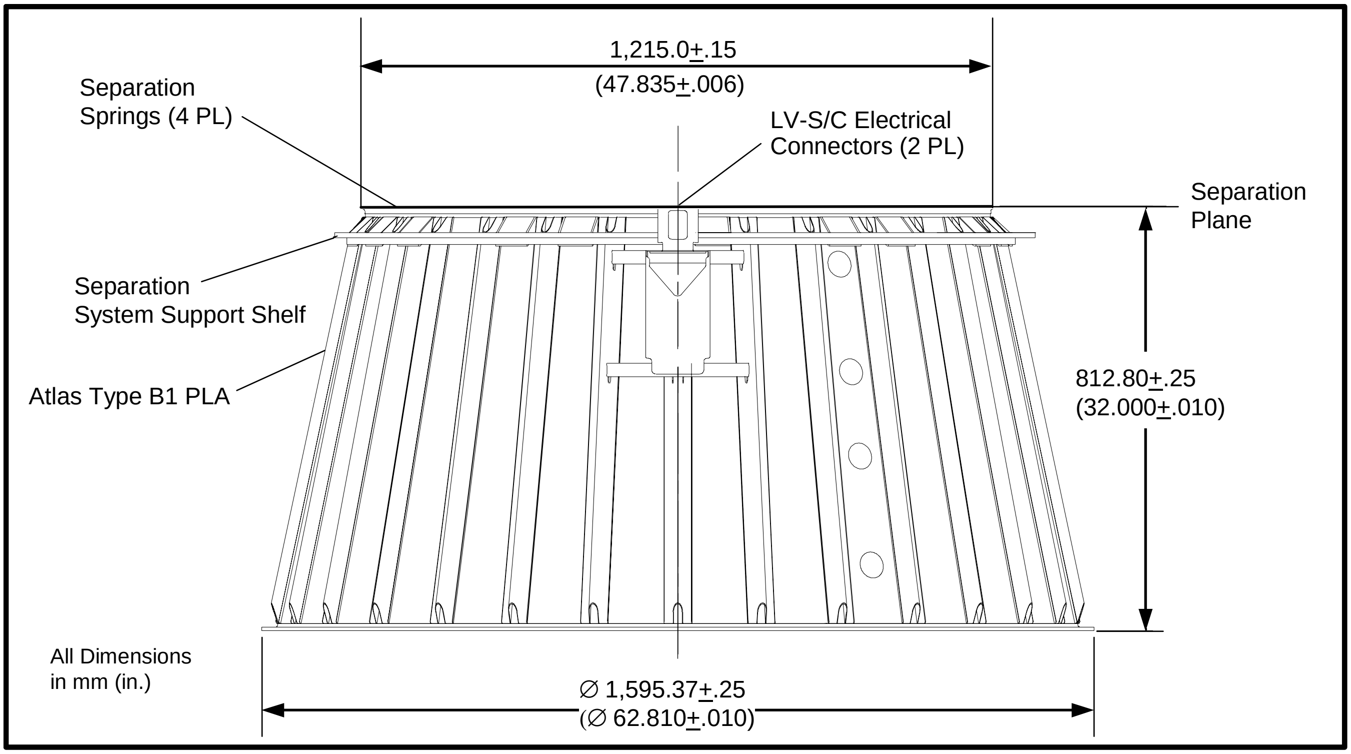


Figure 4.1.2.3-1 Atlas Heritage Payload Adapter Configuration

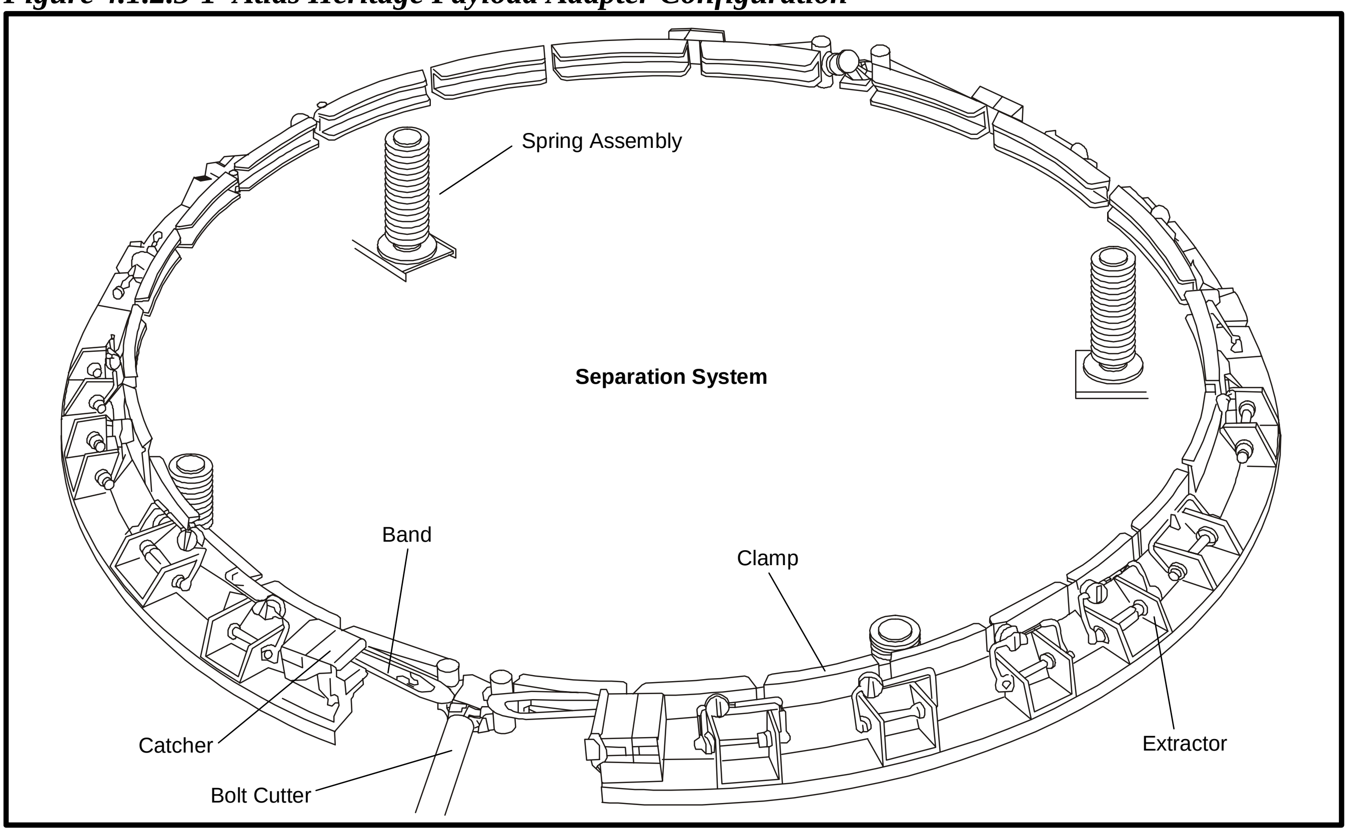


Figure 4.1.2.3-2 Atlas Heritage Payload Separation System Configuration

continuity loops installed in the spacecraft electrical connector and wired to the upper stage instrumentation for monitoring and telemetry verification.

The Atlas Type E adapter is designed for spacecraft that mate to the launch vehicle using six-discrete hard points on an interface diameter of 1.956 mm (77 in.). This payload adapter uses a launch vehicle-provided payload six-separation nut set that attaches the spacecraft to the forward ring of the payload adapter, and a separation spring set that provides the necessary separation energy after the separation nut is actuated. The separation nut set consists of the separation nut, separation stud, and stud catcher that captures the stud on the payload structure after separation. For spacecraft requiring a lower shock environment, there is the option of using a fast-acting shockless separation nut (FASSN) with the Type E adapter. The FASSN system is designed to rapidly separate a bolted joint while producing minimal shock. Separation springs are mounted on the aft side of the payload adapter forward ring, and the pushrod of each spring acts through a hole in the forward ring to push against the payload aft structure. Positive spacecraft separation is detected through continuity loops installed in the spacecraft interface connector and wired to the upper stage instrumentation for monitoring and telemetry verification. Coordinated tooling between the spacecraft and payload adapter is required for this system. Detailed information on each of the Atlas heritage payload adapters may be found in Appendix E.

4.1.2.4 Atlas V Standard Payload Adapters—The Atlas V standard payload adapters have been designed to handle heavier payloads that can take advantage of the higher performance of the Atlas V vehicle. The available systems include the Atlas B1194VS, D1666VS, and F1663 payload adapters.

These payload adapters consist of two major sections: the payload separation ring and the launch vehicle adapter (Fig. 4.1.2.4-1). The payload separation ring is a machined aluminum component in the form of a truncated cone. The forward ring forms the spacecraft separation plane. The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 120 evenly spaced holes that allow it to be joined to the launch vehicle adapter. This symmetrical hole pattern allows the payload separation ring and attached spacecraft to be rotated relative to the launch vehicle in 3° increments to meet mission-specific requirements. The payload separation ring supports all hardware that directly interfaces with spacecraft including the payload separation system, electrical connectors, and mission-specific options.

The launch vehicle adapter is a machined aluminum component in the form of an integrally stiffened cylinder. The forward ring has an outer diameter of 1,595 mm (62.81 in.) and contains 120 holes spaced evenly every 3° that allow it to be joined to the payload separation ring. The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 121 holes that match up with Atlas standard interface plane requirements. The nominal height of the launch vehicle adapter is 558.8 mm (22.00 in.) but this height may be varied from 330.2 mm (13.00 in.) to 736.6 mm (29.00 in.) to meet mission-specific

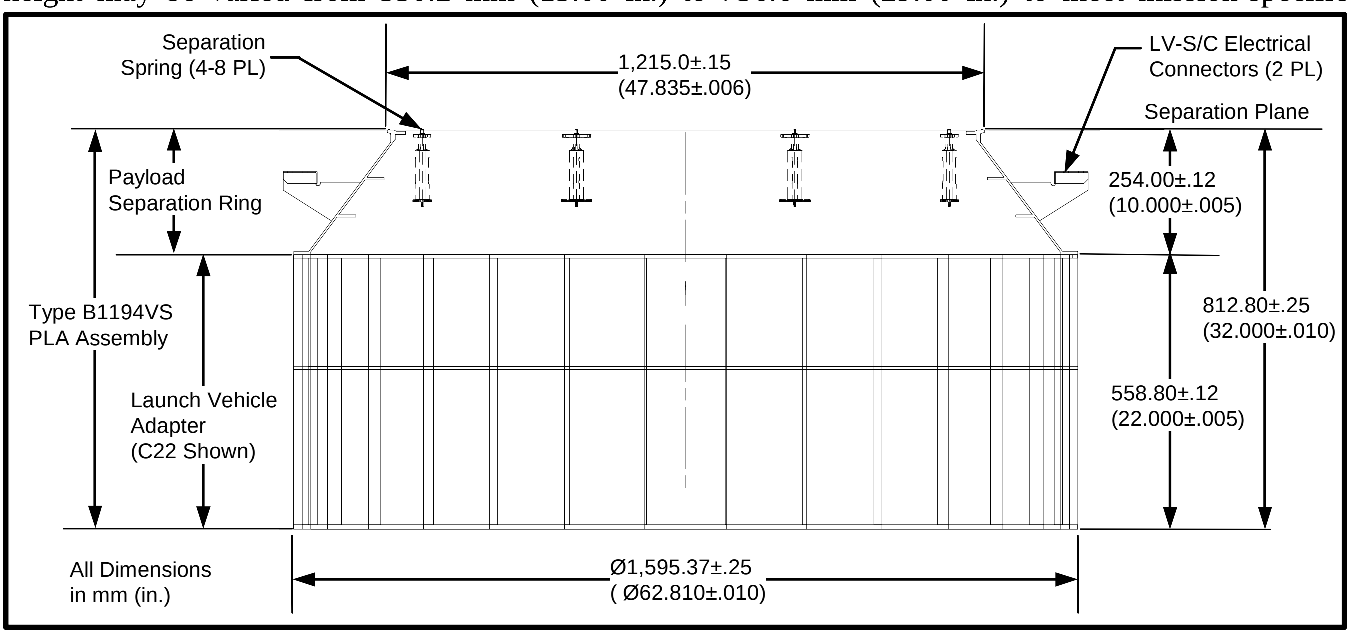


Figure 4.1.2.4-1 Atlas V Standard Payload Adapter Configuration (Type B194VS Shown)

requirements. The launch vehicle adapter includes all provisions for mating to the launch vehicle ground support equipment, including the torus arms and isolation diaphragm, used during ground processing operations.

The Atlas Type B1194VS and D1666VS payload adapters use a launch vehicle-provided Marmon-type clampband payload separation system. This separation system (Fig. 4.1.2.4-2) consists of a clampband set, release mechanism, and separation springs. The clampband set consists of a clampband for holding the spacecraft and adapter rings together plus devices to catch and retain the clampband on the adapter structure after separation. The clampband includes aluminum clamp segments that hold the payload adapter and spacecraft rings together and a single piece aluminum retaining-band that holds the clamp segments in place. The low-shock clampband opening device (CBOD) holds the ends of the retaining band together. The CBOD includes release bolts that engage the ends of the clampband. These release bolts are threaded into a flywheel mechanism. During installation and flight, the flywheel is restrained against rotation by a restraining pin. For separation, a pyrotechnically activated pin-puller retracts this pin from the flywheel, allowing it to rotate and eject the release bolts. This system significantly reduces shock compared to a conventional bolt-cutter system and is resettable, allowing the actual flight hardware to be tested during component qualification and acceptance testing.

The Atlas Type F1663 adapter is being developed to support a payload using a four-hard point interface on an interface diameter of 1,663 mm (65.50 in). This payload adapter uses a launch vehicle-provided FASSN system that attaches the spacecraft to the forward ring of the payload adapter, and a separation spring set that provides the necessary separation energy after the separation nut is actuated. The FASSN set consists of a bolt, actuator, and retractor that capture the bolt on the payload structure after separation. The FASSN system is designed to rapidly separate a bolted joint while producing minimal shock. The separation spring is mounted on the aft side of the payload adapter forward ring, and the pushrod of each spring acts through a hole in the forward ring to push against the payload aft structure. Positive spacecraft separation is detected through continuity loops installed in the spacecraft interface connector and wired to upper stage instrumentation for monitoring and telemetry verification.

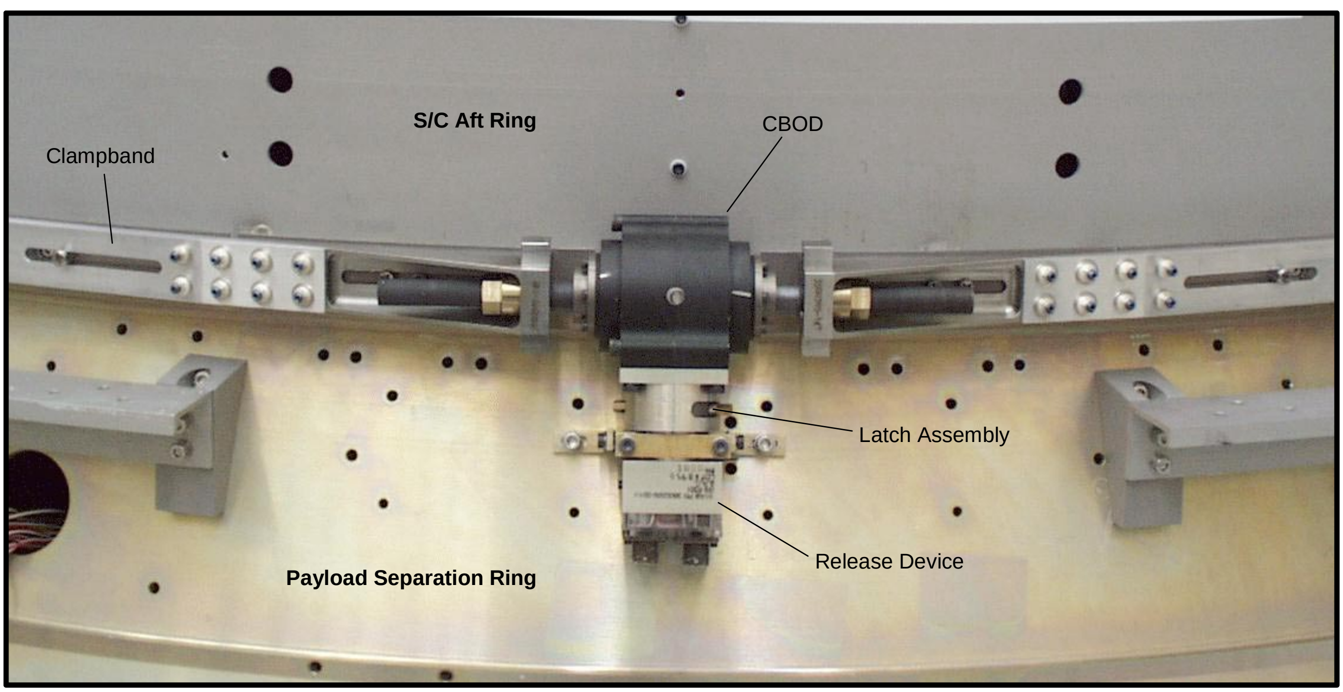


Figure 4.1.2.4-2 Atlas V Low-Shock Payload Separation System Configuration

Coordinated tooling between the spacecraft and payload adapter is required for this system. Detailed information on each Atlas V standard payload adapter is in Appendix E.

4.1.2.5 Atlas V 4,394-mm (173-in.) Diameter Payload Truss—A payload truss with a forward spacecraft interface of 4,394-mm (173-in.) diameter has been developed to meet future payload interface requirements. The 1,168-mm (46-in.) high truss attaches at 12 locations to the forward adapter on the Centaur. The forward ring of the payload truss has 18 equally spaced mounting points for the spacecraft, but could be easily modified to accommodate alternate mounting configurations. The truss has 24 graphite-epoxy struts with titanium alloy end fittings, aluminum alloy forward and aft brackets, and an aluminum alloy forward ring. When this truss is used, the static payload envelope extends to the top surface of the payload support truss forward ring.

4.1.3 Electrical Interfaces

Spacecraft and launch vehicle electrical interfaces are shown in Figures 4.1.3-1 and 4.1.3-2. Standard interfaces include:

1) A spacecraft-dedicated umbilical interface between the umbilical disconnect located on the Centaur upper stage and electrical in-flight disconnects (IFD) at the spacecraft and launch vehicle interface;
2) Spacecraft and launch vehicle separation indicators (continuity loop wiring) located in the spacecraft and launch vehicle in-flight disconnects to verify separation;
3) Standard in-flight disconnects or other customer-supplied connectors that may be required by mission-peculiar requirements. For standard connectors, the following part numbers (or equivalent substitutes) apply:

37 Contact	MS3446E37-50P	(LV)
	MS3464E37-50S	(SC)
61 Contact	MS3446E61-50P	(LV)
	MS3464E61-50S	(SC)

The launch vehicle can also be configured to provide electrical interfaces for various mission-peculiar requirements. The following paragraphs describe the Atlas electrical interfaces in more detail.

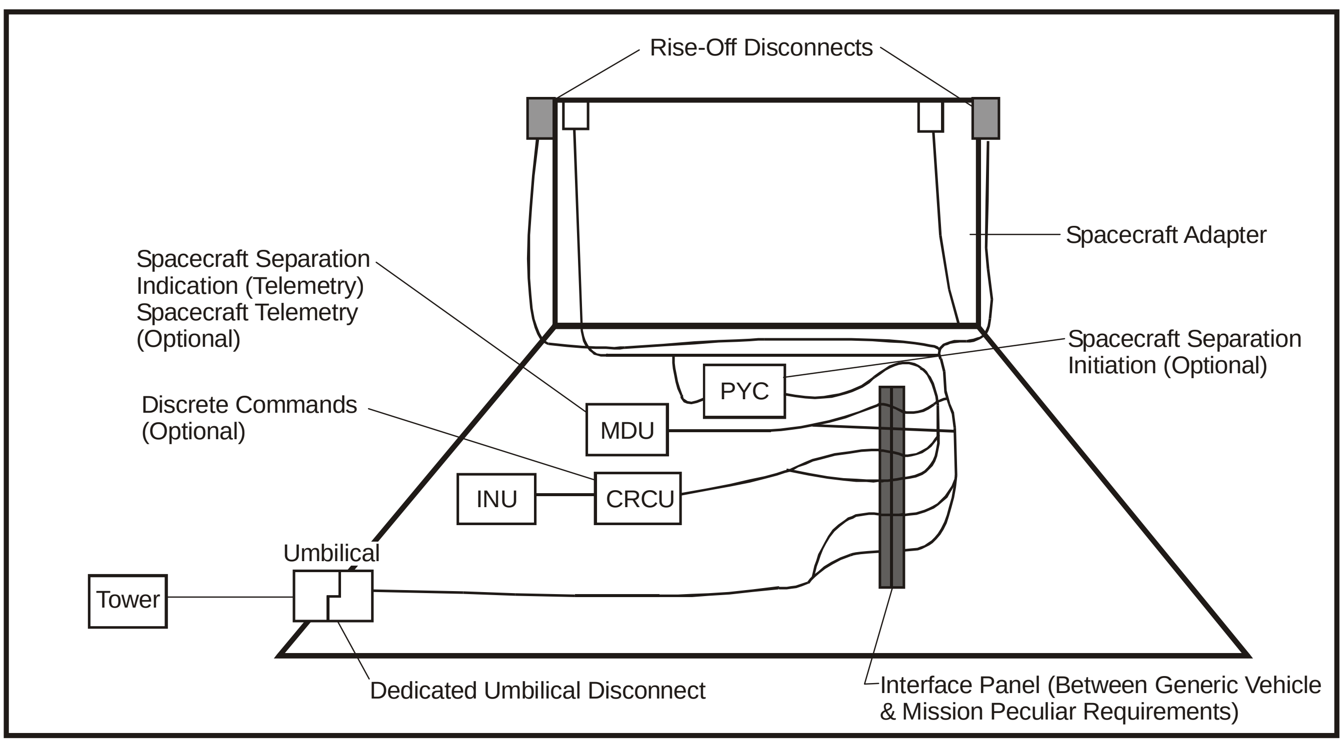


Figure 4.1.3-1 Typical Spacecraft/Launch Vehicle Electrical Interface

4.1.3.1 Umbilical Spacecraft-to-GSE Interface—One spacecraft-dedicated umbilical disconnect for on-pad operations is located on the Centaur forward umbilical panel. This umbilical interface provides signal paths between the spacecraft and GSE for spacecraft system monitoring during prelaunch and launch countdown. The umbilical disconnect separates at liftoff from the Centaur receptacle. The spacecraft in-flight disconnect connectors disengage during separation of the spacecraft from the Centaur. The following are the complement of spacecraft umbilical wires for the Atlas vehicles.

Atlas IIAS, IIIA, IIIB

1) Forty-one twisted shielded wire pairs—20 American wire gauge (AWG),
2) Six twisted shielded wire triples—20 AWG,
3) Six twisted shielded controlled impedance wire pairs—78 ohms.

Atlas V

1) Twelve twisted wire pairs—12 AWG,
2) Sixty twisted shielded wire pairs—20 AWG,
3) Four twisted shielded wire triples—20 AWG,
4) Six twisted shielded controlled impedance wire pairs—78 ohms.

4.1.3.2 Electrical In-Flight Disconnects—Standard payload adapters provide two in-flight disconnect options of either 37 pins or 61 pins for the spacecraft interface. These in-flight disconnects typically provide a spacecraft dedicated umbilical interface between the spacecraft and GSE, and any LV command and monitoring required to support the spacecraft during ascent. Lockheed Martin can provide more disconnects on a mission-peculiar basis if the spacecraft requires them. Mechanical interface requirements for the disconnects are shown in Appendix E.

4.1.3.3 Spacecraft Separation System—The baseline separation system for spacecraft and launch vehicle separation is a pyrotechnic V-type clampband system. The separation sequence is initiated by redundant commands from the upper stage guidance system. Positive spacecraft separation is detected by continuity loops installed in the spacecraft in-flight disconnects that are wired to the Centaur instru-

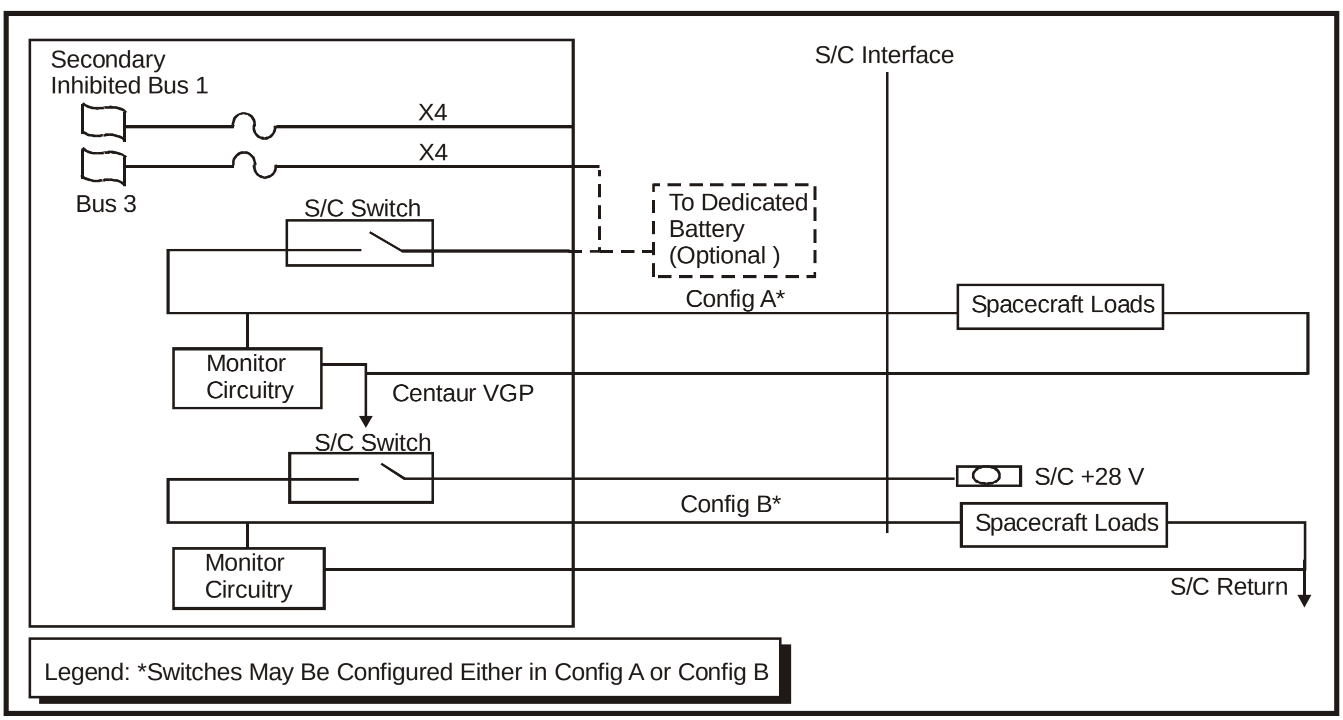


Figure 4.1.3-2 Spacecraft Switch Interface

mentation system. The separation event is then telemetered to the ground.

4.1.3.4 Control Command Interface—For the Atlas V vehicle, the Centaur remote control unit (CRCU) provides as many as 16 control commands to the spacecraft. For the Atlas IIA, IIAS, IIIA, and IIIB vehicles, two switches are reserved for solid rocket booster inadvertent separation and destruct system (ISDS) functions, resulting in up to 14 being available for spacecraft use. These circuits are solid-state switches configured as either launch vehicle 28-Vdc discrete commands or switch closure (dry-loop) commands depending on spacecraft requirements. The inertial navigation unit (INU) controls the switches. Digital data from the INU are decoded in the CRCU and addressed switches are energized or de-energized under INU software control. The basic switch configuration is shown in simplified form in Figure 4.1.3-2. Command feedback provisions are also incorporated to ensure that control commands issued to the spacecraft are received through spacecraft and launch vehicle in-flight disconnects. The spacecraft is responsible for providing a feedback loop on the SC side of the interface, including fault isolation circuits.

4.1.3.5 Spacecraft Environment Instrumentation—Lockheed Martin offers a suite of instrumentation options to capture spacecraft environments.

4.1.3.5.1 Spacecraft Telemetry Interface—Mission Satisfaction Telemetry—A mission satisfaction telemetry kit is provided as an option by the launch vehicle to ascertain vehicle environments and ensure ICD requirements are met. Low frequency data (less than 200 Hz) can be captured through the launch vehicle data acquisition system (DAS) located on the Centaur. The Digital Telepak/tracking and data relay satellite system (TDRSS) transmitter combination can provide up to 32 channels of wideband data (up to 8-kHz data) with a pulse-code modulation (PCM) (binary phase-shift keying [BPSK]) telemetry format. Additional instrumentation may be added optionally up to the bandwidth and interface capabilities of these data systems. Data recorded from these measurements are provided, postlaunch, to evaluate compliance with ICD environmental requirements. The available mission satisfaction instrumentation consists of: (1) two acoustic measurements located inside the payload fairing, (2) three acceleration measurements at the spacecraft and launch vehicle separation plane spaced 120° apart, (3) three orthogonal vibration measurements at the spacecraft and vehicle separation plane, (4) three orthogonal shock measurements at the spacecraft and launch vehicle separation plane, (5) one absolute pressure measurement located in the spacecraft compartment, and (6) one temperature measurement located in the spacecraft compartment**.**

4.1.3.5.2 Spacecraft Telemetry Options—Lockheed Martin offers three options for transmission of spacecraft data.

4.1.3.5.2.1 RF Reradiation GSE Interface—For this option, a modification of the Atlas payload fairing is made to accommodate an RF reradiating antenna system to reradiate spacecraft RF telemetry and command signals from inside the fairing, to a GSE site before launch.

4.1.3.5.2.2 Spacecraft Serial Data Interface—The Atlas launch vehicle can provide transmission of two spacecraft serial data interfaces. Spacecraft data are interleaved with launch vehicle DAS data and serially transmitted in the PCM bit stream. For each data interface, the spacecraft provides nonreturn-to-zero level (NRZ-L) coded data and a clock from dedicated drivers (as an input to the launch vehicle). Spacecraft data and clock signals must be compliant with Electronics Industry Association (EIA) RS-422, Electrical Characteristics of Balanced Voltage Digital Interface Circuits, with a maximum data bit rate of 2 kbps. Spacecraft data are sampled by the launch vehicle on the falling edge of the spacecraft clock signal. The clock-to-data skew must be less than 50 microseconds and the signal and clock duty cycles must be 50% ±5%. Cabling from the signal driver to the launch vehicle has a nominal characteristic impedance of 78 ohms. Data are presented as the original NRZ-L data stream in real time for those portions of prelaunch and flight for which Atlas data are received. For postflight analysis, spacecraft data can be recorded to magnetic tape.

4.1.3.6 Spacecraft Destruct Option—If required for Range Safety considerations, Atlas can provide a spacecraft destruct capability. A safe and arm initiator receives the destruct command from the Centaur flight termination system (FTS). The initiator ignites electrically initiated detonators, which set off a booster charge. The charge ignites a mild detonating fuse that detonates a conically shaped explosive charge that perforates the spacecraft propulsion system.

4.2 SPACECRAFT-TO-GROUND EQUIPMENT INTERFACES

4.2.1 Spacecraft Console

Floor space is allocated on the operations level of the launch control facility (the blockhouse or Launch Service Building for LC-36, the Launch Service Building at SLC-3E, and the payload van at LC-41) for installation of a spacecraft ground control console(s). This console(s) is provided by the user, and interfaces with Lockheed Martin-provided control circuits through the dedicated umbilical to the spacecraft. Control circuits provided for spacecraft use are isolated physically and electrically from those of the launch vehicle to minimize electromagnetic interference (EMI) effects. Spacecraft that require a safe and arm function for apogee motors will also interface with the Range-operated pad safety console. Lockheed Martin will provide cabling between the spacecraft console and the pad safety console. The safe and arm command function for the spacecraft apogee motor must be inhibited by a switch contact in the pad safety console. Pad Safety will close this switch when it is safe to arm the system.

4.2.2 Power

Several types of electrical power are available at the launch complex for spacecraft use. Commercial ac power is used for basic facility operation. Critical functions are connected to an uninterruptible power system (UPS). The dual UPS consists of battery chargers, batteries, and a static inverter. UPS power is available for spacecraft use in the blockhouse, launch service building, umbilical tower, and payload van. Twenty-eight-Vdc power can be provided for spacecraft use in the blockhouse and the launch service building. Facility power supplies are operated on the UPS to provide reliable service.

4.2.3 Liquids and Gases

All chemicals used will be in compliance with requirements restricting ozone-depleting chemicals.

Gaseous Nitrogen (GN_2)—Three pressure levels of GN_2 are available on the service tower or Vertical Integration Facility (VIF) for spacecraft use. Nominal pressure settings are 13,790 kN/m^2 (2,000 psi), 689.5 kN/m^2 (100 psi), and approximately 68.95 kN/m^2 (10 psi). The 10-psi system is used for purging electrical cabinets for safety and humidity control.

Gaseous Helium (GHe)—Gaseous helium is available on mobile service towers at both Cape Canaveral Air Force Station (CCAFS) and Vandenberg Air Force Base (VAFB). At CCAFS, GHe at 15,169 kN/m^2 (2,200 psi) is available. At VAFB SLC-3, GHe at 589.5 kN/m^2 (100 psig) and 34,475 kN/m^2 (5,000 psig) is available.

Liquid Nitrogen (LN_2)—LN_2 is available at the CCAFS launch complex storage facility. LN_2 is used primarily by the Atlas pneumatic and LN_2 cooling systems. Small Dewars can be filled at Range facilities and brought to the Atlas launch complex for spacecraft use.

At VAFB, GN_2 is provided to payload pneumatic panels at SLC-3E. GN_2 is provided at 689.5 kN/m^2 (100 psig); 2,758 kN/m^2 (400 psig); 24,822 kN/m^2 (3,600 psig); and 34,475 kN/m^2 (5,000 psig).

Gaseous Xenon (Xe)—Gaseous xenon is available for use at CCAFS and VAFB. At both locations, xenon can be provided at various pressures and quantities subject to the user's specific needs.

4.2.4 Propellant and Gas Sampling

Liquids and gases provided for spacecraft use will be sampled and analyzed by the Range propellant analysis laboratory. Gases (e.g., helium, nitrogen, xenon, and breathing air) and liquids (e.g., hypergolic fuels and oxidizers), water, solvents, and hypergolic decontamination fluids may be analyzed to verify that they conform to the required specification.

4.2.5 Deleted

4.2.6 Work Platforms

The launch complex service tower provides work decks approximately 10-ft apart in the spacecraft area. Portable workstands will be provided to meet spacecraft mission requirements where fixed work decks do not suffice. Access can be provided inside the encapsulated nose fairing. Access requirements will be developed during the planning stage of each mission.

4.3 RANGE AND SYSTEM SAFETY INTERFACES

4.3.1 Requirements and Applicability

To launch from either CCAFS or VAFB, launch vehicle and spacecraft design and ground operations must comply with applicable launch site Range Safety regulations, U.S. Air Force (USAF) requirements concerning explosives safety, and U.S. consensus safety standards.

In addition, when using spacecraft processing facilities operated by Astrotech International Corporation, the National Aeronautics and Space Administration (NASA), or the USAF, compliance with applicable facility safety policies is also required.

CCAFS and VAFB Range Safety organizations have regularly updated their safety requirements documents. The single safety document for both CCAFS and VAFB (Eastern/Western Range Regulation [EWR] 127-1) was updated on 31 October 1997. Earlier versions of this regulation may still apply to a given spacecraft or mission, depending on when the spacecraft bus was originally designed and constructed and approved by the Eastern and/or Western Range Safety organizations. Earlier versions of Range Safety regulations include the following:

1) Eastern Range;
   a) Eastern Range Regulation (ERR) 127-1, June 1993;
   b) Eastern Space and Missile Center Regulation (ESMCR) 127-1, July 1984.
2) Western Range;
   a) Western Range Regulation (WRR) 127-1, June 1993;
   b) Western Space and Missile Center Regulation (WSMCR) 127-1, 15 December 1989;
   c) WSMCR 127-1, 15 May 1985.
3) Both Ranges;

Eastern and Western Range (EWR) 127-1, 31 March 1995.

Applicable safety compliance documents are determined during negotiations with the Range, Lockheed Martin, and the spacecraft contractor at the outset of the mission integration process. Other safety documents that may also apply to the launch site safety interface are:

1) *Radiation Protection Program, 45 Space Wing Instruction 40-201*;
2) *Air Force Manual (AFM) 91-201, Explosives Safety Standard*;
3) *Air Force Regulation (AFR) 127-12, Air Force Occupational Safety, Fire Prevention, and Health Program*;
4) *MIL-STD 1522A, Standard General Requirements for Safe Design and Operation of Pressurized Missile and Space Systems*;
5) *MIL-STD 1576, Electroexplosive Subsystem Safety Requirements and Test Methods for Space Systems*;

6) *Atlas Launch Site Safety Manual*.

At the start of the safety integration process (Sect. 4.3.2), Range Safety documents applicable to spacecraft design and ground-processing operations will be determined. Safety requirements applicable to new designs and ground processing operations will be reviewed and tailored for specific spacecraft and mission applications.

Lockheed Martin's System Safety engineers will evaluate mission-specific spacecraft designs and ground processing operations and provide guidance for successful completion of the Range review and approval process. Should areas of noncompliance be identified, we will evaluate each area and provide guidance for resolution of specific noncompliance items while still meeting the intent of the applicable safety requirements. For commercial programs, Lockheed Martin will act as the spacecraft contractor's liaison for interface activities with the launch site Range Safety Office.

EWR 127-1 Range Safety regulations require three inhibits (dual-fault tolerance) if system failures could result in catastrophic events and two inhibits (single-fault tolerance) if failures could result in critical events. Critical and catastrophic events are defined in EWR 127-1. The Range typically applies the three-inhibit requirement to safety critical electrical systems, including the spacecraft's Category A ordnance circuits, during ground processing operations at the launch site (e.g., encapsulation of spacecraft at the processing facility, transport of encapsulated SC to LC-36 and/or LC-41, mate of encapsulated assembly to the launch vehicle). During final ground processing of Atlas V payloads, the integrated SC and LV stack (in launch configuration) will be transported from the VIF to the LC-41 pad. When Category A spacecraft circuits use the Atlas V launch vehicle's ordnance controller during transport from VIF to pad, the three-inhibit requirement is satisfied. If the spacecraft's Category A ordnance circuits will be independent from the launch vehicle, SC customers should review their bus designs and ground operations plans and notify the Atlas program if SC systems do not provide the required fault tolerance. As stated above, Lockheed Martin will then assess mission-specific designs, evaluate hazard controls, and work with the Range to develop and implement "meets-intent" resolutions.

For each spacecraft and mission, compliance with applicable 127-1 regulations (as tailored) will be addressed in the mission-specific safety submittals defined in Section 4.3.2.

4.3.2 Safety Integration Process

The process used by the Atlas program to facilitate Range and System Safety coordination and receive Range approval and/or permission to launch is shown in Figure 4.3.2-1. This figure identifies responsibilities of the spacecraft contractor, Lockheed Martin, and the Range. Timelines identified in this process are typical and may vary to accommodate mission-specific needs.

For each mission integration effort, Lockheed Martin will provide qualified engineers to assist the spacecraft contractor during the Range review and approval process. The Atlas program obtains all Range safety and system safety approvals. The following paragraphs summarize our safety integration process and define safety data to be developed by the SC customer during implementation of this process. Refer to Appendix C, Section C.3.6, for additional information on SC data requirements.

Mission Orientation—Soon after contract award, Lockheed Martin and the spacecraft contractor will introduce a new system or mission to the Range during a mission orientation meeting at the Range Safety Office. Figure 4.3.2-1, Block A, shows basic elements of this orientation. The orientation provides a general overview of the mission and provides a forum for coordination of mission-specific requirements, schedules, and data submittals. Mission-peculiar designs and operational issues are reviewed so agreements can be established during the early phase of mission integration. Range Safety requirements that will be imposed on spacecraft designs and ground-processing operations are identified.

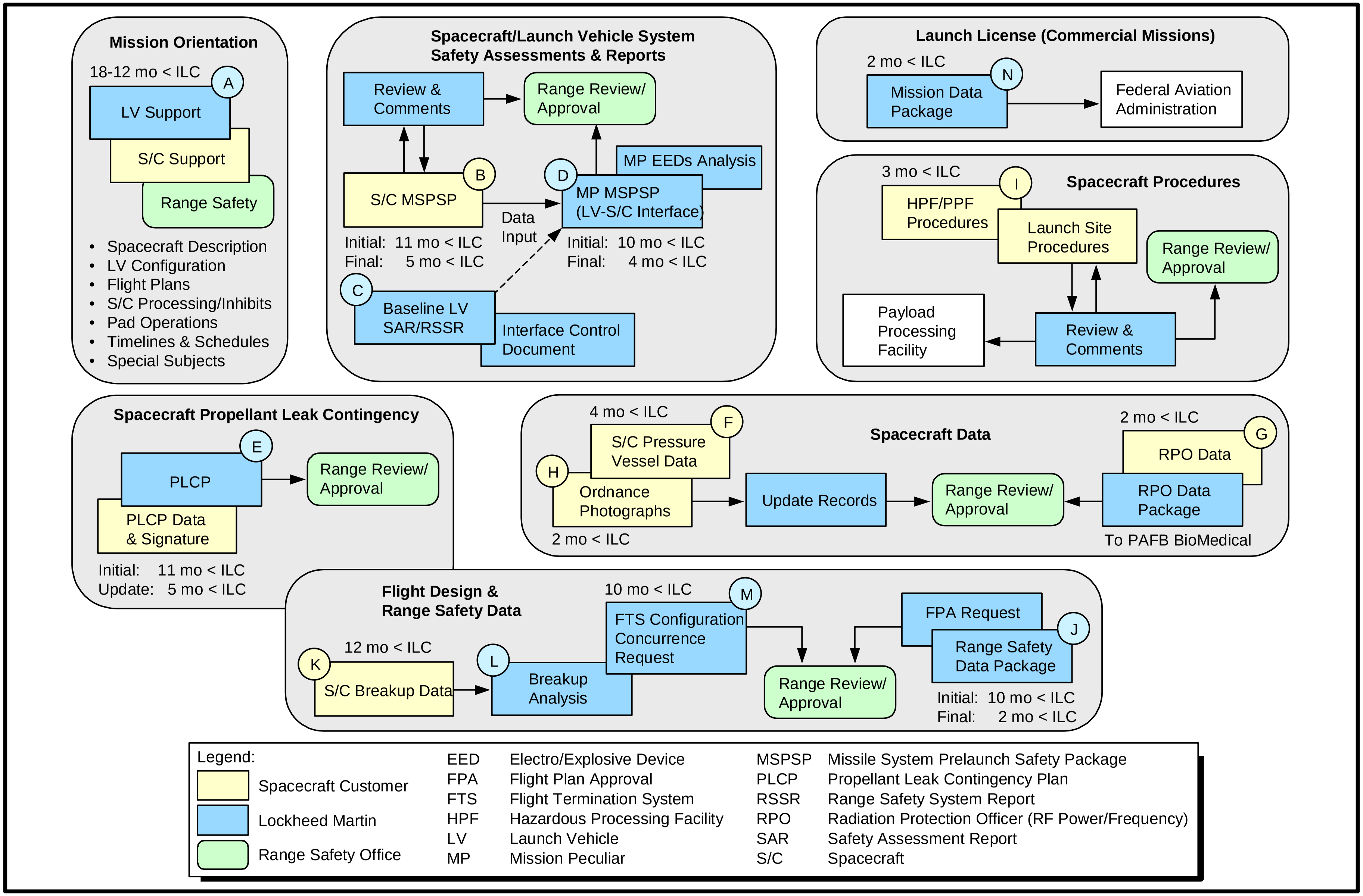


Figure 4.3.2-1 Atlas Safety Integration Process

For follow-on missions, a formal meeting is generally not necessary. Lockheed Martin will develop and submit a mission orientation letter to coordinate mission-specific requirements, schedules, and data submittals. The spacecraft contractor will provide inputs to the mission orientation letter.

Spacecraft and Launch Vehicle Safety Assessments—Mission-specific spacecraft designs and ground processing operations are documented in the Missile System Prelaunch Safety Package (MSPSP) (Fig. 4.3.2-1, Block B). The spacecraft contractor develops the spacecraft MSPSP to describe the spacecraft, document potential hazards associated with ground processing operations at the Range (e.g., pressure systems, ordnance control systems, toxic materials, SC access requirements, RF testing, etc.), and define the means by which each hazard is eliminated or controlled to an acceptable level. Range Safety regulations provide details on the format and contents of the MSPSP.

The initial SC MSPSP is typically submitted to Lockheed Martin approximately 11 months before initial launch capability (ILC). The Atlas program will review the document, provide comments if necessary, and forward the document with comments to the Range for formal review and comment. The Atlas program will then forward SC MSPSP review comments to the spacecraft contractor for incorporation into the final submittal of the MSPSP. The final SC MSPSP is typically submitted to Lockheed Martin about 5 months before scheduled ILC.

Lockheed Martin will combine data from the spacecraft MSPSP with data from existing baseline Atlas launch vehicle safety reports (Fig. 4.3.2-1, Block C) and the mission-specific interface control document (ICD) to perform and document a safety assessment of the launch vehicle-spacecraft interface. Results of this assessment will be delivered to the Range as the mission-unique LV MSPSP (Fig. 4.3.2-1, Block D).

Spacecraft Propellant Leak Contingency—Based on data supplied by the spacecraft contractor (e.g., hardware locations, access requirements, ground support equipment), Lockheed Martin will develop the spacecraft Propellant Leak Contingency Plan (PLCP) (Fig. 4.3.2-1, Block E). The PLCP provides a top-level plan for offload of SC propellants should leakage occur during ground processing operations at the launch pad. The PLCP requires development of detailed procedures by the spacecraft contractor to implement offload operations.

Spacecraft Data—The spacecraft contractor will provide pressure vessel qualification and acceptance test data to the Range (through the Atlas program) for review and acceptance. These data are shown in Figure 4.3.2-1, Block F.

The spacecraft contractor will also submit data specifying the type and intensity of RF radiation that the spacecraft will transmit during ground testing, processing, and launch at the Range. Lockheed Martin will forward these data to the Radiation Protection officer (RPO) for review and approval of RF-related operations to be performed at the launch site. RPO data are shown in Figure 4.3.2-1, Block G. Refer to Appendix C, Section C.3.6.3, for a description of data required by the Range RPO.

The Range requires photographs showing locations of ordnance items installed on the spacecraft. These data are shown in Figure 4.3.2-1, Block H. SC ordnance photographs may be submitted to the Range through the Atlas program, or the spacecraft contractor may submit ordnance photographs directly to the Range Safety Office. If the spacecraft contractor selects the direct submittal option, the Atlas program requires notification that photographs have been delivered. A follow-up meeting between the Range and the SC contractor is typically required to review ordnance data.

Spacecraft Procedures—Through the Atlas program, the spacecraft contractor will submit onsite-processing procedures (payload processing facility [PPF] procedures and launch pad procedures) (Fig. 4.3.2-1, Block I) to the operator of the payload processing facility (e.g., Astrotech, NASA, or Air Force) and the Range for review and approval. As indicated in Section 4.3.1, PPF procedures must comply with the applicable processing facility's safety policy. Procedures to be implemented at the launch pad will comply with applicable Lockheed Martin and Range Safety regulations.

Flight Design and Range Safety Data—Lockheed Martin's flight design group will develop a Range Safety data package (Fig. 4.3.2-1, Block J) that describes the basic spacecraft configuration, the preliminary flight profile, and the time of launch. The Atlas program will submit the preliminary (initial) package to the Range approximately 10 months before ILC. The initial data package will include a flight plan approval (FPA) request to receive preliminary approval to fly the mission on the Range, as designed. Approximately 2 months before ILC, the Atlas program will submit the final Range Safety data package with a request for final FPA. Final FPA is usually received from the Range approximately 7 days before ILC.

To support development of the Range Safety data package, the spacecraft contractor will provide spacecraft breakup data (Fig. 4.3.2-1, Block K) to Lockheed Martin. The Atlas program will use these breakup data to perform a breakup analysis on the spacecraft under expected mission conditions (Fig. 4.3.2-1, Block L). Refer to Appendix C, Section C.3.6.4, for additional information on spacecraft breakup data and analysis.

Based on results of the breakup analysis, the Atlas program will submit an FTS configuration concurrence request to the Range (Fig. 4.3.2-1, Block M). The purpose of this concurrence request is to obtain an agreement with the Range regarding requirements for a designated spacecraft destruct capability. Because there are no appreciable and/or additional public safety hazards with typical communications satellites, Lockheed Martin typically pursues FTS concurrence without a separate spacecraft destruct system.

Launch License (Commercial Missions)—For commercial missions, the Atlas program maintains a launch license from the Federal Aviation Authority (FAA). The Atlas launch license requires periodic updates to address each commercial mission. Lockheed Martin will develop a mission-specific addendum to the baseline license for each commercial flight and submit this data package (Block N, Fig. 4.3.2-1) to the FAA. Spacecraft information included in the FAA data package will include MSPSP approval status and overviews of hazardous spacecraft commodities (propellants, pressure systems, batteries, etc).

5.0 MISSION INTEGRATION AND MANAGEMENT

5.1 INTEGRATION MANAGEMENT

Clear communication between spacecraft and launch vehicle contractors is vital to mission success accomplishment. Procedures and interfaces have been established to delineate areas of responsibility and authority.

The mission integration and management process defined in this section has been successfully used on all recent commercial and government Atlas missions. These identified processes and interfaces have enabled mission integration in as little as 4 months, with 12 months being a more typical schedule for commercial missions.

In the interest of providing our customers with the highest probability of an on-time launch, International Launch Services (ILS) has introduced the “Mutual Backup” concept. This contractual offering permits Atlas customers to purchase a “backup” option on the Proton launch vehicle and Proton customers a “backup” on Atlas. For the case of Atlas backing up Proton, Atlas mission integration tasks will be performed such that the Proton customer may exercise their backup option and launch on Atlas in the event the Proton vehicle experiences technical or schedule difficulties. Contact ILS for further information on “Mutual Backup.”

As an additional information resource, Appendix C of this document details the preferred approach and format for the submission of data required for mission integration. When necessary, deviations from these specified practices can be accommodated.

5.1.1 Launch Vehicle Responsibilities

Lockheed Martin is responsible for Atlas design, integration, checkout, and launch. This work is performed primarily at the Lockheed Martin Astronautics Operations (LMAO) Waterton plant in Denver, Colorado (Fig. 5.1.1-1). Additional facilities in San Diego, California, and Harlingen, Texas,

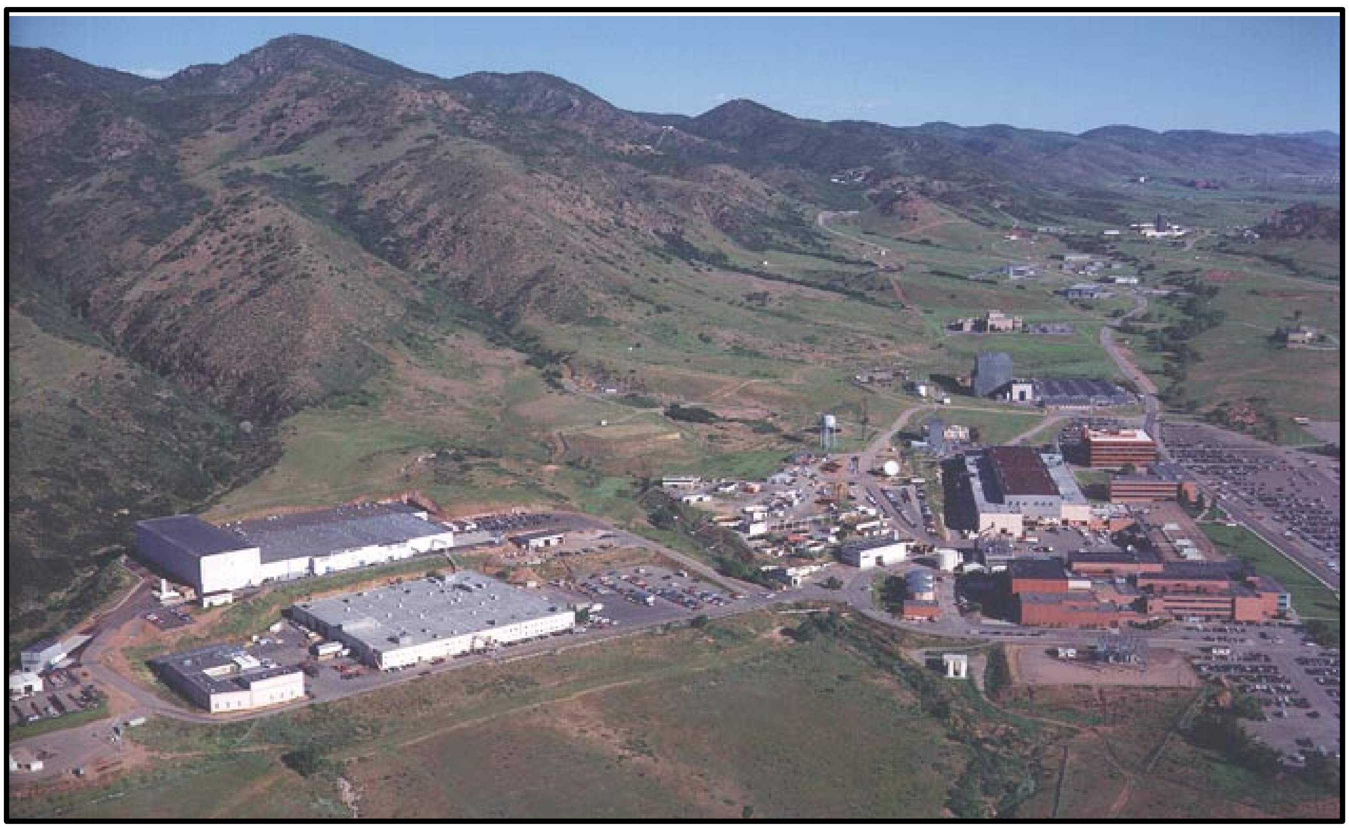


Figure 5.1.1-1 Lockheed Martin Facilities in Denver, Colorado

support hardware manufactures with major tank welding and structural assembly. Major subcontractors include Pratt & Whitney (upper-stage main engines and Atlas III/V booster engines), Honeywell (inertial navigation unit [INU]), Rockwell International-Rocketdyne (Atlas IIAS booster engines), Contraves (5-m payload fairings), Thiokol (solid rocket boosters [SRB] for Atlas IIAS) and Aerojet (solid rocket boosters for Atlas V). As the spacecraft-to-launch vehicle integrating contractor, Lockheed Martin is responsible for spacecraft integration, including electrical, mechanical, environmental, and electromagnetic compatibilities; guidance system integration; mission analysis; software design; Range Safety documentation and support; launch site processing and coordination.

5.1.2 Spacecraft Responsibilities

Since each spacecraft mission has unique requirements, interested Atlas customers are encouraged to discuss their particular needs with Lockheed Martin. Appendix C, Spacecraft Data Requirements, can be used as a guide to initiating dialog. Shaded items in Appendix C should be used as the basis for the first meeting between Lockheed Martin and the customer to assist in determining spacecraft and launch vehicle compatibility.

Customers are encouraged to contact Lockheed Martin to verify the latest launch information, including:

1) Hardware status and plans,
2) Launch and launch complex schedules,
3) Hardware production schedule and costs.

5.1.3 Integration Organization

For all Atlas missions, ILS management assigns a program director. The program director is responsible for overall management of particular customer activities at Lockheed Martin facilities and the launch site. This person is the primary interface with the customer for all technical and launch vehicle and satellite interface and integration matters. The program director works within the Lockheed Martin management structure to establish priorities, schedule assets, and complete arrangements and understandings among program participants and Lockheed Martin management. Formal internal processes provide the formal direction and requirements flowdown for implementation by the Atlas program. Lockheed Martin’s approach to integration management is through establishment of a formal interface control document (ICD) agreement and formal configuration control after ICD signature. Existing ICDs may be adapted to reduce development time. Coordination of tasks necessary to develop and maintain the ICD is accomplished through management and technical working groups, as required.

The LMAO Atlas Mission Integration organization assigns a program manager for each Atlas mission. The program manager coordinates Astronautics Operations’ resources to ensure timely delivery of Atlas vehicle hardware through the production process and is responsible for coordinating the engineering integration of the spacecraft with the Atlas launch vehicle.

The ILS organization is aligned to provide low-risk launch services. ILS responds to a customer order by arranging services from several organizations such as Astrotech, U.S. Air Force (USAF), and LMAO. Astrotech spacecraft integration facilities are contracted for most Atlas launches, with additional support from other government facilities when required. LMAO is the subcontractor responsible for Atlas production, launch, and mission-peculiar integration processing. ILS has contracts with the U.S. government for use of Atlas launch complexes and payload integration facilities and with the USAF for range and launch site services.

To provide maximum efficiency in managing the many launch site operations, a launch site-based launch site payload integrator is assigned to each mission. The launch site payload integrator represents the program director during development, integration, and installation of all spacecraft-peculiar items at the launch site and at arrival of the spacecraft. This program organization concept has been used successfully for the Atlas program.

5.1.4 Integration Program Reviews

During integration, reviews focus management attention on significant milestones during the launch system design and launch preparation process to support major program milestones. As with working group meetings, these reviews can be tailored to customer requirements; however, for a first-of-a-kind launch, they may include a leading-edge design review, a mission-peculiar design review (MPDR), and a launch readiness review (LRR). For typical communications satellite launches, only one design review is required. Additionally, program management reviews (PMR) are convened periodically to provide status.

Lockheed Martin schedules launch system meetings and reviews according to the mission integration schedule. Spacecraft customer representatives may have access to Lockheed Martin facilities to attend reviews, meetings, and related activities. The incremental launch preparation review process provides management with an assessment of the readiness of the launch vehicle systems to proceed with launch preparation, and assurance that all mission functional and support elements are ready to support Range Safety, countdown, and launch activities. The program-level review process is the primary mechanism for providing management with the visibility required to establish maximum confidence in mission success achievement for Atlas launch systems.

Technical working group meetings are convened at the Lockheed Martin's discretion during the mission integration effort to define technical interfaces and resolve technical issues. At a minimum, technical meetings include one ground operations working group (GOWG) and ground operations readiness review (GORR).

5.1.4.1 Mission-Peculiar Design Reviews—As Lockheed Martin has significant experience with most communications satellite contractors, we have found that in most cases one MPDR can successfully meet goals and requirements that were, in the past, met by both a preliminary design review (PDR) and a critical design review (CDR). Approximately midway through the mission integration process, the MPDR is conducted to ensure that customer requirements have been correctly and completely identified and that integration analyses and designs meet these requirements. Lockheed Martin prepares and presents the review with participation from the spacecraft contractor, launch services customer, and launch vehicle management.

For missions that require unique spacecraft interfaces with the launch vehicle or the launch vehicle mission is unlike other previously flown Atlas missions, two design reviews may be proposed by Lockheed Martin after discussions with the launch services customer and spacecraft contractor.

The MPDR includes the following subjects:

1) Requirements updated since the mission integration kickoff meeting;
2) Mission design, performance capability, and margin results;
3) Coupled loads analysis status;
4) Integrated thermal analysis results;
5) Spacecraft separation analysis preliminary results;
6) Guidance system accuracy analysis preliminary results;
7) Mission-peculiar flight software and parameter implementation and design updates (if required);
8) Radio frequency (RF) compatibility analysis, electromagnetic interference/electromagnetic compatibility (EMI/EMC) analysis, and link margin analysis preliminary results;
9) Mission-peculiar electrical and mechanical interface hardware design;
10) Launch site implementation of unique requirements.

5.1.4.2 Launch Readiness Review—This review, conducted approximately 2 days before launch, provides a final prelaunch assessment of the integrated spacecraft/launch vehicle system and launch facility readiness. The LRR provides the forum for final assessment of all launch system preparations and contractors' individual certifications of launch readiness. The purpose of the LRR is to ensure that spacecraft systems, launch vehicle systems, facilities and aerospace ground equipment (AGE), and all supporting organizations are ready and committed to support the final launch preparations, countdown, and launch. Lockheed Martin management representatives from Denver and the launch site participate in the LRR, along with representatives from spacecraft customer organizations. Representatives from each key organization summarize their preparations and rationale for their readiness to proceed with the final launch preparations and countdown. The meeting concludes with a poll of each organization to express their readiness and commitment to launch.

5.1.4.3 Ground Operations Working Group—Lockheed Martin convenes one GOWG meeting at the launch site during the early part of the standard integration cycle. The GOWG includes representatives of the spacecraft customer, contractors, and all launch site organizations involved in operations. The GOWG provides a forum for coordinating launch site activities and resolving operational issues and concerns. It is usually co-chaired by the launch site payload integrator and the spacecraft customer launch operations lead. At the GOWG, the following items are coordinated: the ground operations activities flow, operational timeline modifications for mission-peculiar spacecraft operational considerations, ICD interface requirements definition for launch site facilities and ground support equipment, hazardous operations with the Range, and ground test requirements. The GOWG reviews and approves the system documentation, operational timelines, and operational procedures required to process, test, and launch the integrated Atlas/spacecraft vehicle. For spacecraft customers with nearly identical spacecraft (follow-on) and who are very familiar with Atlas site processing, the GOWG may be waived.

5.1.4.4 Ground Operations Readiness Review—Lockheed Martin conducts one GORR just before the arrival of the spacecraft at the launch site spacecraft processing facility. The meeting objectives are to formally kickoff the launch campaign; review the readiness of the facility to receive the spacecraft; and ensure that processing plans, schedules, procedures, and support requirements are coordinated. The technical chairperson is the Atlas Launch Operations chief for payload operations and the launch site payload integrator is responsible for documenting meeting minutes and action items. A poll at the conclusion of the meeting ensures that all participating agencies, including spacecraft customer representatives, concur with the plan to be implemented for the launch campaign.

5.1.5 Integration Control Documentation

5.1.5.1 Program Master Schedule—This top-level schedule is prepared by Lockheed Martin and monitored by the Mission Integration Team. It maintains visibility and control of all major program milestone requirements, including working group meetings, major integrated reviews, design and analysis requirements, and major launch operations tests. It is developed from tasks and schedule requirements identified during initial integration meetings and is used by all participating organizations and working groups to develop and update subtier schedules. The mission integration schedule facilitates a systematic process to manage program activities. The mission integration schedule is used to track and monitor the mission progress to avoid significant schedule issues and possible cost impacts. The mission integration schedule contains sufficient mission details and contract statement of work (SOW) task-level representation to assist the program director and program manager in managing the launch service.

5.1.5.2 Interface Requirements Documents (IRD)—The customer creates the IRD to define technical and functional requirements imposed by the spacecraft on the launch vehicle system. The document contains applicable spacecraft data identified in Appendix C. Information typically includes:

1) Mission Requirements—Including orbit parameters, launch window parameters, separation functions, and any special trajectory requirements, such as thermal maneuvers and separation over a telemetry and tracking ground station;
2) Spacecraft Characteristics—Including physical envelope, mass properties, dynamic characteristics, contamination requirements, acoustic and shock requirements, thermal requirements, and any special safety issues;
3) Mechanical and Electrical Interfaces—Including spacecraft mounting constraints, spacecraft access requirements, umbilical power, command and telemetry, electrical bonding, and EMC requirements;
4) Mechanical and Electrical Requirements for Ground Equipment and Facilities—Including spacecraft handling equipment, checkout and support services, prelaunch and launch environmental requirements, spacecraft gases and propellants, spacecraft RF power, and monitor and control requirements;
5) Test Operations—Including spacecraft integrated testing, countdown operations, and checkout and launch support.

5.1.5.3 Interface Control Document—This document defines spacecraft-to-launch vehicle and launch complex interfaces. All mission-peculiar requirements are documented in the ICD. The ICD is prepared by Lockheed Martin and is under configuration control after formal signoff. The document contains appropriate technical and functional requirements specified in the IRD and any additional requirements developed during the integration process. The ICD supersedes the IRD and is approved with signature by Atlas program management and the launch service customer. Subsequent changes to the mission ICD require agreement of the signing parties. If any conflict or inconsistency exists between the signed mission ICD and the IRD or the contract SOW, the signed mission ICD is given precedence.

ICD technical development is led by the program manager with management decisions coordinated by the program director. The ICD is the top-level interface requirements document between the launch vehicle and the spacecraft. It contains physical, functional, environmental, operational, and performance requirements for the interface and is a contractually binding document. The document establishes how each interface requirement is to be verified to ensure that all interface details have been accomplished in compliance with ICD requirements. It identifies interface verification activities that link the designed, built, and tested interface back to the functional and performance requirement the interface was meant to satisfy.

5.2 MISSION INTEGRATION ANALYSIS

To support a given mission, Lockheed Martin will perform analyses summarized in Table 5.2-1. This table indicates the specific output of analyses to be performed, required spacecraft data, the timing during the integration cycle that the analysis is completed, and the application of analyses to first-of-a-kind and follow-on missions. In this context a follow-on mission is an exact copy of a previous mission, with no change to functional requirements or physical interfaces. For a follow-on mission, all analyses are reassessed to ensure the original analysis is still applicable. Table 5.2-1 represents standard integration analyses. For many missions, Lockheed Martin uses generic versions of these analyses and may not be required to perform a mission-peculiar version.

Figure 5.2-1 is our generic schedule for a typical Atlas mission. The full-scale integration process begins at approximately L-12 months.

Table 5.2-1 Summary of Typical Atlas Mission Integration Analyses

Analysis	S/C Data	Analysis Products	No. of Cycles	Schedule	First-of-a-Kind	Follow-On
1) Coupled Loads	S/C Dynamic Math Model	• S/C Loads • Dynamic Loss of Clearance • Launch Availability • PLF Jettison Evaluation	2	Model Delivery + 4 mo	X	
2) Integrated Thermal	S/C Geometric & Thermal Math Models & Power Dissipation Profile	• S/C Component Temperature • Prelaunch Gas Cond & Set Points	1	Model Delivery + 6 mo	X	
3) PLF Venting	S/C Venting Volume	• Press Profiles • Depressurization Rates	1	S/C Data + 2 mo	X	
4) Critical Clearance	S/C Geometric Model; S/C Dynamic Model	• S/C-to-PLF Loss of Clearance (Dynamic + Static)	2	S/C Model Del + 4 mo	X	
5) PLF Jettison Clearance	S/C Geometric Model	• P/L Clearance Margin During PLF Jettison Event	1	Design Review	X	
6) S/C Separation & Clearance	S/C Mass Properties	• S/C Sep Clearance • S/C Sep Att & Rate & Spin-Up Verification	1	Design Review	X	
7) Postseparation Clearance		• LV-S/C Sep History	1	Design Review	X	
8) Pyro Shock	S/C Interface Definition	• S/C Shock Environment	1	Design Review	X	
9) Acoustics	S/C Geometry Fill Factors	• S/C Acoustics Environment	1	Design Review	X	
10) EMI/EMC	• S/C Radiated Emissions Curve • S/C Radiated Susceptibility Curve • S/C Rec Op & Demise Thresholds • S/C Diplexer Rejection	• Confirmation of Margins • Integrated EMI/EMC Analysis	1	Design Review	X	
11) Contamination	S/C Contamination Limits for Sensitive, Critical, Vertical & Horizontal Surfaces	• Contamination Analysis or Contamination Assessment	1	Design Review	X	
12) RF Link Compatibility & Telemetry Cov (Airborne)	S/C Transmitter & Receiver Characteristics	• Link Margins • Frequency Compatibility • EEDs Susceptibility	1	Design Review	X	
13) RF Link Compatibility & Telemetry Cov (Ground)	S/C Transmitter & Receiver Characteristics	• Link Margins • Identify Required Hardware	1	Design Review	X	
14) Performance	S/C Mass & Mission Requirements Definition	• Config, Perf & Wt Status Report • Performance Margin	3	ATP + 3-mo Dsn Rev Final Targeting	X	X
15) Stability		• Control System Margins • RCS Use	1	L-2 mo	X	
16) Mass Properties		• Mass Properties of LV		Coincident with Perf Reports	X	X

Table 5.2-1 (concl)

Analysis	S/C Data	Analysis Products	No. of Cycles	Schedule	First-of-a-Kind	Follow-On
17) Trajectory Analysis	S/C Mass & Mission Requirements Definition	• LV Ref Trajectory • Performance Margin	1	Design Review	X	X
18) Guidance Analysis	Mission Requirements	• Guidance S/W Algorithms • Mission Targeting Capability & Accuracies	1	Design Review	X	
19) Injection Accuracy	Mission Requirements	• LV System Orbit Injection Accuracy	1	Design Review	X	
20) Launch Window	Window Definition	• Window Durations	1	Design Review	X	X
21) Wind Placard	S/C Mass Properties	• LV Ground & Flight Winds Restrictions	1	Design Review	X	
22) Range Safety	S/C Breakup Data & Propulsion Characteristics	• Trajectory Data & Tapes for Range Approval	2	L-1 yr, Prelim L-7 wk, Final	X	
23) Electrical Compatibility	Electrical Interface Requirements	• End-to-End Circuit Analysis	1	Design Review	X	
24) Postflight		• Eval of Mission Data, LV Perf & Environment	1	L + 60/days	X	X
25) Destruct Sys		• Confirmation of Mtg Range Safety Rqmts	1	Design Review	X	
26) Mission Targeting	Orbit Requirements	• Flight Constants Tapes • Firing Tables	1	L-1 mo	X	X
27) Flight Software	Mission Requirements	• FCS Software	1	L-3 weeks	X	X

The Atlas program has demonstrated the effectiveness of “Class Analysis” within Mission Integration with initiatives such as the standard park orbit for geosynchronous transfer orbit (GTO) missions and generic flight software. The Atlas V program with its robust margins is able to extend the use of class analysis to all applicable mission design processes and analyses. Class analysis mission integration is based on broad envelopes of parameters rather than parameters for a specific mission. Analyses are performed upfront in the development phase of the program to encompass the known variation in individual vehicles and missions. Vehicle-to-vehicle variations and mission-to-mission variations are accounted for when defining boundaries of a class. Class analysis reduces recurring mission integration span time and cost while maintaining mission success by eliminating recurring characterization of the LV and minimizing the need for recurring analyses. The class analysis method is enabled by the robustness and flexibility of the hardware design, software design, and standard payload interface. Standardized designs can use standardized (class) analyses. Recurring analyses are replaced by assessments of the applicability of the previously performed class analysis and replace recurring flight software development with recurring parameter generation. The mission-unique designs that require mission-unique analyses are only those needed to verify unique requirements in the ICD.

The following paragraphs describe integration analyses.

5.2.1 Coupled Loads Analysis (CLA)

During the Atlas program, a set of test-correlated three-dimensional (3-D) analytical launch vehicle models is generated for the mission-peculiar dynamic CLA. Lockheed Martin will perform mission-peculiar analyses where launch vehicle and payload parameters may be affected (Sect. 3.2.1). While not all flight events are analyzed in each load cycle for each vehicle configuration, analyses that are typically performed as part of the CLA are:

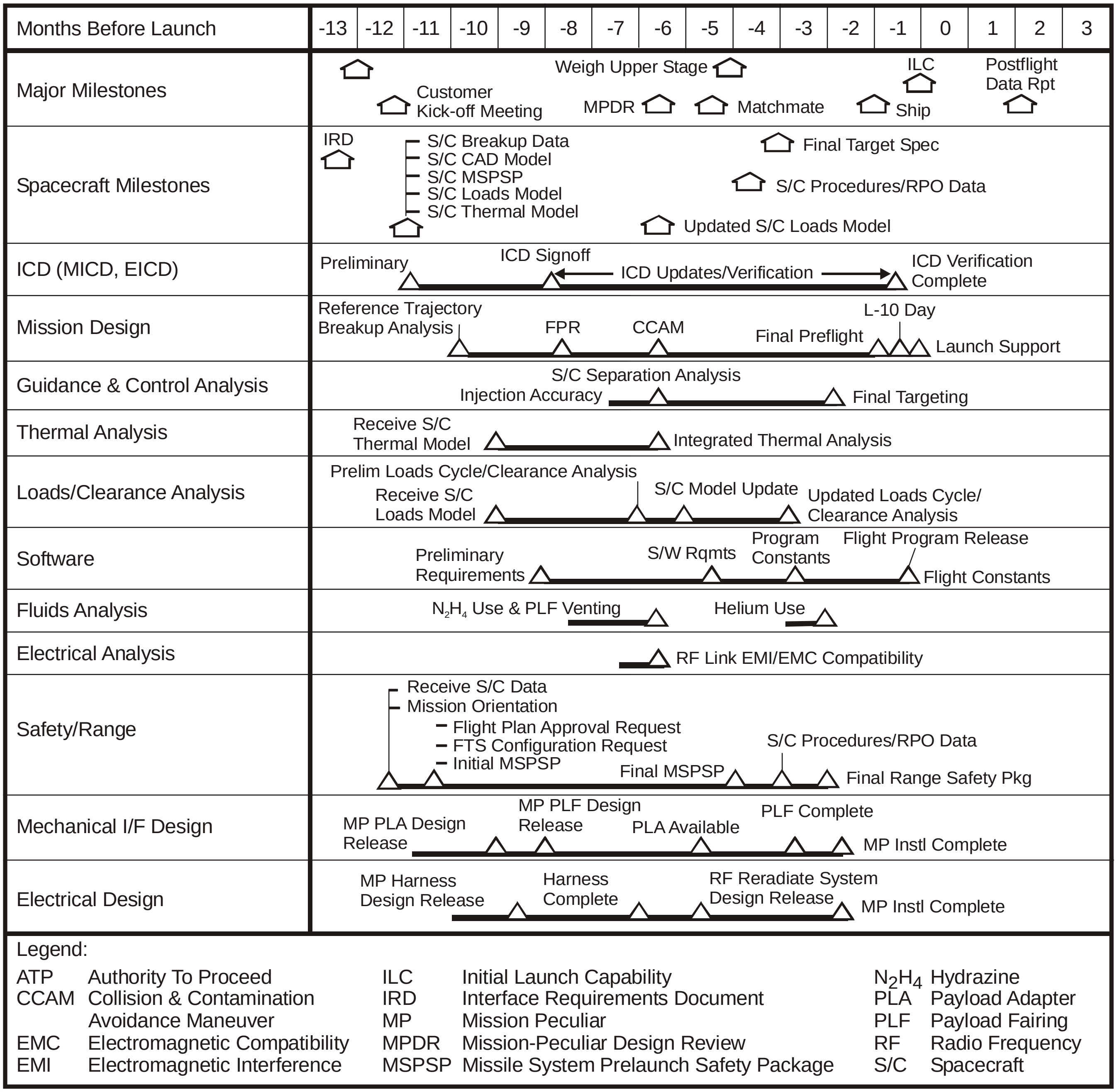


Figure 5.2-1 Twelve-month Generic Mission Integration Schedule

1) Spacecraft loading for critical flight events: liftoff, transonic buffet and gust, maximum dynamic pressure buffet and gust, SRB air-lit ignition, SRB burnout and jettison, booster engine cutoff (BECO), maximum g-loading, booster package jettison (BPJ), booster engine thrust coupling (BETC), Centaur main engine cutoff (MECO), and Centaur longitudinal event (CLE);
2) Flight wind launch availability assessment;
3) Payload fairing (PLF) jettison evaluation applicable to a mission spacecraft;
4) Spacecraft loss of clearance evaluation for all critical flight events.

Analysis of all events uses state-of-the-art finite element models of the booster coupled with a customer-supplied dynamic math model of the spacecraft. Appendix C of this document describes the type and format of the spacecraft dynamic model.

5.2.2 Integrated Thermal Analysis (ITA)—Preflight and Flight

Lockheed Martin performs an integrated launch vehicle-spacecraft analysis of thermal environments imposed on the spacecraft under prelaunch conditions and for flight mission phases up to spacecraft separation. The ITA is performed with customer-supplied spacecraft geometric and thermal math models and a detailed spacecraft power dissipation timeline. Results are provided to the customer for evaluation and can be used to design thermal interfaces and mission operations to maintain predicted spacecraft temperatures within allowable limits.

In addition to the ITA, Lockheed Martin performs PLF aeroheating analyses, PLF gas conditioning analyses, and free molecular heating analyses to verify compliance with customer ICD thermal requirements and thermal requirements derived from the ITA.

Thermal analyses ensure that vehicle design is compatible with and has adequate margins over proposed spacecraft thermal constraints. Analyses include assessment of vehicle aeroheating, PLF surface temperature ranges, maximum and minimum prelaunch air conditioning temperatures and velocities, and spacecraft-to-Centaur interface temperature ranges.

Prelaunch payload gas velocity analyses verify that impingement velocities are compatible with the defined spacecraft. A worst-case analysis is performed (using the maximum air conditioning supply rate) to determine flow conditions inside the PLF.

The gas conditioning thermal analysis predicts air conditioning gas temperature variations along the spacecraft during prelaunch operations. The analysis is performed by combining the extremes of air conditioning inlet temperature and flow rate conditions with spacecraft power dissipation levels.

PLF internal surface temperature ranges are predicted by analyzing flight aerodynamic heating.

PLF jettison time is selected to meet the spacecraft free molecular heating constraint. Atlas missions ensure a benign spacecraft thermal environment by selecting jettison time based on a flight program calculation of 3-sigma maximum qV during flight.

5.2.3 PLF Venting Analysis (Ascent Phase)

A venting analysis is performed on the PLF to determine mission-peculiar pressure profiles in the payload compartment during launch vehicle ascent. Existing models that have been validated with flight data are used for this analysis. The analysis incorporates the customer-provided spacecraft venting configuration and any mission-specific PLF requirements (e.g., thermal shields). Analysis outputs provided to the customer include PLF pressure profiles and depressurization rates as a function of flight time.

5.2.4 Critical Clearance Analysis (Loss of Clearance)

The static payload envelope defines the usable volume for a spacecraft. This envelope represents the maximum allowable spacecraft static dimensions (including manufacturing tolerances) relative to the spacecraft and payload adapter interface. For clearances between the spacecraft and payload fairing, the primary clearance concerns are for dynamic deflections of the spacecraft and payload fairing and the resulting relative loss of clearance between these components. A critical clearance analysis is performed to verify that these deflections do not result in contact between the spacecraft and launch vehicle hardware. This analysis considers spacecraft and payload fairing static tolerances and misalignments, dynamic deflections, and out-of-tolerance conditions, and ensures that a minimum 25-mm (1-in.) clearance between the spacecraft and the payload fairing is maintained. During this analysis, dynamic deflections are calculated for ground handling, flight (from the coupled dynamic loads analysis, Sect. 5.2.1), and payload fairing jettison conditions. Clearance layouts and analyses are performed for each spacecraft configuration, and if necessary, critical clearance locations are measured after the spacecraft is encapsulated inside the fairing to ensure positive clearance during flight.

5.2.5 PLF Jettison and Loss of Clearance Analyses

Verification of payload clearance during PLF jettison is performed using effects of thermal preload, disconnect forces, shear pin forces, actuator forces, and dynamic response in a 3-D nonlinear analysis.

5.2.6 Spacecraft Separation Analysis

Extensive Monte Carlo analysis of preseparation dynamics, using a 3-degree-of-freedom (DOF) simulation of the vehicle and attitude control system, demonstrates compliance with all spacecraft attitude pointing and angular rate and spin rate requirements under nominal and 3-sigma dispersions.

A two-body 6-DOF Monte Carlo simulation of the Centaur and spacecraft separation event is performed using finalized spacecraft mass properties to verify the Centaur will not recontact the spacecraft after separation system release. This analysis demonstrates minimum relative separation velocity, ensuring that adequate separation distance is achieved before initiating postseparation Centaur maneuvers.

5.2.7 Spacecraft Postseparation Clearance Analysis

After the spacecraft has separated from the Centaur vehicle, Centaur performs a collision and contamination avoidance maneuver (CCAM). The Centaur reaction control system uses 12 hydrazine thrusters. Four thrusters are dedicated to propellant settling (axial) control and eight are allocated to roll, pitch, and yaw control. Thrusters are located on the aft bulkhead of the liquid oxygen (LO_2) tank inboard of the 3.0-m (10-ft) tank diameter. Before spacecraft separation, this location precludes a direct line of impingement to the spacecraft. In addition, thrust directions are either 90° or 180° away from the spacecraft.

The CCAM is designed to positively preclude physical recontact with the spacecraft and eliminate the possibility of significant impingement of Centaur effluents on the spacecraft. The CCAM consists of two or three attitude maneuvers, combined with axial thrust from the reaction control system (RCS) settling motors and blowdown of the Centaur tanks. For Atlas II and III vehicles, shortly after spacecraft separation, the Centaur turns 4° from the separation attitude. The Centaur then typically turns 50° and activates the settling motors to impart a ΔV to move the Centaur a significant distance from the spacecraft. This maneuver minimizes any plume flux to the spacecraft. The final maneuver turns the Centaur normal to the flight plane. In this attitude, the tank blowdown is executed at approximately 1.8 km (1 nmi) from the spacecraft. This CCAM sequence ensures adequate in-plane and out-of-plane separation between the Centaur and the spacecraft and minimizes the RCS motor plume flux at the spacecraft.

The Atlas V CCAM is different only in that it bypasses the 4° turn and goes directly to 50° as its first maneuver. The remaining sequence is similar to the Atlas II and III vehicles.

5.2.8 Pyroshock Analysis

The spacecraft pyroshock environment is maximum for the spacecraft separation event. PLF separation and Atlas/Centaur separation are also significant events, but the distances of the shock sources from the spacecraft/Centaur interface make them less severe for the spacecraft than activation of the payload separation system (PSS). Verification of this environment has been accomplished by ground PSS testing of our existing separation systems.

5.2.9 Acoustic Analysis

Analysis of the acoustic environment of the payload compartment includes effects of noise reduction of the PLF and payload fill factors. Verification includes flight measurements taken from several Atlas/Centaur flights and ground acoustic testing of representative PLF/payload configurations.

5.2.10 Electromagnetic Interference/Electromagnetic Compatibility Analysis

Lockheed Martin maintains an EMI/EMC plan to ensure compatibility between all avionics equipment. This plan covers requirements for bonding, lightning protection, wire routing and shielding, and procedures. Lockheed Martin analyzes intentional and unintentional RF sources to confirm 6-decibel (dB) margins with respect to all general EMI/EMC requirements. In addition, electroexplosive device (EED) RF susceptibility analyses are performed to Range requirements for both the launch vehicle and spacecraft. The spacecraft analysis is performed by the spacecraft manufacturer and reviewed by Lockheed Martin. The presence of an RF environment will affect safety margins of EEDs. This analysis is intended to confirm a minimum 20-dB margin with respect to the direct current (dc) EED no-fire power level. The purpose of the EED susceptibility analysis is to demonstrate that safety margins of each EED are maintained when exposed to the flight vehicle and site sources RF environment. Comprehensive reports are published describing requirements and results of these analyses.

5.2.11 Contamination Analysis

Control of contamination (to meet analysis assumptions) is discussed in Section 3.1.3 for ground operations and in Section 3.2.7 for flight operations.

Lockheed Martin provides spacecraft with an assessment of contamination contributions from Atlas launch vehicle sources, as required. Starting from PLF encapsulation of the spacecraft through CCAM, contamination sources are identified and analyzed. This provides a qualitative assessment of the factors affecting spacecraft contamination to allow the spacecraft customer to approximate final on-orbit contamination budgets. A more detailed mission-peculiar analysis can be provided to the spacecraft customer if mission-unique deposition requirements are specified in the ICD.

5.2.12 RF Link Compatibility and Telemetry Coverage Analysis (Airborne)

Lockheed Martin conducts an airborne link analysis on all RF links between ground stations and the Atlas/Centaur vehicle to determine whether the signal strength between the RF system on the launch vehicle and the RF system at the receiving station meets mission requirements. Systems analyzed are: the S-band telemetry system, the active C-band vehicle tracking system, and the flight termination system. Lockheed Martin uses a program that considers airborne and ground station equipment characteristics, vehicle position, and attitude. This analysis includes maximizing link margins with the receiving ground stations and the Tracking and Data Relay Satellite System (TDRSS) when the TDRSS-compatible transmitter is used on Centaur. A comprehensive report is published describing link requirements and results.

Lockheed Martin conducts an RF compatibility analysis between all active airborne RF transmitters and receivers to ensure proper function of the integrated system. Transmit frequencies and their harmonics are analyzed for potential interference to each active receiver. If interference exists, a worst-case power-level analysis is performed to determine what effect the interference frequency has on the receiver’s performance. In addition, strong site sources, such as C-band site radar, are also analyzed. The spacecraft contractor provides details of active transmitters and receivers for this analysis. A comprehensive report is published describing analysis requirements and results.

5.2.13 RF Link Compatibility and Telemetry Coverage Analysis (Ground)

For customers who require communication with their spacecraft during prelaunch activities, Lockheed Martin conducts a ground link analysis on spacecraft RF systems to ensure that a positive link exists between the spacecraft and the spacecraft checkout equipment to checkout the spacecraft telemetry and command system. The RF reradiate system will provide sufficient margin to minimize effects of deviations or fluctuations in RF power and will provide consistent system performance to ensure positive link margin during the required time periods. Information about spacecraft requirements is in the ICD. A technical report is published describing link requirements and implementation of the link system.

5.2.14 Performance Analysis

The capability of Atlas to place the spacecraft into the required orbit(s) is evaluated through our trajectory simulation tools. Vehicle performance capability is provided through our configuration, performance, and weight status report. This report is tailored to accommodate needs of specific missions.

The status report shows the current launch vehicle propellant margin and flight performance reserve (FPR) for the given mission and spacecraft mass. A comprehensive list of the vehicle configuration status, mission-peculiar ground rules and inputs, and vehicle masses for performance analysis is included. The report also provides the more commonly used payload partial derivatives (tradeoff coefficients) with respect to the major vehicle variables (e.g., stage inert weights, propellant loads, stage propulsion parameters). The detailed trajectory simulation used for the performance assessment is provided as an appendix to the report.

5.2.15 Stability and Control Analysis

Linear stability analysis, primarily frequency response and root-locus techniques, and nonlinear time-varying 6-DOF simulation are performed to determine Atlas and Centaur autopilot configurations; establish gain and filter requirements for satisfactory rigid body, slosh, and elastic mode stability margins; verify vehicle and launch stand clearances; and demonstrate Centaur RCS maneuver and attitude hold capabilities. Uncertainties affecting control system stability and performance are evaluated through a rigorous stability dispersion analysis. Tolerances are applied to vehicle and environmental parameters and analyzed using frequency response and nonlinear simulation methods, ensuring that the Atlas autopilot maintains robust stability throughout the defined mission. Correlation of simulation results with previous postflight data have confirmed the adequacy of these techniques.

5.2.16 Mass Properties Analysis

Lockheed Martin performs mass properties analysis, reporting, and verification to support performance evaluation, structural loads analysis, control system software configuration development, ground operations planning, airborne shipping requirements, and customer reporting requirements.

5.2.17 Trajectory Analysis and Design

The Lockheed Martin trajectory design process ensures that all spacecraft, launch vehicle, and range-imposed environmental and operational constraints are met during flight, while simultaneously providing performance-efficient flight designs. This process typically provides propellant margin (PM) above required performance reserves.

The trajectory design and simulation process provides the vehicle performance capability for the mission. It provides the basis, by simulation of dispersed vehicle and environmental parameters, for analyses of FPR and injection accuracy. Telemetry coverage assessment, RF link margins, PLF venting, and in-flight thermal analyses also rely on the reference mission design. The trajectory design is documented in the status report (Sect. 5.2.14). Detailed insight into the tradeoffs used for the trajectory design is provided in the trajectory design report.

Our trajectory analysis tools incorporate detailed propulsion, mass properties, aerodynamic, and steering control modeling, as well as oblate Earth and gravity capability, selectable atmospheric models, and other selectable routines, such as Sun position and tracker locations, to obtain output for these areas when they are of interest.

These simulation tools interface directly with actual flight computer software. This feature bypasses the need to have engineering equivalents of flight software. Another powerful feature is compatibility with 6-DOF modeling of the vehicle, which will facilitate key dynamic analyses for our vehicle family. Other features include significant flexibility in variables used for optimization, output, and simulation interrupts.

5.2.18 Guidance Analysis

Analyses are performed to demonstrate that spacecraft guidance and navigation requirements are satisfied. Analyses include targeting, standard vehicle dispersions, extreme vehicle dispersions, and guidance accuracy. The targeting analysis verifies that the guidance program achieves all mission requirements across launch windows throughout the launch opportunity. Standard vehicle dispersion analysis demonstrates that guidance algorithms are insensitive to 3-sigma vehicle dispersions by showing that the guidance program compensates for these dispersions while minimizing orbit insertion errors. Extreme vehicle dispersions (e.g., 10 sigma) and failure modes are selected to stress the guidance program and demonstrate that the guidance software capabilities far exceed the vehicle capabilities.

5.2.19 Injection Accuracy Analysis

The guidance accuracy analysis combines vehicle dispersions and guidance hardware and software error models to evaluate total guidance system injection accuracy. Hardware errors model off-nominal effects of guidance system gyros and accelerometers. Software errors include INU computation errors and vehicle dispersion effects. Positive and negative dispersions of more than 30 independent vehicle and atmospheric parameters that perturb Atlas and Centaur performance are simulated. The accuracy analysis includes sensor noise, effects of vehicle prelaunch twist and sway on guidance system alignment during gyro compassing, and the covariance error analysis of the guidance hardware.

5.2.20 Launch Window Analysis

Launch window analyses are performed to define the open and close of mission-specific launch windows that satisfy mission-specific requirements on each launch day within the launch period. The Atlas/Centaur launch vehicle can accommodate launch windows, any time of day, any day of the year within performance capability constraints for a given mission design. Customers are requested to provide opening and closing times for the maximum launch window the spacecraft is capable of supporting. If the launch windows are several hours long or multiple windows in a single day, then a span within the total launch opportunity will be jointly chosen by Lockheed Martin and the customer. This decision can be made as late as a few days before launch. The selected span will be chosen based on operational considerations, such as preferred time of day or predicted weather.

Some missions may have more complicated window constraints requiring analysis by Lockheed Martin. For example, launch system performance capability constrains windows for missions that require precise control of the right ascension of the ascending node. That control is achieved by varying the trajectory as a function of launch time. We have successfully analyzed a variety of window constraints for past missions, and we are prepared to accommodate required window constraints for future missions.

Any launch window duration can be accommodated. However, a window of 30 minutes or more is recommended. Shorter windows increase the risk of a launch delay if exceeded due to weather or technical problem resolution. Windows longer than 4 hours for Atlas IIAS/III (3 hours for Atlas V) may be limited by liquid oxygen supplies or crew rest limits.

5.2.21 Wind Placard Analysis (Prelaunch, Flight)

Wind tunnel tests of the Atlas IIA and IIAS configurations have been performed to determine loading for ground and flight wind conditions and have been further applied to the Atlas IIIA and IIIB configurations. This information, combined with launch site wind statistics, is used to determine the wind placards and subsequent launch availability for any given launch date. Atlas IIAS/III vehicle configurations provide at least 85% annual launch availability. Atlas V vehicle configurations annual launch availability is being analyzed and the goal is at least 85% annual launch availability.

5.2.22 Range Safety Analyses

Lockheed Martin performs flight analyses required to comply with Eastern/Western Range regulations for both the request for preliminary flight plan approval and the more detailed submittal for final flight plan approval. These submittals occur approximately 1 year before launch for the initial request and approximately 45 days before launch for the second. Reports and magnetic tapes of required information are provided to the Range agency in required formats and include nominal and dispersed trajectories and impact locations of jettisoned hardware. During spacecraft integration, a Range support plan is prepared documenting our planned coverage.

5.2.23 End-to-End Electrical Compatibility Analysis

Lockheed Martin conducts an end-to-end electrical circuit analysis to verify proper voltage and current parameters and any required timing and sequencing interfaces between all spacecraft and launch vehicle airborne interfaces (through to the end function). This analysis requires spacecraft data, such as contact assignments, wiring interfaces, and circuit detail of avionics (first level) to verify end-to-end (spacecraft-to-launch vehicle) compatibility. All “in-between” wiring and circuits are analyzed to verify proper routing, connections, and functionality of the entire system interface. This analysis is documented as part of the ICD verification process and used to generate inputs for all necessary launch site interface testing.

5.2.24 Postflight Data Analysis

For Atlas missions, Lockheed Martin uses proven analysis techniques to obtain the individual stage and payload performance information derived from available launch vehicle telemetry data. Main outputs of the analysis are: (1) Atlas stage performance with respect to the predicted nominal (given in terms of Centaur propellant excess), (2) Centaur stage performance with respect to its predicted nominal (given also in terms of Centaur propellant excess), and (3) the average thrust and specific impulse of the Centaur stage. In addition to these outputs, the postflight performance report presents historical data for past flights of similar family and statistics of the parameters of interest. The report provides a trajectory listing of simulated Centaur flight that effectively matches observed data from the actual flight.

A primary input into the postflight vehicle performance analysis task is flight telemetry data. Telemetered outputs from the PU system are used to obtain propellants remaining in the LO_2 and liquid hydrogen (LH_2) tanks at Centaur final cutoff. Times of key vehicle mark events are also required. The actual vector states of radius and velocity at Atlas stage shutdown, compared to the predicted nominal values, provide sufficient knowledge to obtain the Atlas stage performance. The flight propellant excess at Centaur final cutoff (from the PU system data) and the actual burn times for Centaur provide key data to determine the thrust and specific impulse for the Centaur stage.

In addition to the performance evaluation of the launch vehicle, the postflight report provides an assessment of injection conditions in terms of orbital parameters and deviations from target values and spacecraft separation attitude and rates. The report also documents payload environments to the extent that the launch vehicle instrumentation permits. These environments could include interface loads, acoustics, vibration, and shock.

Finally, the report presents analyses of individual launch vehicle system performance and documents any anomalies noted during the mission. Launch vehicle and landline telemetry data provide the primary source of information for these analyses. Additionally, results of the review of optical data (from both fixed cameras at the launch site and tracking cameras), and radar data are also presented in the report.

5.2.25 Destruct System Analysis

Lockheed Martin's launch vehicle destruct system analysis is provided in our Range Safety System reports (RSSR). Atlas IIA, IIAS, IIIA, and IIIB vehicle configurations are addressed in RSSR, LMCA-Atlas-99-009. Atlas V vehicle configurations are addressed in RSSR, LMCA-Atlas-XX-XXX. The reports comply with requirements specified in Appendix 4A of EWR 127-1, October 1997.

RSSR documents provide an overview of each vehicle configuration and detailed descriptions of the flight termination system (FTS), C-band tracking system, S-band telemetry system, and ground support equipment for Eastern and Western Range Safety systems. Component and system-level testing is also described. Antenna patterns, link margins, and FTS battery load capacity analyses are included.

As indicated in Section 4.3, the Atlas program develops a spacecraft FTS configuration concurrence request for each mission (dedicated SC destruct capabilities are generally not required for communications satellites).

5.2.26 Mission Targeting

Mission targeting is conducted to define target orbit parameters that will be used to guide the launch vehicle into the desired orbit. This process requires a target specification from the spacecraft agency and results in publication of flight constants parameter loads used for the flight computer and mission-peculiar firing tables documentation.

5.2.27 Mission-Peculiar Flight Software

Our mission-peculiar software activity for mission integration is a controlled process that ensures the generation and release of validated flight control subsystem (FCS) software to support the launch schedule. Our modular software design minimizes the impact of changes due to mission-peculiar requirements. This is achieved through the generic software design philosophy, which has been applied during development and evolution of the Atlas IIAS, III, and V FCS architectures. A parameterized software design is implemented for each of these Atlas vehicle configurations, so that baseline FCS software is able to support all functionality necessary to fly most Atlas missions. Parameters are then selected to properly implement the required mission-peculiar functionality.

Periodic updates to FCS software baselines are scheduled to support updates in the vehicle hardware configuration or to implement capability enhancements as required by the Atlas program.

A rigorous software validation test program is run using the specific-mission trajectory and targeting parameters to validate the flight software and parameter data load under nominal, 3-sigma dispersed, severe stress, and failure mode environments, before release for flight. Testing and validation are completed in our Systems Integration Laboratory (SIL), which includes flight-like avionics components operating within a real-time simulation environment.

5.3 POLICIES

This section provides potential and current launch services customers with information concerning some management, integration, and production policies to ensure efficient integration and launch of the customer's payload.

5.3.1 Launch Vehicle Logos

As part of our standard launch service, the Atlas program offers customers the option of placing a mission or company logo on portions of that mission's PLF hardware. The logo can be placed in standard locations on the PLF cylindrical section. To support manufacture of the mission PLF, the Atlas program typically needs to have final artwork for the logo by 6 months before launch. This timeframe allows the Lockheed Martin engineering organization to transform the artwork into a template to be used for application of the final logo artwork onto the fairing. Delivery of the customer PLF logo design is a schedule milestone required to support nominal assembly spans for PLF fabrication. Changes to the logo shall be supplied at a time that supports scheduled PLF completion date.

5.3.2 Launch Scheduling

Atlas launch capability is nominally six missions per year depending on vehicle mix from CCAFS, LC-36B (Atlas IIAS/III) and eight launches per year from CCAFS, LC-41 (Atlas V) starting in 2003. The nominal launch rate capability from SLC-3 at VAFB is projected to be four per year.

Missions are contracted and scheduled into available launch opportunities typically 12 to 18 months in advance. Missions that are reflights of an existing bus will typically be 6 months or less. The earlier a desired schedule position is contracted for the more likely it will be available.

Scheduling and rescheduling launches in the manifest require the equitable treatment of all customers (Table 5.3.2-1). Sequential scheduling of launches in the queue, the customer's position in the queue, and vehicle processing flow time will dictate earliest launch date(s). Lockheed Martin endeavors to fill each position in the queue. Consequently, once in queue, close coordination is required should the customer desire rescheduling. Rescheduling requires mutual agreement on the selection of available launch opportunities offered by Lockheed Martin.

5.3.3 Spacecraft Launch Window Options

Atlas can be launched at any time of the day year round. However, seasonal weather patterns should be considered in setting launch windows when possible. To ensure on-time launches and avoid cost or schedule delays, missions that may be scheduled during June, July, August, and September should be planned for morning launches. Launches in the afternoon during these months have an increased probability of delays due to seasonal thunderstorm activity. Scheduling in the morning will reduce the risk of such delays and avoid cost associated with them. Options for afternoon summer launches may be available with recognition of the additional schedule delay potential.

Table 5.3.2-1 Scheduling Guidelines for Original Manifesting or Manifest Changes Due to Delays

General Policy
• Scheduling Missions in the Manifest & Adjusting Schedules Due to Atlas Delays or Customer Delays; Require Consistent & Even-Handed Treatment While Minimizing Cost & Revenue Impacts to Both the Customer & the Atlas Program • Positions Are Assigned To Satisfy Contract Provisions & Maintain Customer Satisfaction, While Avoiding or Minimizing Aggregate Delays To Manifest
Groundrules
• Publish a Publicly Releasable 18-mo Look-Ahead Manifest Annually in July • Every Effort Is Made To Conduct Customer Launches in a Period Desired by the Customer; However, Should a Customer's Scheduled Launch Date Conflict with That of Another Customer, That Customer with the Earlier Effective Contract Date May Be Considered in Determining Which Customer Is Entitled To Launch First • If a Customer Contracts for Two or More Launches, Payloads May Be Interchanged Subject to Mutual Agreement
Delays
• Once in Queue, Customers Will Stay in Queue if Lockheed Martin Causes a Delay or if the Customer Causes a Short Delay • Customer Delays (Announced or Anticipated) May Require Resequencing to the Next Available Launch Opportunity • A Customer-Directed Delay May Be Considered Equivalent to a New Contract Award Date for Priority Determination
Exceptions
• Reflight or Replacement Launches for Satellite or Launch Vehicle Failure May Be Given Priority Within the Manifest Guidelines • Planetary Window Missions Will Be Given Special Consideration Within the Manifest Guidelines
Note: Mission Scheduling Guidelines Are General Guidelines Only & Subject To Change

6.0 SPACECRAFT AND LAUNCH FACILITIES

Lockheed Martin has formal agreements with the United States Air Force (USAF), the National Aeronautics and Space Administration (NASA), and Astrotech for use of payload and launch vehicle processing facilities at and near Atlas launch sites at Cape Canaveral Air Force Station (CCAFS), Florida. Similar agreements for sites at Vandenberg Air Force Base (VAFB) in California have been implemented. Long-term use agreements are in place for Launch Complexes (LC) 36 and 41 at CCAFS and for Space Launch Complex (SLC) 3 at VAFB (for Atlas IIAS only) that encompass facilities, range services, and equipment.

This section summarizes launch facilities capabilities available to Atlas customers. Both CCAFS and VAFB facilities are discussed. Additional CCAFS facilities data can be found in the *Atlas Launch Services Facilities Guide*. Additional VAFB facilities data can be obtained by contacting Lockheed Martin. This section is organized as follows: 6.1—CCAFS Spacecraft Facilities, 6.2—Atlas IIAS/III Launch Site Facilities, 6.3—Atlas V Launch Site Facilities, 6.4—VAFB Spacecraft Facilities, and 6.5—Atlas IIAS SLC-3.

6.1 CCAFS SPACECRAFT FACILITIES

CCAFS facilities include spacecraft processing facilities available to commercial and U.S. government users. Figures 6.1-1 and 6.1-2 illustrate the location of facilities between Launch Complex (LC) 36 and Astrotech.

6.1.1 Astrotech

The Astrotech commercial payload processing facility (PPF), owned and operated by Spacehab Inc., is the primary facility for processing Atlas class civil government, and commercial spacecraft. This facility (Fig. 6.1.1-1) contains separate nonhazardous and hazardous processing buildings, storage buildings, and offices. The facilities and floor plans are described in the following sections. Astrotech complies fully with all applicable federal, state, regional, and local statutes, ordinances, rules, and regulations relating to safety and environmental requirements.

Should the Astrotech facility not adequately satisfy spacecraft processing requirements, government facilities, as outlined in Lockheed Martin/NASA and Lockheed Martin/USAF agreements, are available. These facilities also are described in the following sections.

Astrotech PPF—Astrotech Building 1, with its high bay expansion, is considered the primary PPF. Its floor plan is depicted in Figures 6.1.1-2 and 6.1.1-3. With overall dimensions of approximately 61.0 m (200 ft) by 38.1 m (125 ft) and a height of 14 m (46 ft), the building's major features are:

1) An airlock;
2) High bays (three identical and one larger);
3) Control rooms (two per high bay);
4) Office complex, administrative area, communications mezzanine, conference rooms, and support areas.

Table 6.1.1-1 lists details of room dimensions, cleanliness, and crane capabilities of this facility.

Astrotech Hazardous Processing Facility (HPF)—Astrotech's Building 2 is considered the primary HPF. With overall dimensions of approximately 48.5 m (159 ft) by 34.1 m (112 ft) and a height of 14 m (46 ft), the major features are:

1) An airlock,
2) Two spacecraft processing high bay and operations rooms,
3) Two encapsulation high bays,
4) One spin balance bay.

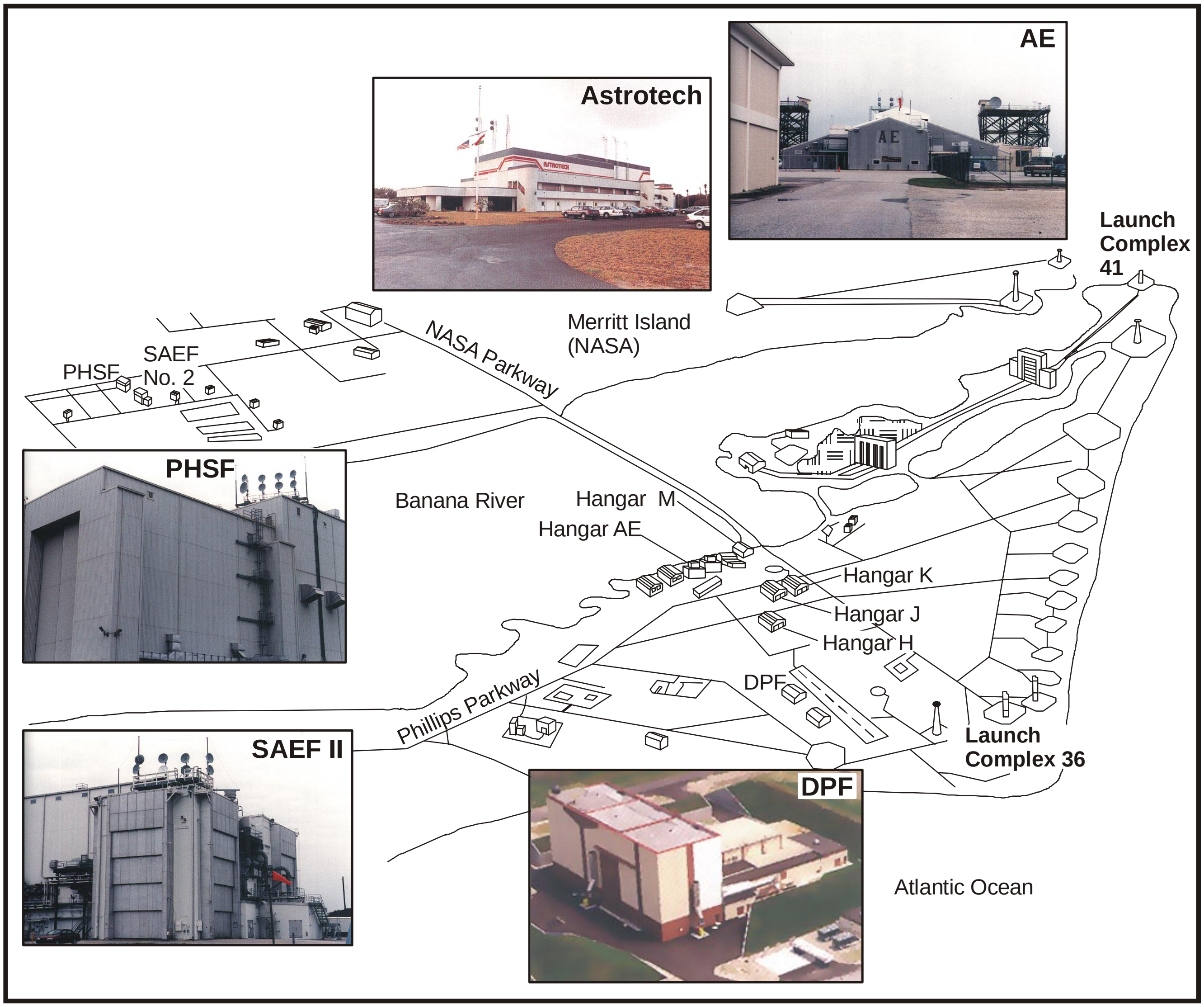


Figure 6.1-1 Facilities at Cape Canaveral Air Force Station, Florida

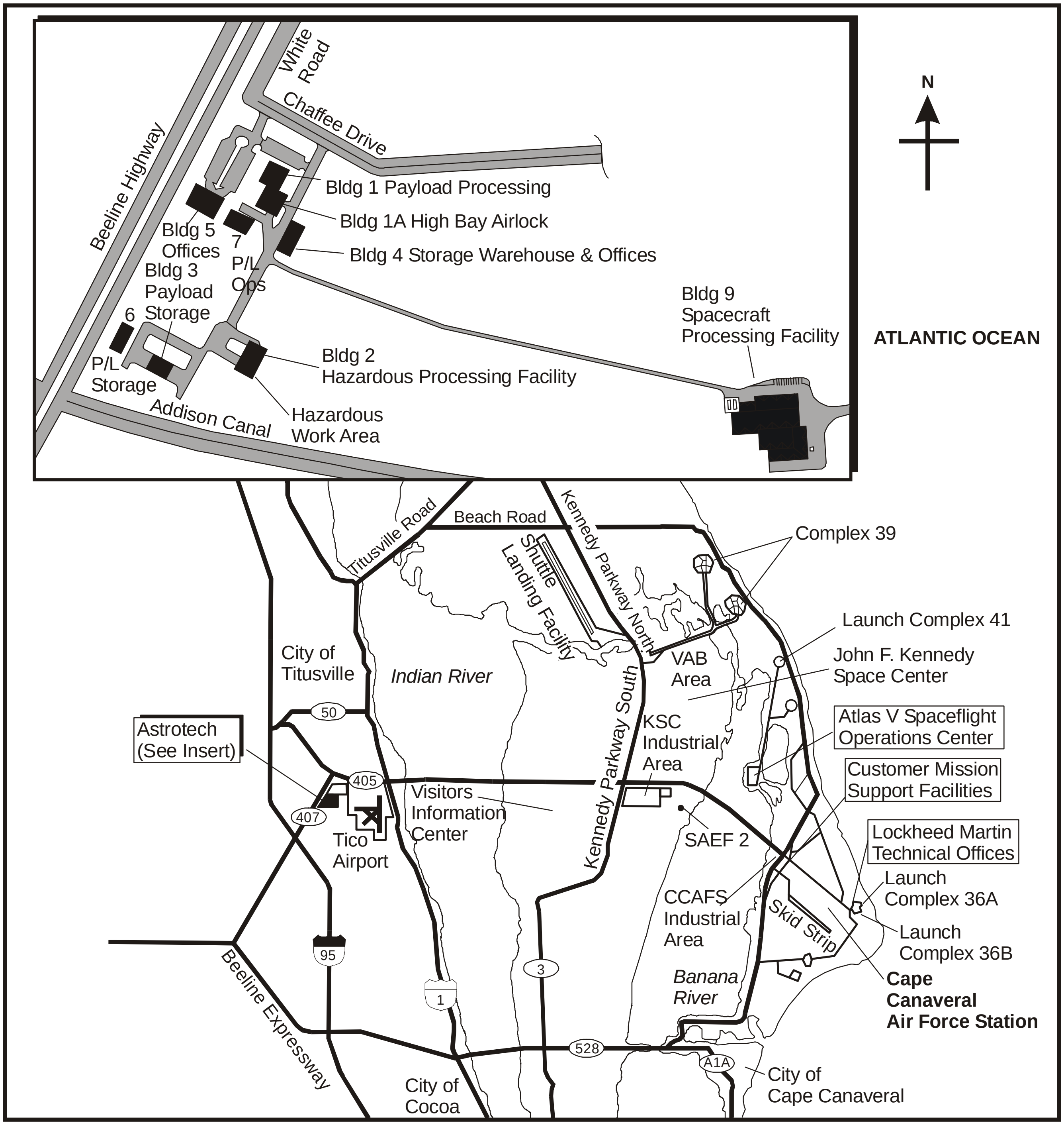


Figure 6.1-2 Facility Locations at CCAFS

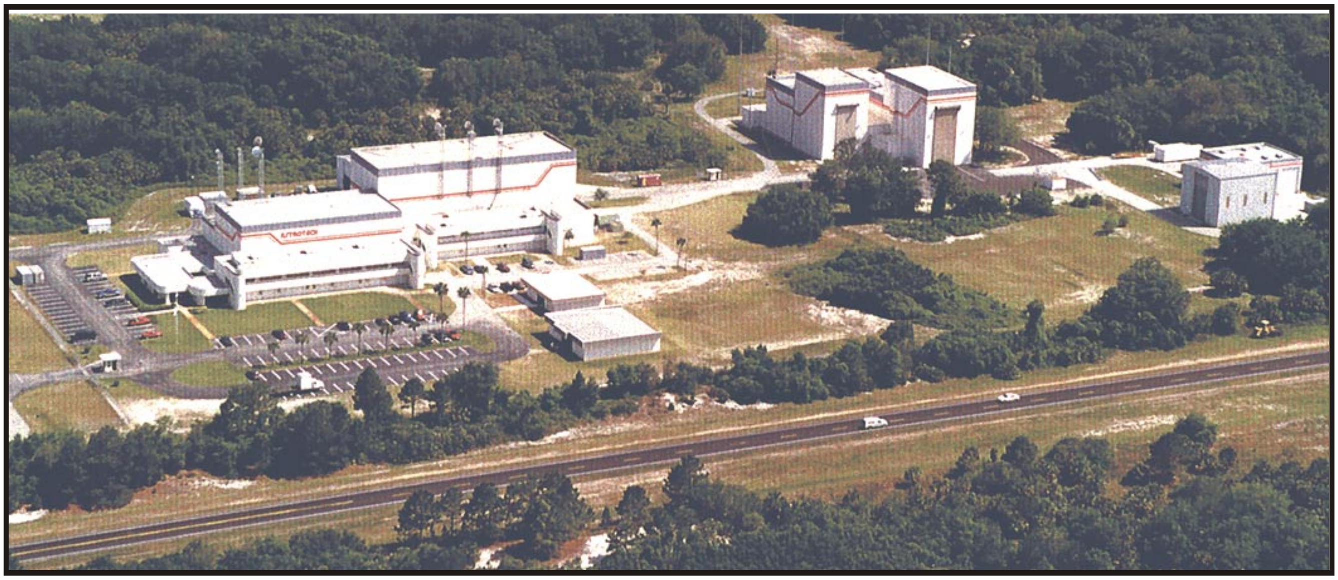
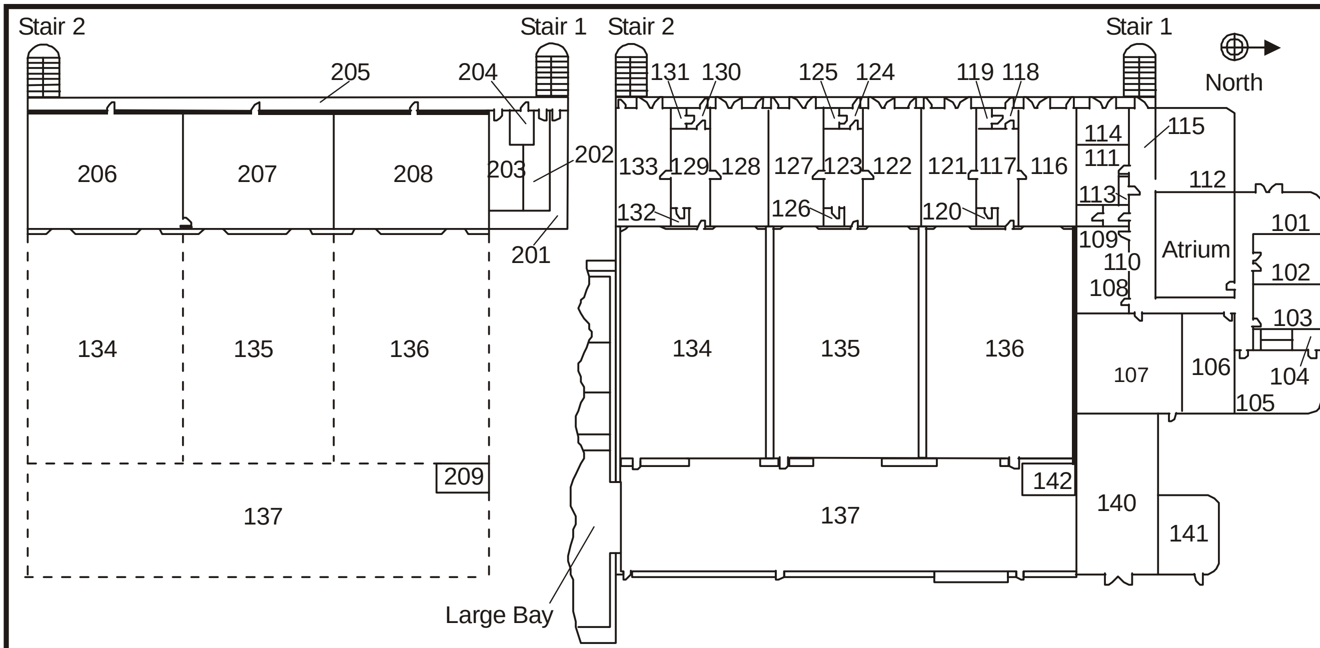
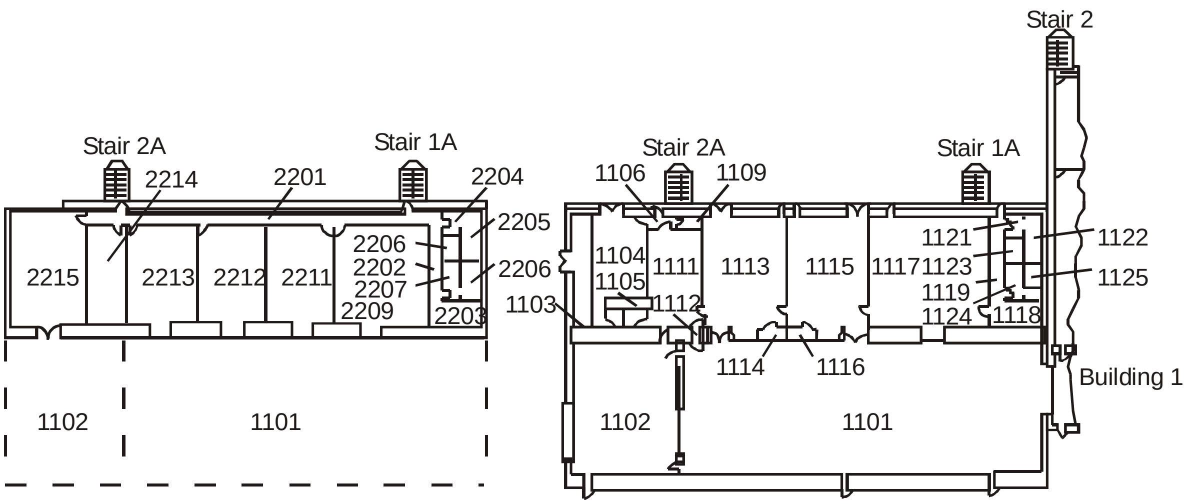


Figure 6.1.1-1 Astrotech Facility



Room	Function	Room	Function	Room	Function
108	Conference Room	121	Control Room A2	133	Control Room C2
109	Women's Restroom	122	Control Room B1	134	High Bay C
110	Women's Lounge	123	Change Room B	135	High Bay B
111	Men's Restroom	124	Vestibule B	136	High Bay A
112	Break/Lunch Room	125	Storage B	137	Common Airlock
114	Machine Shop	126	Restroom B	202	Women's Restroom
115	Corridor	127	Control Room B2	203	Men's Restroom
116	Control Room A1	128	Control Room C1	205	Corridor
117	Control Room A	129	Change Room C	206	Office Area C
118	Vestibule 1	130	Vestibule C	207	Office Area B
119	Storage A	131	Storage C	208	Office Area A
120	Restroom A	132	Restroom C		

Figure 6.1.1-2 Astrotech Building 1 Detailed Floor Plan



Room	Function	Room	Function	Room	Function
1101	Large High Bay D	1117	Office Area D1	2205	Men’s Restroom
1102	Large Airlock	1118	Break Room	2207	Women’s Washroom
1104	Conference D1	1119	Corridor	2208	Women’s Restroom
1105	Closet	1121	Men’s Washroom	2209	Office Area D2
1106	Restroom	1122	Men’s Restroom	2211	Office Area D3
1107	Restroom	1124	Women’s Washroom	2212	Office Area D4
1108	Vestibule	1125	Women’s Restroom	2213	Office Area D5
1111	Change Room D	2201	Corridor	2214	Office Area D2
1112	Air Shower	2202	Corridor	2215	
1113	Control Room D2	2203	Breakroom		
1115					

Figure 6.1.1-3 Astrotech Building 1 Large Bay Detailed Floor Plan—First Floor

Table 6.1.1-1 Astrotech's Building 1 features four high bay areas.

Building 1 High Bays (3): Class 100,000 Clean Room		
Temperature	23.8 ±2.8°C*	75 ±5°F*
Relative Humidity	50 ±5%*	
Usable Floor Space	18x12.19 m	60x40 ft
Ceiling Height	13.2 m	43.5 ft
Crane Type (Each Bay)	Bridge	
Crane Capacity	9.07 tonne	10 ton
Crane Hook Height	11.3 m	37.1 ft
Door Size (wxh)	6.1x7 m	20x23 ft
High Bay Control Rooms		
Size	9.1x12.8 m	30x42 ft
Ceiling Height	2.67 m	8.75 ft
Door Size (wxh)	2.44x2.44 m	8x8 ft
Bay Window Size	1.22x2.44 m	4x8 ft
Temperature	23.8 ±2.8°C	75 ±5°F
Airlock: Class 100,000 Clean Room		
Temperature	23.8 ±2.8°C*	75 ±5°F*
Relative Humidity	50 ±5%*	
Usable Floor Space	36.6x9.14 m	120x30 ft
Ceiling Height	7.0 m	23 ft
Door Size (wxh)	6.1x7 m	20x23 ft
Building 1 Large Bay Control Rooms (2)		
Size	9.1x10.7 m	30x35 ft
Ceiling Height	2.8 m	9.33 ft
Door Size (wxh)	2.44x2.44 m	8x8 ft
Temperature	23.8 ±2.8°C	75 ±5°F

Offices (Second Floor) (A, B, C High Bays)		
Floor Size	12.5x7.3 m	41x24 ft
Acoustic Ceiling Height	2.43 m	8 ft
Door Size (wxh)	.91x2.04 m	3x6.7 ft
Bay Window Size	1.22x1.22 m	4x4 ft
Building 1 High Bay: 100,000 Clean Room		
Temperature	23.8 ±2.8°C	75 ±5°F
Relative Humidity	50 ±5%	
Usable Floor Space	38.1x15.5 m	125x51 ft
Ceiling Height	18.3 m	60 ft
Crane Type (Each Bay)	Bridge	
Crane Capacity[1]	27.21 tonne	30 ton
Crane Hook Height	15.2 m	50 ft
D High Bay Airlock Door Size (wxh)	6.1x15.2 m	20x50 ft
Main Airlock Door Size (wxh)	6.1x7 m	20x23 ft
Building 1 Large Bay Airlock		
Temperature	23.8 ±2.8°C	75 ±5°F
Relative Humidity	50 ±5%	
Usable Floor Space	12.2x15.5 m	40x51 ft
Ceiling Height	18.3 m	60 ft
Door Size (wxh)	6.1x15.2 m	20x50 ft
Crane Hook Height	15.2 m	50 ft
Crane Capacity[1]	27.21 tonne	30 ton
Offices (Second Floor) (D High Bay)		
Floor Size (D1)	10.7x12.5 m	35x41 ft
Floor Size (D2)	9.1x10.7 m	30x35 ft
Floor Size (D3)	7.6x9.1 m	25x30 ft
Ceiling Height	2.8 m	9.33 ft

[1] Building 1 D high bay and airlock crane capacity is 27.21 tonnes (30 tons). A 15-ton hook is installed as standard, however the 30-ton hook can be installed at the request of the customer. The crane is normally proof-tested for the 15-ton hook only.

The Building 2 floor plan is depicted in Figure 6.1.1-4. Each high bay is built to explosion-proof or equivalent standards to support operations involving liquid propellant transfer, solid propellant motor preparations, and ordnance installation. Table 6.1.1-2 lists the facility's room dimensions, cleanliness levels, and crane capabilities.

Astrotech Spacecraft Processing Facility (SPF)—Astrotech's Building 9 provides an additional hazardous processing facility. This facility is being built to process the larger 5-m fairings associated with Atlas V vehicles. Final preparations of the spacecraft and their fairings will be carried out in this facility in a similar manner to those in the HPF. Major areas of the SPF are:

1) One airlock,
2) One encapsulation bay,
3) Two high bays,
4) Two propellant cart storage rooms,
5) Two garment change rooms,
6) Three control rooms.

The Building 9 floor plan is depicted in Figure 6.1.1-5. Each processing cell is built to explosion-proof or equivalent standards to support operations involving liquid propellant transfer, solid propellant motor preparations, and ordnance installation. Table 6.1.1-3 lists the facility's room dimensions, cleanliness levels, and crane capabilities.

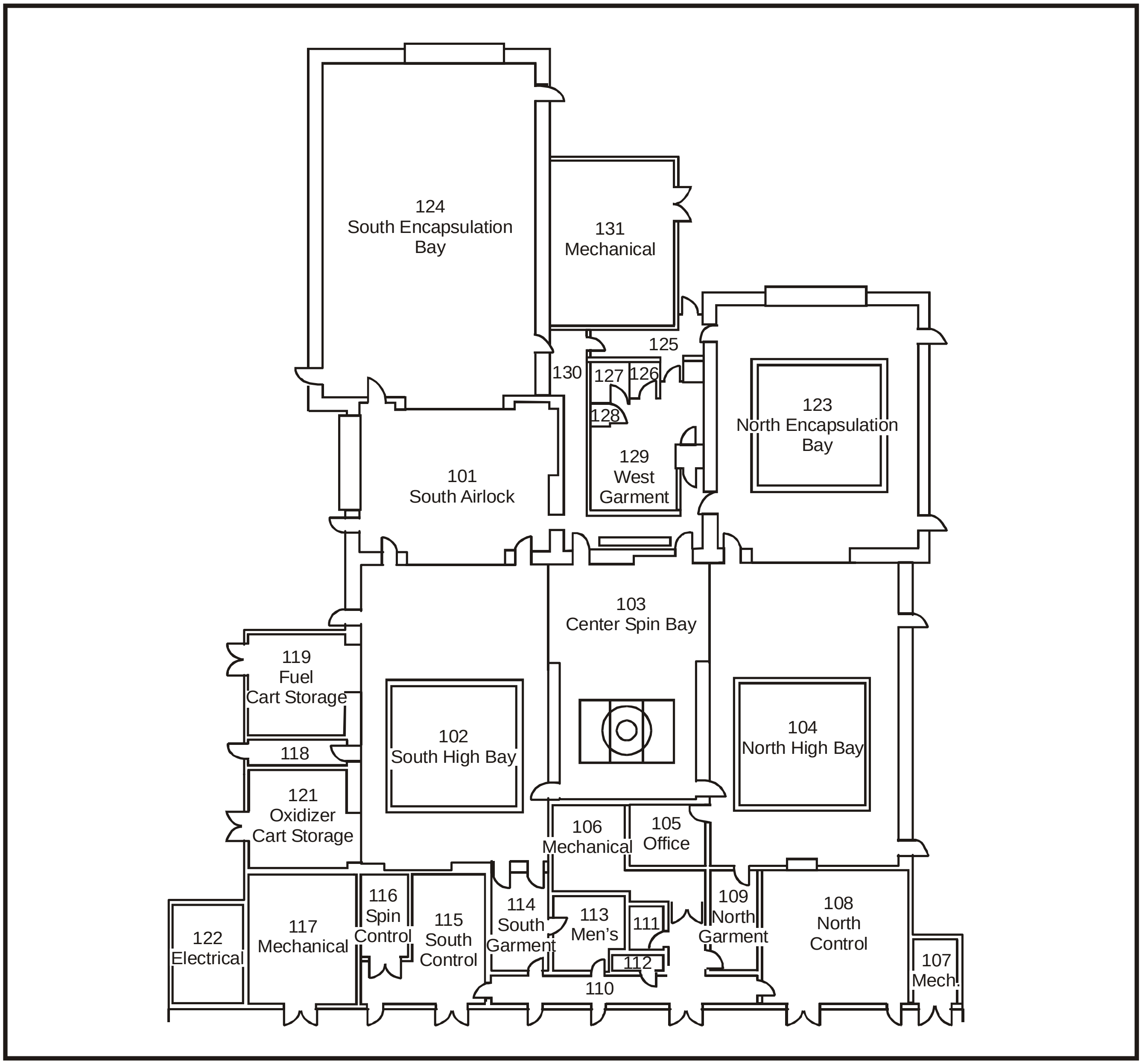


Figure 6.1.1-4 Astrotech's Building 2 provides excellent hazardous operations facilities for liquid and solid propellant based satellites.

Table 6.1.1-2 Astrotech's Building 2 features three blast-proof processing rooms.

South & North High Bays (Rooms 102, 104)		
Cleanliness	Class 100,000	
Temperature	23.9±2.8°C*	75±5°F*
Relative Humidity	50 ±5%*	
Floor Space	18.3x11.3 m	60x37 ft
Fuel Island	7.6x7.6 m	25x25 ft
Ceiling Height	13.1 m	43 ft
Doors to Room 101 & 123 (wxh)	6.1x12.2 m	20x40 ft
Door to Room 103 (wxh)	6.1x13.1 m	20x43 ft
Crane Type (Each Bay)	Bridge	
Crane Capacity	9.1 tonne	10 ton
Crane Hook Height	11.0 m	36.2 ft
North High Bay Control Room (Room 108)		
Temperature	21.1 to 23.9°C	70 to 75°F
Floor Space	7.6x9.1 m	25x30 ft
Ceiling Height	2.84 m	9.33 ft
Door to Outside (wxh)	2.4x2.4 m	8x8 ft
Bay Window (wxh)	0.81x0.76 m	2.67x2.50 ft
South High Bay Control Room (Room 115)		
Temperature	21.1 to 23.9°C	70 to 75°F
Floor Space	7.6x7.6 m	25x25 ft
Ceiling Height	2.84 m	9.33 ft
Door to Outside (wxh)	2.4x2.4 m	8x8 ft
Bay Window (wxh)	0.81x0.76 m	2.67x2.50 ft
Spin Balance High Bay (Room 103)		
Cleanliness	Class 100,000	
Temperature	23.9±2.8°C*	75±5°F*
Relative Humidity	50±5%*	
Floor Space	14.6x8.2 m	48x27 ft
Ceiling Height	13.1 m	43 ft
Doors to Room 102 & 104, (wxh)	6.1x13.1 m	20x43 ft
Crane Type	Bridge	
Crane Capacity	9.1 tonne	10 ton
Crane Hook Height	11.0 m	36.2 ft
Spin Balance High Bay Control Room (Room 116)		
Temperature	23.9±2.8°C	75±5°F
Floor Space	3x4.6 m	10x15 ft
Ceiling Height	2.84 m	9.33 ft
Door to Room 115 (wxh)	1.8x2.03 m	6x6.67 ft

Airlock (Room 101)		
Cleanliness	Class 100,000	
Temperature	23.9±2.8°C*	75±5°F*
Relative Humidity	50±5%*	
Floor Space	11.6x8.8 m	38x29 ft
Ceiling Height	13.1 m	43 ft
Door to Room 124 (wxh)	6.1x12.2 m	20x40 ft
Door to Room 102 (wxh)	6.1x12.2 m	20x40 ft
Crane	Monorail	
Crane Capacity	1.8 tonne	2 ton
Crane Hook Height	11.3 m	37 ft
Prop Cart Rooms (Rooms 119, 121)		
Floor Space	6.1x6.1 m	20x20 ft
Ceiling Height	2.84 m	9.33 ft
Door to Room 102 (wxh)	3.1x2.4 m	10x8 ft
Door to Outside (wxh)	1.8x2.03 m	6x6.67 ft
North Encapsulation High Bay (Room 123)		
Cleanliness	Class 100,000	
Temperature	23.9±2.8°C*	75±5°F*
Relative Humidity	50±5%*	
Floor Space	15.2x12.2 m	50x40 ft
Fuel Island	7.6x7.6 m	25x25 ft
Ceiling Height	19.8 m	65 ft
Door to Outside (wxh)	6.1x15.2 m	20x50 ft
Door to Room 104 (wxh)	6.1x12.2 m	20x40 ft
Crane Type	Bridge	
Crane Capacity	27.2 tonne	30 ton
Crane Hook Height	16.99 m	55.75 ft
South Encapsulation High Bay (Room 124)		
Cleanliness	Class 100,000	
Temperature	23.9±2.8°C*	75±5°F*
Relative Humidity	50±5%*	
Floor Space	13.7x21.4 m	45x70 ft
Ceiling Height	19.8 m	65 ft
Door to Outside (wxh)	7.0x16.8 m	23x55 ft
Door to Room 101 (wxh)	6.1x12.2 m	20x40 ft
Crane Type	Bridge	
Crane Capacity	27.2 tonne	30 ton
Crane Hook Height	17.1 m	56.7 ft

* Can be Adjusted as Needed

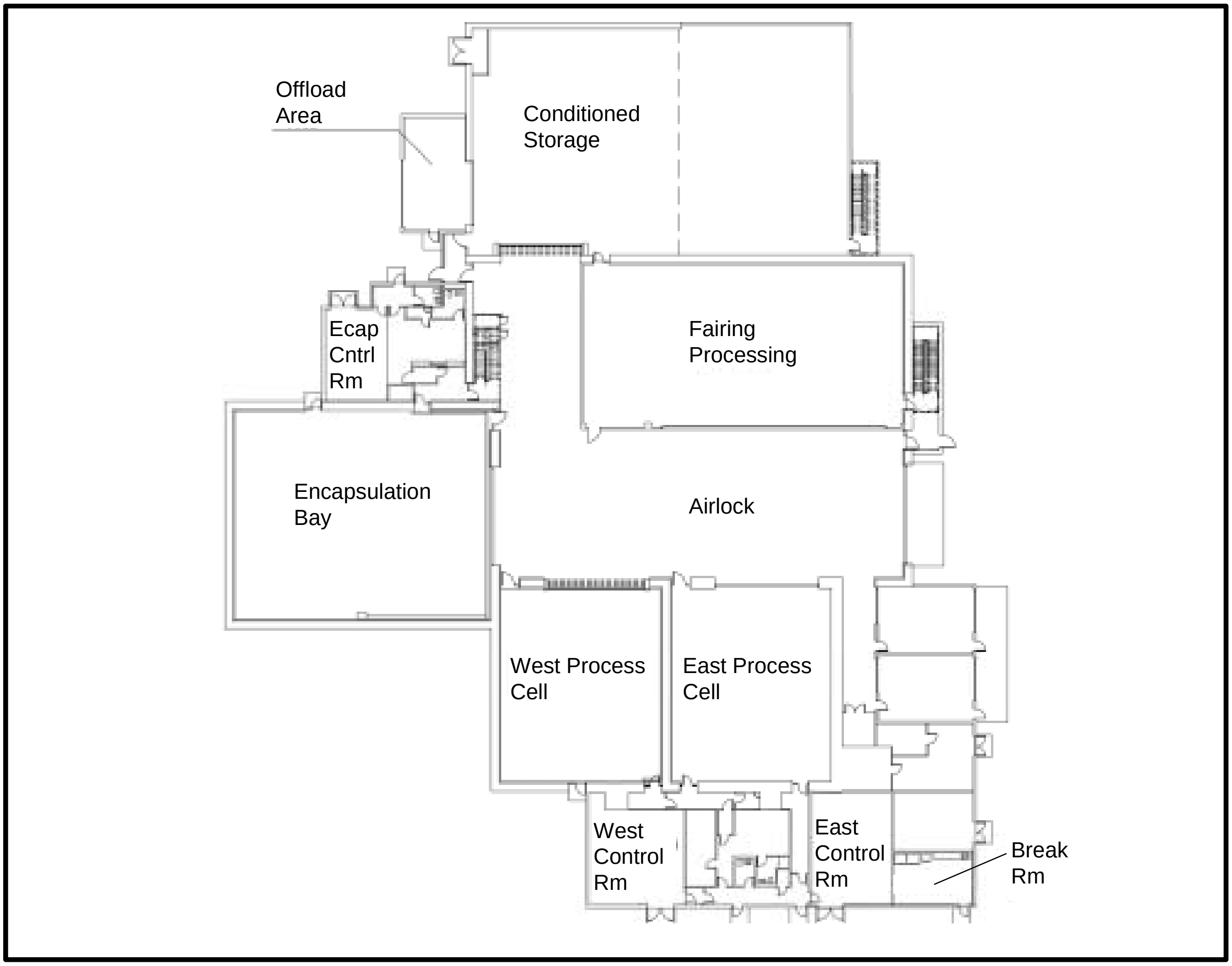


Figure 6.1.1-5 Astrotech's Building 9 Layout

Table 6.1.1-3 Astrotech's Building 9 features two processing rooms.

Airlock		
Cleanliness	Class 100,000	
Temperature	23.9±2.8°C*	75±5°F*
Relative Humidity	50±5%*	
Floor Space	38.7x30.2 m	127x99 ft
Ceiling Height	30.8 m	101 ft
Doors to Est/Wst High Bays (wxh)	9.1x19.8 m	30x65 ft
Door to Encapsulation Bay (wxh)	9.1x27.7 m	30x91 ft
Door to Outside (wxh)	9.1x27.7 m	30x91 ft
Door to Fairing Process Rm (wxh)	21.3x7.6 m	70x25 ft
Door to Fairing Cnd Storage (wxh)	7.6x19.8 m	25x65 ft
Crane	Bridge/Trolley	
Crane Capacity	27.15 tonne	30 ton
Crane Hook Height	27.7 m	91 ft
East High Bay		
Cleanliness	Class 100,000	
Temperature	23.9±2.8°C*	75±5°F*
Relative Humidity	50 ±5%*	
Floor Space	18.3x15.2 m	60x50 ft
Ceiling Height	24.4 m	80 ft
Access Door (wxh)	0.9x2.1 m	3x7 ft
Crane Type (Each Bay)	Bridge/Trolley	
Crane Capacity	22.67 tonne	25 ton
Crane Hook Height	22.3 m	73 ft
East High Bay Control Room		
Temperature	21.1–23.9°C	70–75°F
Floor Space	10.7x7.6 m	35x25 ft
Ceiling Height	3.2 m	10.5 ft
Door for EGSE Installation (wxh)	2.4x3.0 m	8x10 ft
Access Door (wxh)	0.9x2.1 m	3x7 ft
West High Bay		
Cleanliness	Class 100,000	
Temperature	23.9±2.8°C*	75±5°F*
Relative Humidity	50 ±5%*	
Floor Space	18.3x15.2 m	60x50 ft
Ceiling Height	24.4 m	80 ft
Access Door (wxh)	0.9x2.1 m	3x7 ft
Crane Type (Each Bay)	Bridge/Trolley	
Crane Capacity	27.21 tonne	30 ton
Crane Hook Height	22.3 m	73 ft
West High Bay Control Room		
Temperature	21.1–23.9°C	70–75°F
Floor Space	9.3x9.3 m	30.5x30.5 ft
Ceiling Height	3.2 m	10.5 ft
Door for EGSE Installation (wxh)	2.4x3.0 m	8x10 ft
Access Door (wxh)	0.9x2.1 m	3x7 ft
East/West Garment Change Room		
Floor Space	6.7x4.4 m	22x14.5 ft
	2.7 m	9 ft
Encapsulation Bay		
Cleanliness	Class 100,000	
Temperature	23.9±2.8°C*	75±5°F*
Relative Humidity	50±5%*	
Floor Space	24.4x19.8 m	80x65 ft
Ceiling Height	33.5 m	110 ft
Crane Type	Bridge/Trolley	
Crane Capacity	45.25 tonne	50 ton
Crane Hook Height	30.5 m	100 ft
Encapsulation Control Room		
Temperature	21.1–23.9°C	70–75°F
Floor Space	8.5 x5.8 m	28x19 ft
Ceiling Height	3.2 m	10.5 ft
Door for EGSE Installation (wxh)	2.4x3.0 m	8x10 ft
Access Door (wxh)	0.9x2.1 m	3x7 ft
Encapsulation Bay Garment Change Room		
Floor Space	5.2x7.3 m	17x24 ft
Ceiling Height	2.7 m	9 ft
Prop Cart Rooms		
Floor Space	6.7x4.4 m	22x14.5 ft
Height	2.7 m	9 ft
Door to Airlock (wxh)	3.05x3.0m	10x10 ft
Door to Outside (wxh)	3.7x3.0 m	12x10 ft
Break Room		
Floor Space	7.6x5.2 m	25x17 ft
Ceiling Height	2.7 m	9 ft
Fairing Processing Area		
Floor Space	30.8x15.7 m	101x51.5 ft
Ceiling Height	9.1 m	30 ft
Conditioned Storage Area		
Floor Space	30.8x15.7 m	101x51.5 ft
Ceiling Height	19.8 & 6.1 m	65 & 20 ft
Conditioned Storage Airlock		
Floor Space	11.6x9.1 m	38x30 ft
Ceiling Height	11.6 m	11.6 m
Door to Outside (wxh)	6.1x9.1 m	20x30 ft
Door to Airlock (wxh)	6.1x9.1 m	20x30 ft
Crane Capacity	13.6 tonne	15 ton
Crane Hook Height	9.1 m	30 ft
*Can be Adjusted as Needed		

Astrotech Spacecraft Support Facilities—Building 3 at Astrotech is a thermally controlled, short-term payload storage area with payload processing activities or long-term satellite storage. Storage bay and door dimensions and thermal control ranges are identified in Table 6.1.1-4.

Astrotech's Building 4 is a warehouse storage area without environmental controls and is suitable for storage of shipping containers and mechanical ground support equipment (GSE). Table 6.1.1-5 details this facility's dimensions.

Astrotech's Building 5 provides 334.4 m^2 (3,600 ft^2) of customer office space divided into 17 offices with a reception area sufficient to accommodate up to three secretaries.

Building 6 at Astrotech provides storage primarily for Lockheed Martin payload fairing (PLF) support equipment. However, if customers require additional nonenvironmentally controlled storage, this additional space can be made available.

Spacecraft Services—Full services for payload processing and integration can be provided at Astrotech.

Electrical Power and Lighting—The Astrotech facility is served by 480-Vac/three-phase commercial 60- and 50-Hz electrical power that can be redistributed as 480-Vac/three-phase/30-A, 120/208-Vac/three-phase/60-A, or 120-Vac/single-phase, 20-A power to any location in Buildings 1 and 2. Commercial power is backed up by a diesel generator during critical testing and launch periods. Astrotech can provide 35 kW of 230/380-Vac/three-phase 50-Hz power, which is also backed up by a diesel generator.

Table 6.1.1-4 Astrotech's payload storage building is used for short-term hardware storage.

Building 3: Thermally Controlled Storage Facility		
Temperature Control	25.6°C	to 70 to 78°F
Relative Humidity	50±10%	
Floor Space		
• Bays A, C, D & F	7.6x6.7 m	25x22 ft
• Bays B & E	7.6x7.3 m	25x24 ft
Height (All Bays)	8.5 m	28 ft
Door Size (wxh)		
• Bays A, C, D & F	6.1x7.6 m	20x25 ft
• Bays B & E	5.5x7.6 m	18x25 ft
Crane	None	

Table 6.1.1-5 Astrotech's warehouse storage building is suitable for storage of items not requiring climate controls.

Building 4: Storage Without Environmental Control		
Floor Space	15.2x38.1 m	50x125 ft
Height	8.5 m	28 ft
Door Size	6.1x7.9 m	20x26 ft
Crane	None	

The high bays and airlocks in Buildings 1 and 2 are lighted by 400-W metal halide lamps to maintain 100 fc of illumination. Control rooms, offices, and conference areas have 35-W fluorescent lamps to maintain 70 fc of illumination.

Telephone and Facsimile—Astrotech provides all telephone equipment, local telephone service, and long-distance access. A Group 3 facsimile machine is available.

Intercommunication Systems—Astrotech provides a minimum of three channels of voice communications among all work areas. The facility is connected to the NASA/USAF Operational Intercommunications System and Transistorized Operational Phone System (TOPS) to provide multiple-channel voice communications between the Astrotech facility and selected locations at CCAFS.

Closed-Circuit Television (CCTV)—CCTV cameras are located in high bays of Building 2 and can be placed in high bays of Building 1, as required, to permit viewing operations in those areas. CCTV can be distributed within the Astrotech facility to any location desired. In addition, Astrotech has the capability to transmit and receive a single channel of video to and from Kennedy Space Center (KSC)/CCAFS via a dedicated microwave link.

Command and Data Links—Astrotech provides wideband and narrowband data transmission capability and the KSC/CCAFS cable transmission system to all locations served by the KSC/CCAFS network. If a spacecraft requires a hardline transmission capability, the spacecraft is responsible for providing correct signal characteristics to interface to the KSC/CCAFS cable transmission system.

Astrotech provides antennas for direct S-band, C-band, and Ku-band airlinks from the Astrotech facility to LC-36 and antennas for S-band, C-band, and Ku-band airlinks between Astrotech Buildings 1 and 2. There is also a ground connection between the PPF and HPF for hardline radio frequency (RF) transmissions up to Ku-band.

Customer Local Area Network (LAN)—Astrotech provides customer assistance for connectivity into an existing LAN that has drops available in the PPF and HPF. System interface is via standard RJ 45 connectors. End-item instruments, such as hubs, are customer provided. A limited number of drops on this system are available at the mission director's center (MDC) in Hangar AE, the LC-36 blockhouse, and both launch services buildings (LSB). If the customer has arranged for a T1 line from an outside provider for LAN connectivity to their home facilities, Astrotech will ensure it is properly connected to the customer's PPF control room. Customers are responsible for verifying connectivity of the T1 line from the Astrotech facility back to their own facility.

Remote Spacecraft Control Center—Astrotech has the capability to link remote ground stations (voice and data) between Astrotech and CCAFS resources.

Temperature and Humidity Control—The environment of all Astrotech high bays and airlocks is maintained at a temperature of 24 ±2.8°C (75 ±5°F) and a relative humidity of 50 ±5%. The environment of all other areas is maintained by conditioned air at a temperature between 21 and 25°C (70 to 78°F) and a comfortable humidity.

Compressed Air—Regulated compressed air at 125 psi is available in Buildings 1 and 2.

Security and Emergency Support—Perimeter security is provided 24 hours a day. Access to the Astrotech facility is via the main gate, where a guard is posted during working hours to control access. Cypher locks on all doors leading into payload processing areas provide internal security. Brevard County provides emergency medical support and the City of Titusville provides emergency fire support. In an accident, personnel will be transported to Jess Parish Hospital in Titusville. Both medical and fire personnel have been trained by NASA.

Foreign Trade Zone—Astrotech has been designated as a foreign trade zone. Astrotech will coordinate all licensing requirements to meet governmental regulations for importing and exporting support hardware for the duration of mission support.

6.1.2 USAF and NASA Facilities

Building AE Mission Operations Facility—Building AE is a spacecraft and missions operations facility originally constructed by the USAF. With overall dimensions of approximately 36.6 m (120 ft) by 97.5 m (320 ft), it features:

1) Spacecraft checkout areas (high and low bays),
2) Mission director's center and guest observation room (Fig. 6.1.2-1),
3) Telemetry ground station and laboratory (including Astrotech's communications link with the Range).

Building AE is located in the CCAFS industrial area on Hangar Road. Table 6.1.2-1 provides a detailed description of Building AE facilities.

Spacecraft Assembly and Encapsulation Facility (SAEF) No. 2 PPF—SAEF 2 (Fig. 6.1.2-2) is a NASA facility located southeast of the KSC industrial area. It features:

1) One high bay,
2) Two low bays,
3) One test cell,
4) Two control rooms.

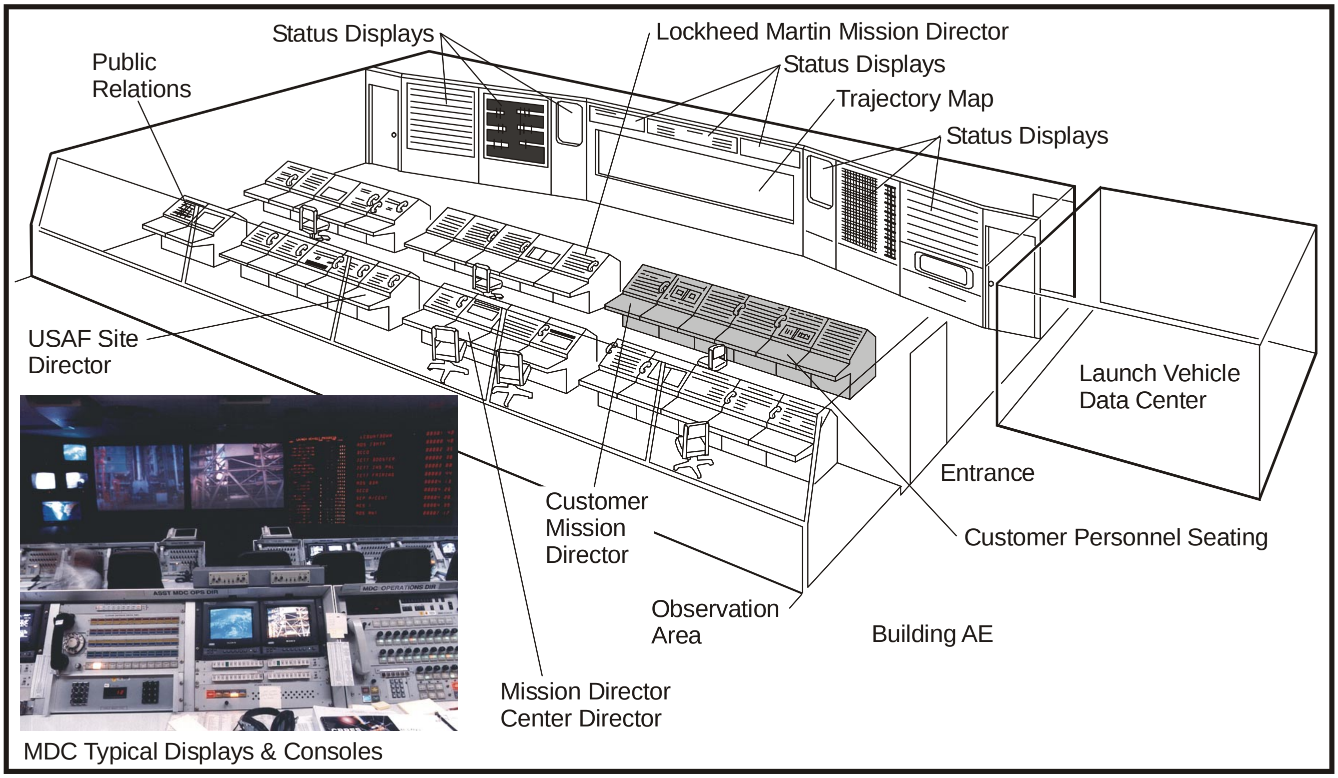


Figure 6.1.2-1 Mission Director's Center

Table 6.1.2-1 Building AE provides adequate space for smaller spacecraft processing activities.

Spacecraft Laboratory			**Airlock**		
Low Bay Floor Area	108 m^2	1,1610 ft^2	Floor Space	5.5x14.6 m	18x48 ft
Height	6.1 m	20 ft	Floor Area	80.3 m^2	864 ft^2
Door Size	2.4x3 m	8x10 ft	Room Height	12.8 m	42 ft
Crane Type	Bridge		Crane Type	Monorail	
Crane Capacity	1,814 kg	2 ton	Crane Capacity	5,443 kg	6 ton
Crane Hook Height	11.3 m	37 ft	Crane Hook Height	10.3 m	33.75 ft
High Bay Clean Room (Four Rooms)			**Airlock Outside Door**		
Test Area			Width	4.9 m	16 ft
Floor Space	14.6x19.2 m	48x63 ft	Height	11.1 m	36.5 ft
Floor Area	280.32 m^2	3,024 ft^2	**Test & Storage Area Rooms**		
Room Height	12.8 m	42 ft	Southside Room		
Crane Type	Bridge	Not Clean Room Compatible	Floor Space	9.4x12.2 m	31x40 ft
Crane Capacity	2,722 kg	3 ton	Floor Area	114.68 m^2	1,240 ft^2
Crane Hook Height	10.3 m	33.75 ft	Room Height	3 m	10 ft
Crane Type	Monorail		South Door	3x3 m	10x10 ft
Crane Capacity	1,814 kg	2 ton	North Door	2.5x2(h) m	8.5x7(h) ft
Crane Hook Height	11.3 m	37 ft			

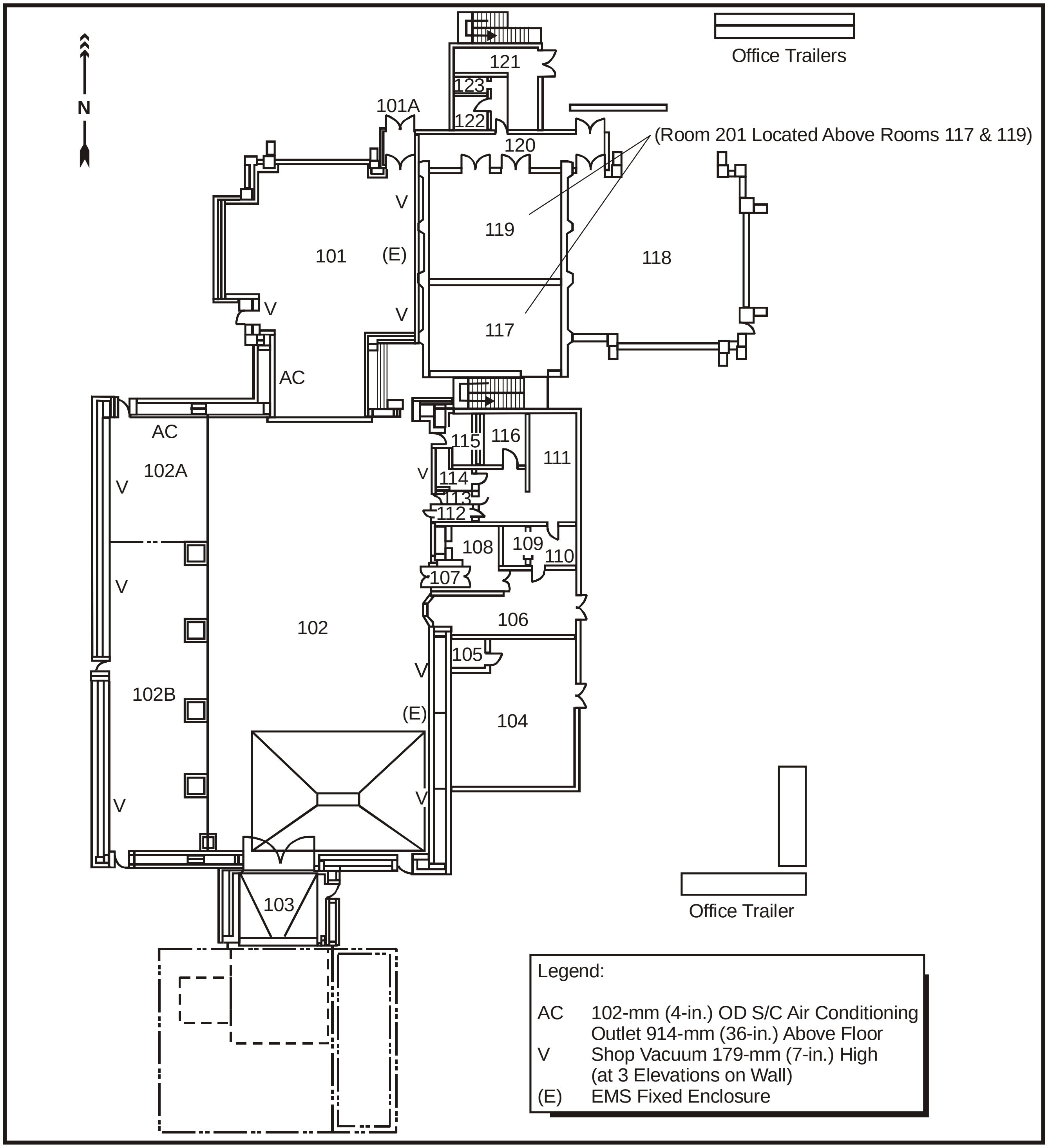


Figure 6.1.2-2 Building Floor Plan for SAEF 2

Details of the SAEF 2 facility are listed in Table 6.1.2-2.

Payload Hazardous Servicing Facility (PHSF)—The PHSF is a NASA facility located southeast of the KSC industrial area (next to SAEF 2). Additional features of the PHSF Service Building are described in Table 6.1.2-3.

6.1.3 Defense System Communication Satellite (DSCS) Processing Facility (DPF)

The DSCS DPF is a USAF facility accommodating hazardous and nonhazardous payload processing operations. It provides an area in which to process and encapsulate payloads off pad. Figure 6.1.3-1 is an overview of the DPF site. The facility was designed to accommodate a DSCS III class payload consisting of a DSCS III satellite and integrated apogee boost subsystem.

The facility can accommodate 9,000 kg (20 klb) of bipropellant and/or 9,000 kg (20 klb) of solid rocket motors (SRM). The DPF is partitioned into two primary operating segments. The HPF segment consists of two high bay test cells; the nonhazardous PPF segment consists of one low bay test cell plus all other test and facility operations support areas.

Each HPF bay is a Class 100,000 clean room with approximate dimensions of 15.2x15.2x16.8-m high (50x50x55-ft high). The two cells in the HPF have been assigned the following functions:

1) East Bay (EB)—Prelaunch processing of the PLF and encapsulation of the payload within the fairing. It can also be used as a fueling cell to assemble SRMs.
2) West Bay (WB)—Bipropellant loading cell that can also be used to assemble SRMs. There is no overhead crane in this room.
3) Main Bay—Intended for nonhazardous electrical and mechanical operations and integration of payload elements before fueling. Leak testing and ordnance installation may be accomplished with approval from Range Safety. The main bay is a Class 100,000 clean room. Room environment is typically maintainable at 70 ±5°F with a relative humidity of 30 to 50%.

Table 6.1.2-2 SAEF 2 is the larger of the alternate hazardous processing facilities.

Control Rooms (2)		
Floor Space	9.14x10.97 m	30x36 ft
Floor Area	100.3 m^2	1,080 ft^2
Raised Flooring	0.31 m	1 ft
Ceiling Height	2.44 m	8 ft
High Bay		
Floor Size	14.94x30.18 m	49x99 ft
Floor Area	450.9 m^2	4,851 ft^2
Clear Ceiling Height	22.56 m	74 ft
Filtration	Class 100,000	
Crane Type (Each Bay)	Bridge	
Crane Capacity	9,072 kg	10 ton
Low Bays (2)		
Floor Size (No. 1)	5.79x21.95 m	19x72 ft
Floor Area (No. 1)	127.1 m^2	1,368 ft^2
Clear Height (No. 1)	7.62 m	25 ft
Floor Size (No. 2)	5.79x8.23 m	19x27 ft
Floor Area (No. 2)	47.7 m^2	513 ft^2
Test Cell		
Floor Size	11.28x11.28 m	37x37 ft
Floor Area	127.2 m^2	1,369 ft^2
Clear Ceiling Height	15.85 m	52 ft
Door Size	6.7x12.2 m	22x40(h) ft

Table 6.1.2-3 The PHSF Building is a recent addition to NASA's hazardous processing facility capabilities.

Service Bay		
Floor Space	18.4x32.6 m	60x107 ft
Ceiling Height	28.9 m	94 ft 10 in.
Door Dimensions	10.8x22.9 m	35x75 ft
Crane Capacity	45,400 kg	50 ton
Hook Height	25.5 m	83 ft 6 in.
Airlock		
Floor Space	15.3x25.9 m	50x80 ft
Ceiling Height	27.4 m	89 ft 10 in.
Door Dimensions	10.8x22.9 m	35x75 ft
Crane Capacity	13,600 kg	15 ton
Hook Height	22.9 m	75 ft
Equipment Airlock		
Usable Space	4.1x8.0 m	14x26 ft
Ceiling Height	3.2 m	10 ft 4 in.
Door Dimensions	3.0x3.0 m	10x10 ft
Environmental Controls		
Filtration	Class 100,000	
Air Change Rate	Four per hour Minimum	
Temperature	21.7±3.3°C	71±6°F
Relative Humidity	55% Maximum	

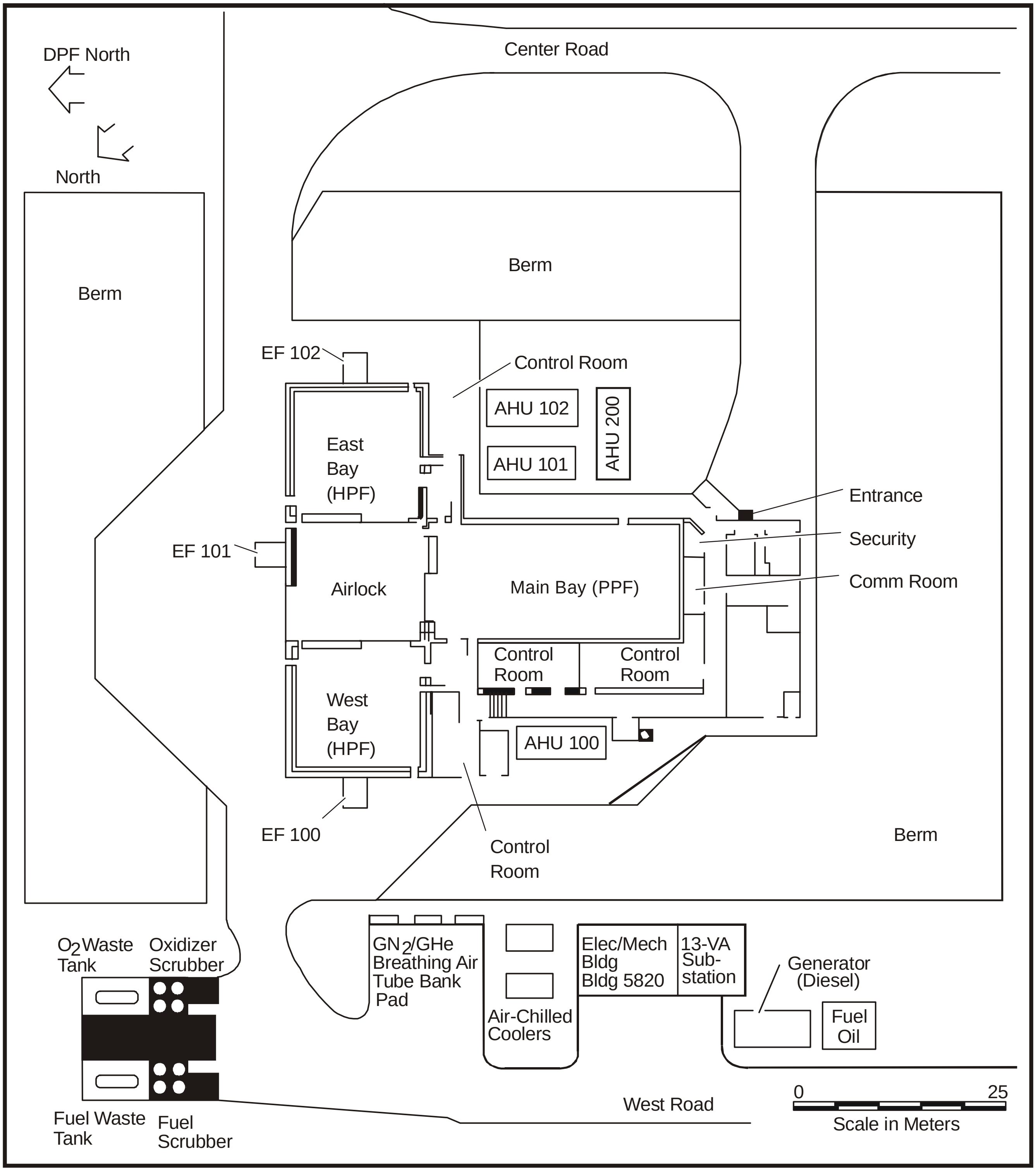


Figure 6.1.3-1 DPF Area Detail Site Plan

The bay is 30.5-m (100-ft) long north-south, approximately 15.2-m (50-ft) wide east-west, and 7.6-m (25-ft) high. It is equipped with a 4,500-kg (10,000-lb) crane with a hook height of 6.1 m (20 ft).

6.1.4 Spacecraft Instrumentation Support Facilities

CCAFS area facilities described in this section can be used for spacecraft checkout as limited by compatibility to spacecraft systems. Special arrangements and funding are required to use these assets.

TEL4 Telemetry Station—The Eastern Range (ER) operates an S-band telemetry receiving, recording, and real-time relay system on Merritt Island. This system is used for prelaunch checkout of launch vehicles and spacecraft. A typical ground checkout configuration would include a reradiating antenna at the PPF, HPF, or launch pad directed toward the TEL4 antenna. Telemetry data can be recorded on magnetic tape or routed by hardline data circuits to the spacecraft ground station for analysis. TEL4 also acts as the primary terminal for telemetry data transmitted from the ER downrange stations.

Goddard Space Flight Center (GSFC)/Merritt Island Launch Area (MILA) Station—The GSFC station is also located on Merritt Island and is the ER launch area station for NASA's Ground Spaceflight Tracking and Data Network (GSTDN) Tracking and Data Relay Satellite System (TDRSS). Included are satellite ground terminals providing access to worldwide communications. Circuits from MILA to HPF, PPF, and LC-36 are available to support checkout and network testing during prelaunch operations and spacecraft telemetry downlinking during launch and orbital operations. The MILA station can also support ground testing with TDRSS-compatible spacecraft to include TDRSS links to White Sands, New Mexico. Special arrangements and documentation are required for TDRSS testing.

Jet Propulsion Laboratory (JPL) MIL-7.1 Station—This station is colocated at MILA on Merritt Island and is an element of the JPL Deep Space Network (DSN). This station can be configured for ground tests similar to TEL4. In addition, data from spacecraft that are compatible with the DSN can be relayed to JPL in Pasadena, California.

Eastern Vehicle Checkout Facility (EVCF)/Transportable Vehicle Checkout Facility (TVCF)—The EVCF/TVCF is an Air Force Space Control Network (AFSCN) ground station. It provides an S-band interface to AFSCN resources.

6.1.5 Payload Transport to LC-36

The payload transport trailer is an air-ride suspension transporter dedicated to transporting the encapsulated spacecraft from the HPF to the launch complex. It has a self-contained gaseous nitrogen (GN_2) purge system and an environmental control system (ECS). One system is designated as prime and the other is designated backup depending on the anticipated ambient conditions during transport. These systems maintain a positive pressure of GN_2 or air in the payload fairing during the entire transport period. The transporter includes an environmental monitoring instrumentation system that provides continuous digital display of the payload fairing and ECS environment during transport.

6.1.6 Payload Transport to LC-41

The primary payload transporter is nine-axle KMAG dedicated to transporting the encapsulated spacecraft from the HPF to the launch complex. It has a self-contained GN_2 purge system and an ECS. One system is designated as prime and the other is designated backup depending on the anticipated ambient conditions during transport. These systems maintain a positive pressure of GN_2 or air in the payload fairing during the entire transport period. The transporter includes an environmental monitoring instrumentation system that provides continuous digital display of the payload fairing and ECS environment during transport.

6.2 ATLAS IIAS/III LAUNCH SITE FACILITIES

The Atlas East Coast launch facility for Atlas IIAS and III vehicle configurations is LC-36 (Fig. 6.2-1), located at CCAFS. Major facilities include mobile service towers (MST), umbilical towers (UT), and the blockhouse. Figures 6.2-2 and 6.2-3 show diagrams of LC-36A and LC-36B, respectively.

6.2.1 Mobile Service Towers

LC-36 MSTs (Figs. 6.2.1-1 and 6.2.1-2) are open steel structures with interior enclosures that contain retractable vehicle servicing and checkout levels/platforms. Primary functions of the MST are to:

1) Erect Atlas and Centaur and mate the encapsulated spacecraft.
2) Provide work areas for personnel and equipment during spacecraft mate and flight readiness checkouts.
3) Provide environmental protection to the spacecraft and launch vehicle.
4) Provide a capability for performing operational activities during the daylight, darkness, and inclement weather.

Both MSTs contain an electric, trolley-mounted overhead bridge crane used to hoist spacecraft, fairings, and the upper stage vehicle into position. The crane and the LC-36B MST are rated at 9,072 kg (10 tons). The LC-36A MST crane is rated at 18,140 kg (20 tons). Elevators serve all MST levels at LC-36A and LC-36B.

Figure 6.2-1 Launch Complex 36 (LC-36)

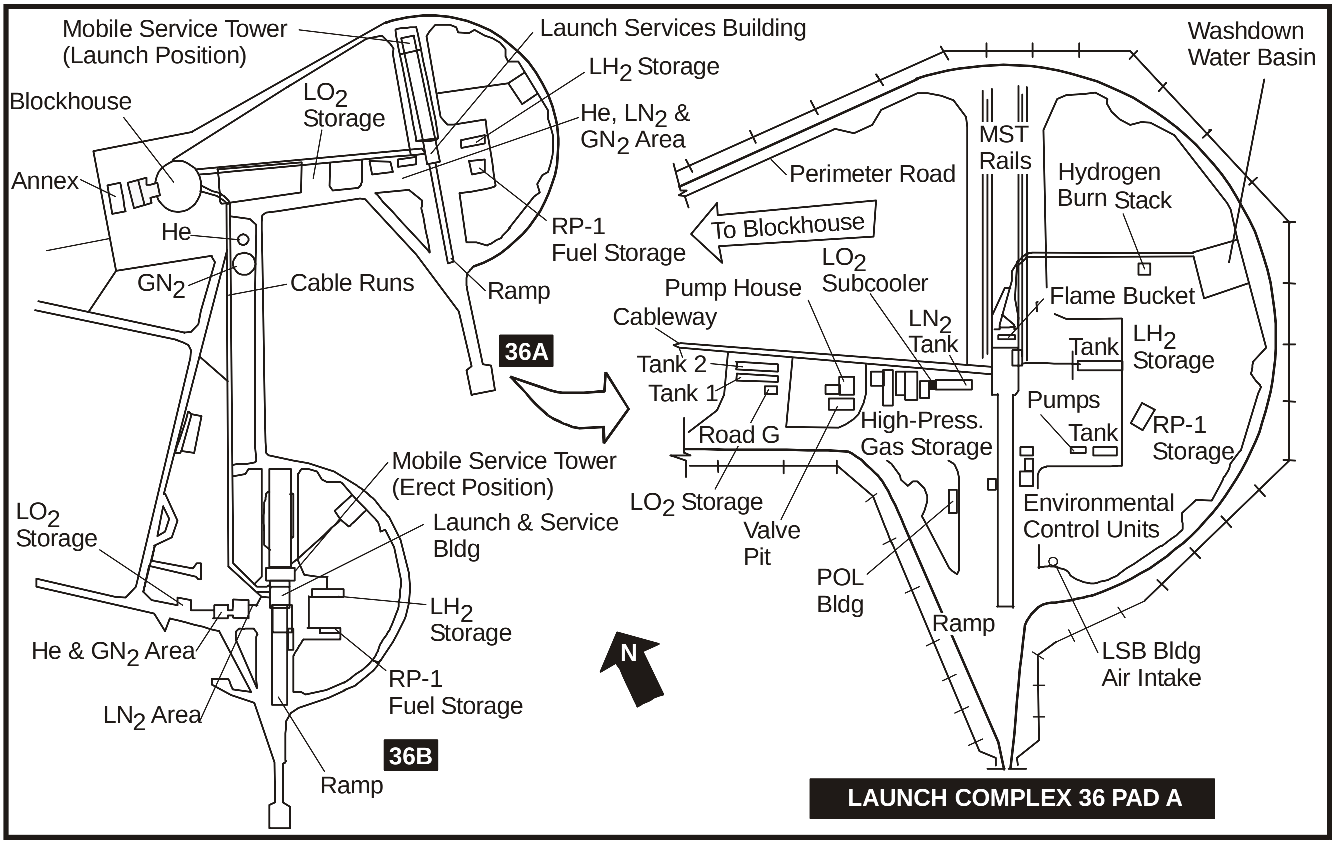


Figure 6.2-2 LC-36A at CCAFS

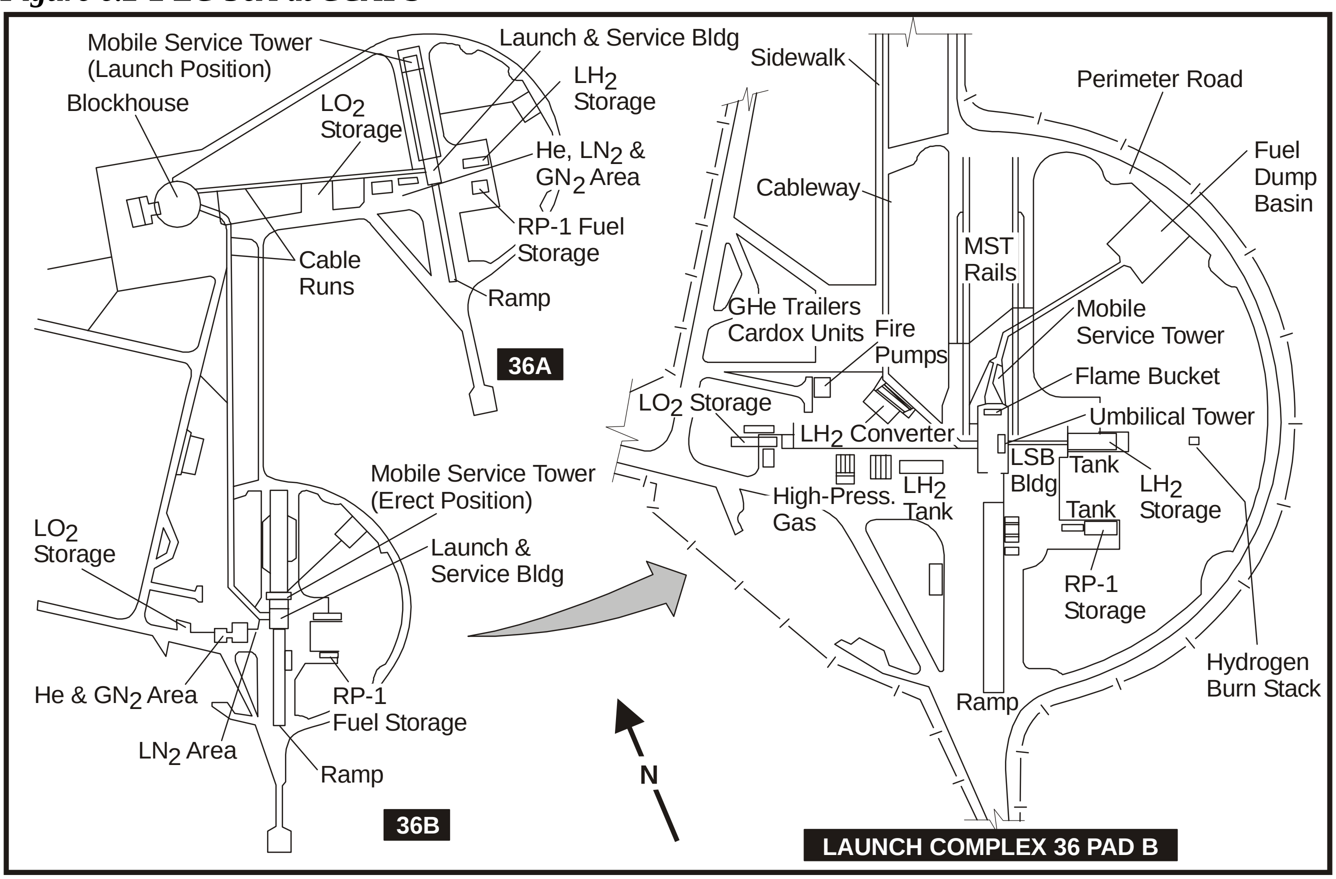


Figure 6.2-3 LC-36B at CCAFS

The entire MST assembly rests on four-wheel bogies that are electrically powered and provide service structure mobility along a rail system extending from the vehicle servicing (stand) position at the launch and service building to the stowed (launch) position.

MST platforms are arranged so that the platform above acts as ceiling of the platform below.

Access to the MST is from ground level via platforms and stairways on east and west sides of the MST; from the stairs from the launch deck; or from the umbilical mast via four crossovers at MST Levels 8, 11, 13, and 17. Staircases rise to Level 4 on the east and west sides of the MST. From Level 4, a single staircase provides access to Level 20 on the north side of the structure. Two elevators, on the east and west sides of the MST, service MST Levels 1 through 18 and platform Levels 1 through 8. Emergency egress is by stair only.

The open framework of the MST is made of structural and tubular-shaped steel, which is bonded together and grounded. All platforms, access stands, and stairs are made of metal with nonslip surfaces. Sliding door panels protect the opening on the south side of the MSTs.

Spacecraft Work Areas—Access to the encapsulated spacecraft from the LC-36B MST is provided at Levels 15 and 16 (Fig. 6.2.1-2) and at Levels 14 and 15 at the LC-36A MST (Fig. 6.2.1-1). From the LC-36B MST, Platforms 29 and 30 elevations approximately coincide with Levels 15 and 16. Platform 29 provides access to the aft area of the spacecraft via four large standard doors in medium-, large-, and extended-length large fairings. Additional access to the spacecraft can be provided from access platforms to mission-peculiar doors accessible from initial mate until fairing closeout.

Spacecraft Interface Panel—A special power panel is supplied from the critical power bus to ensure continual spacecraft support power. In case of commercial power failure, emergency power is generated at the complex by standby generator sets.

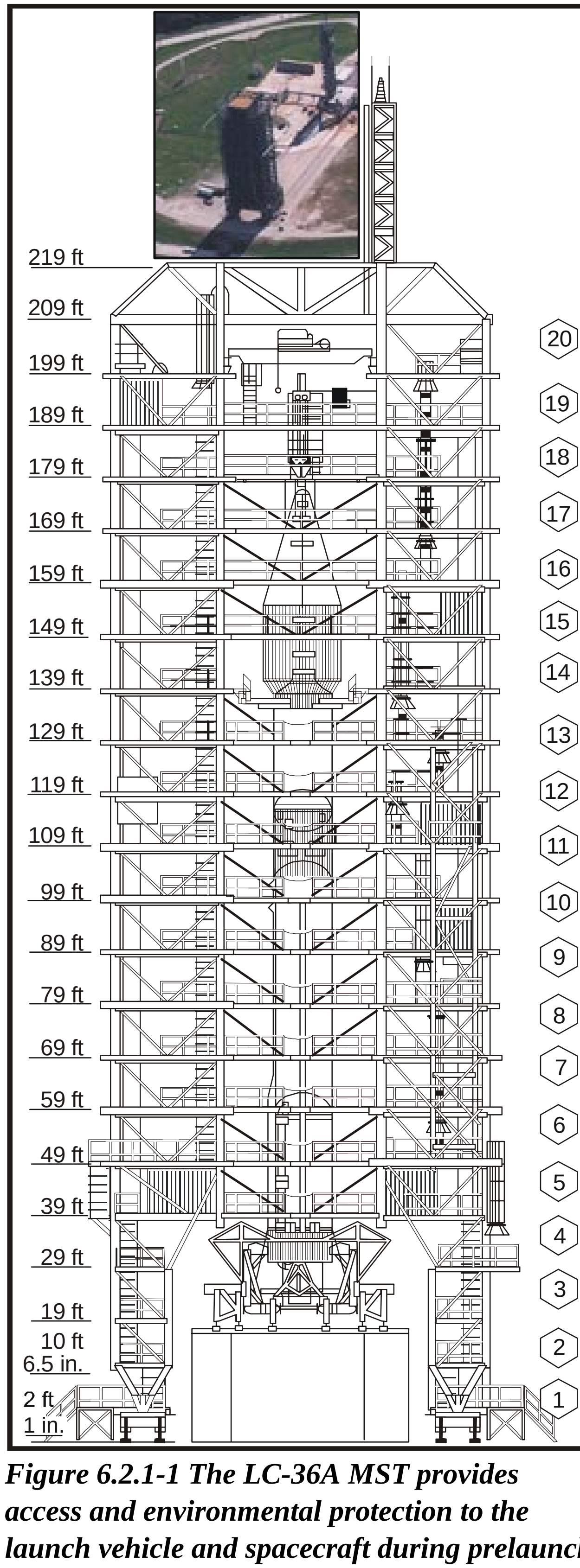


Figure 6.2.1-1 The LC-36A MST provides access and environmental protection to the launch vehicle and spacecraft during prelaunch processing.

Spacecraft Access Doors—Access to the spacecraft is provided in the service tower by doors in the PLF cylinder and through standard doors in the PLF boattail. During payload integration

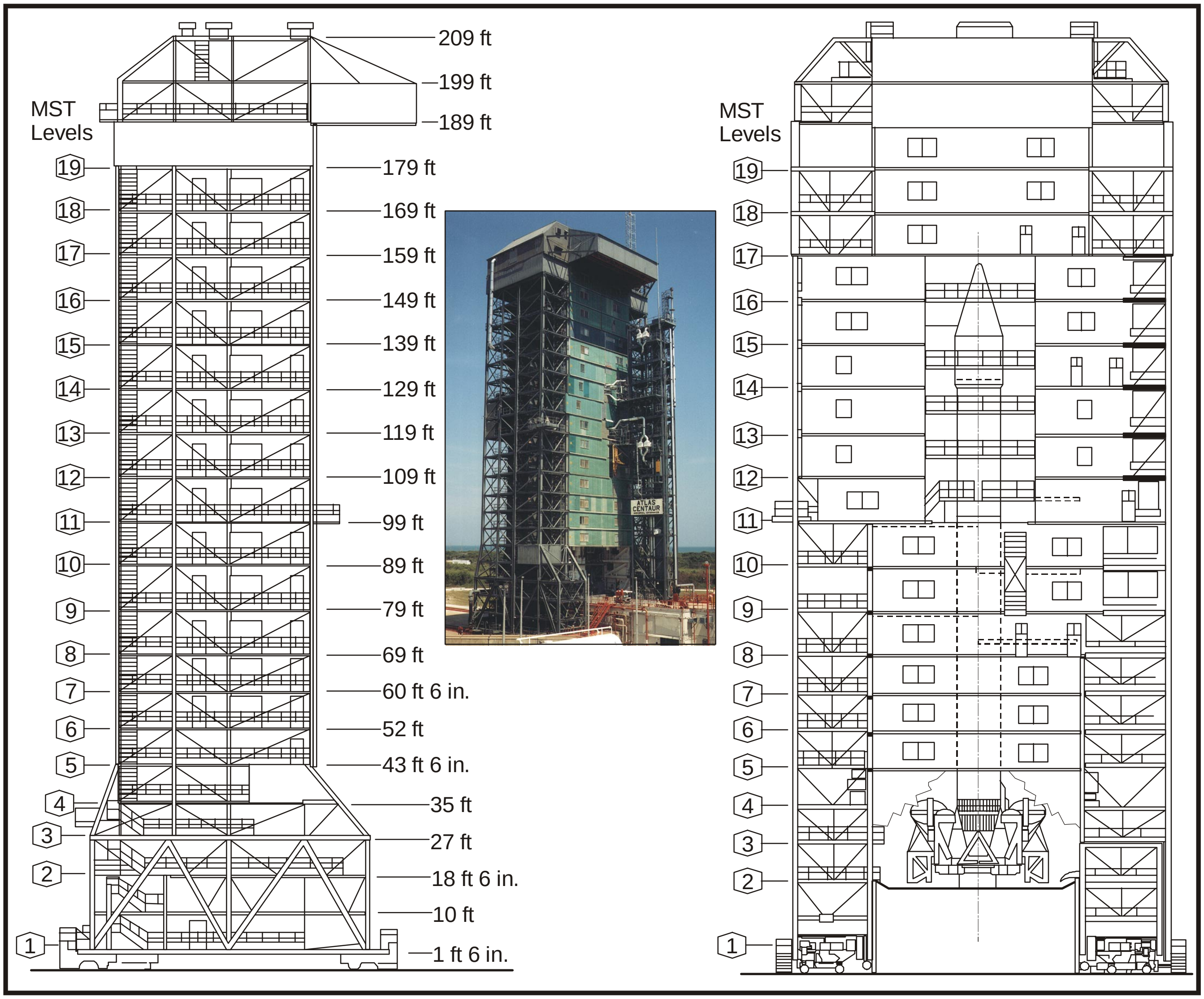


Figure 6.2.1-2 The LC-36B MST

operations, GSE doors are used when access to the spacecraft is not required. GSE doors are typically replaced with flight doors on Launch-1 day. A GSE adapter frame is used to limit use of the flight hardware. GSE doors provide contamination and electromagnetic interference (EMI) shielding (Fig. 6.2.1-3).

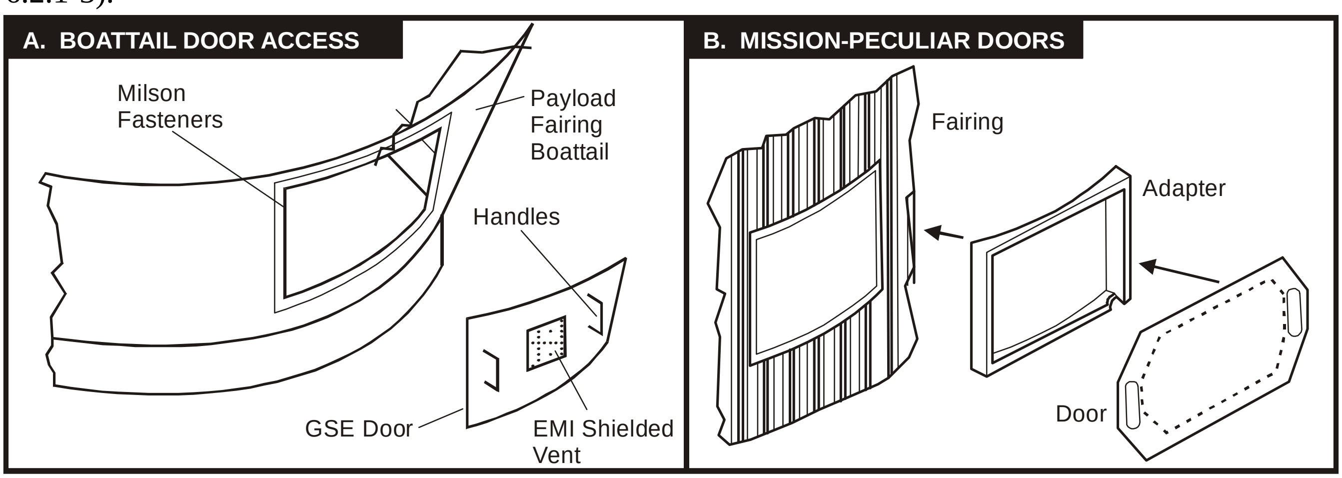


Figure 6.2.1-3 Spacecraft access at LC-36 is provided through 4-m boattail and mission-peculiar doors.

Spacecraft Access Platforms—Access through PLF cylinder doors may be provided by two portable access platforms. The type used depends on the required reach from the PLF outer skin to the spacecraft item being serviced. One platform allows the technician access into the PLF cavity, is portable (Fig. 6.2.1-4), but is secured to the service tower deck when in use and during storage. Access through the PLF boattail doors is provided by portable platforms shown in Figure 6.2.1-5.

Communications—Four types of communications equipment are installed on the MST: voice, CCTV, RF, and a public address (PA) system.

Blackphones and hazardous TOPS units comprise the voice communications system. TOPS units are located on all work levels. Blackphones are installed on spacecraft work levels.

CCTV cameras are installed on MST Level 29 to monitor launch vehicle and spacecraft processing activities.

S-band and Ku-band equipment may be installed on the MST per mission-peculiar requirements of the launch vehicle and/or spacecraft programs. RF cabling and reradiating antennas are available for S-band, C-band, and Ku-band telemetry.

PA speakers are installed inside and outside the enclosure. The PA system is the primary means of alerting personnel and making routine announcements. Three PA microphone stations are available. The primary station is in the blockhouse, secondary stations are in and on the deck of the Launch Services Building (LSB).

Vehicle Restraint System—LC-36 MSTs have a vehicle restraint system that enables the mated nose fairing/launch vehicle stack to be placed into a secure “stretch” configuration during adverse weather conditions. This system is designed to withstand 240-kph (150-mph) winds.

6.2.2 Umbilical Tower (UT)

The UT (Fig. 6.2.2-1) is a fixed structural steel tower extending above the launch pad. Retractable service booms are attached to the UT. The booms provide electrical power, instrumentation, propellants, pneumatics, and conditioned air or GN_2 to the vehicle and spacecraft. These systems also provide

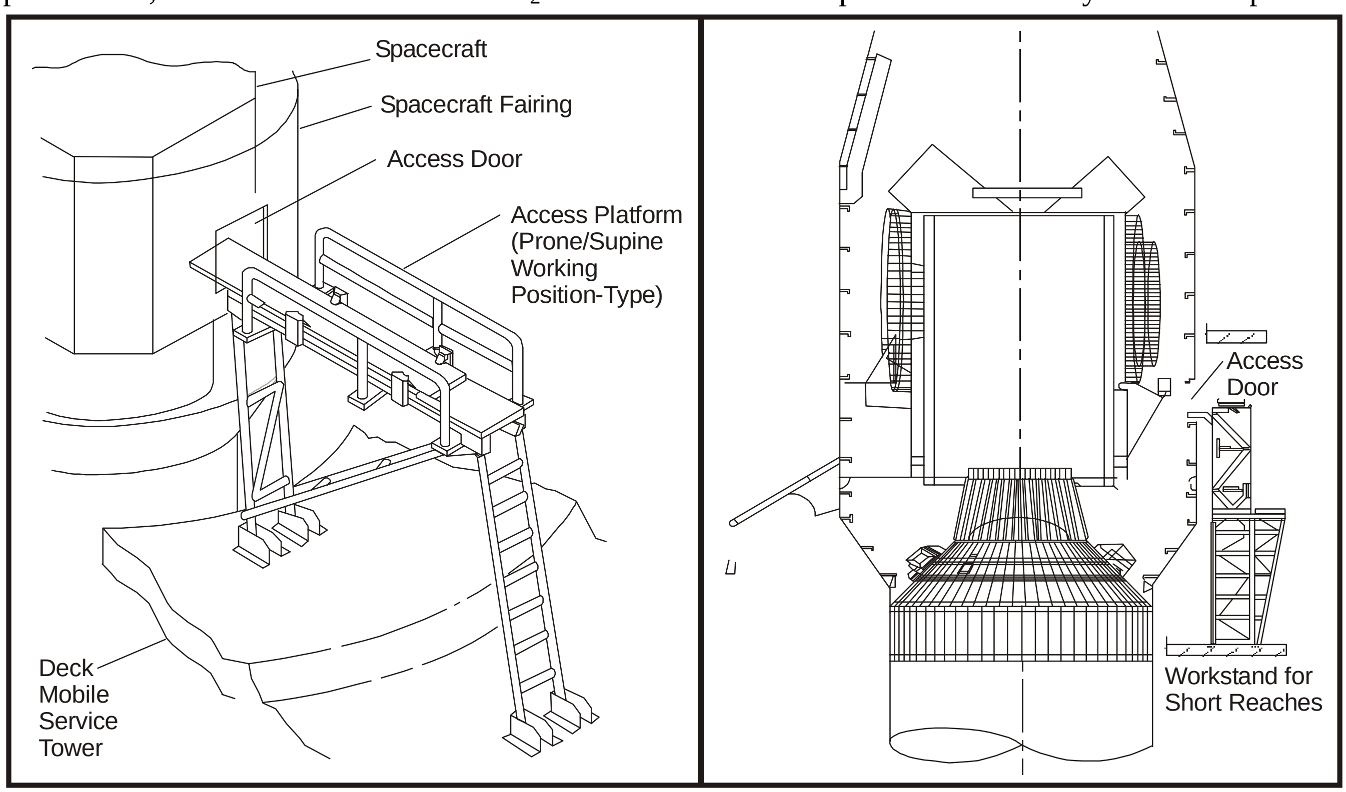


Figure 6.2.1-4 Mobile platforms provide spacecraft access if required.

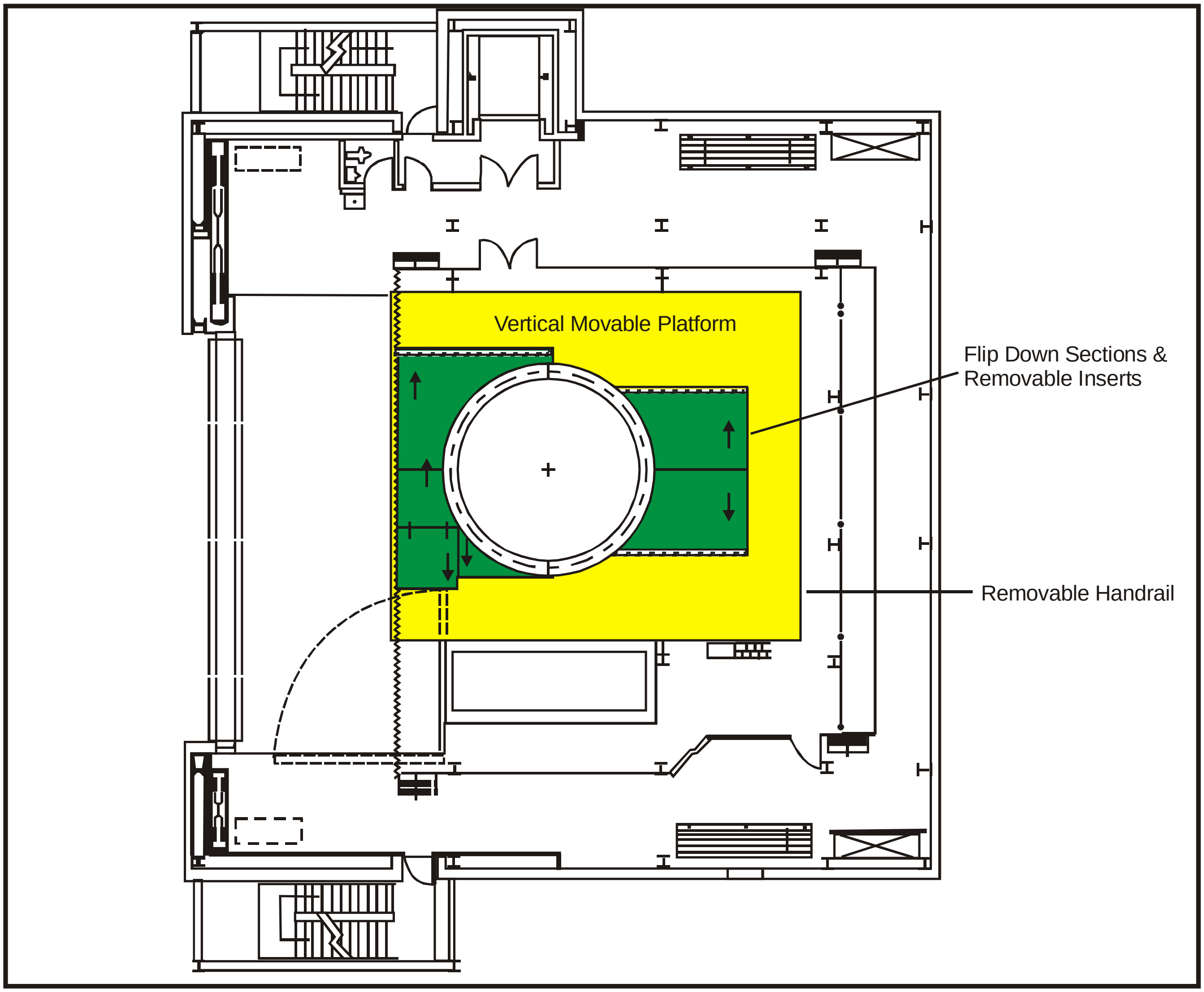


Figure 6.2.1-5 MST Movable Platforms (Typical)

quick-disconnect mechanisms at the respective vehicle interface and permit boom retraction at vehicle launch. A payload umbilical junction box interconnects the spacecraft to the electrical GSE. Limited space is available within this junction box to install spacecraft-unique electrical GSE.

6.2.3 Launch Pad Ground System Elements

The launch complex is serviced by GN_2, gaseous helium (GHe), and propellant storage facilities within the complex area. ECSs exist for the launch vehicle and the spacecraft. Detailed descriptions of capabilities of these systems to provide spacecraft activity are in Section 4.2, Spacecraft-to-Ground Equipment Interfaces.

6.2.4 Blockhouse

The blockhouse (Figs. 6.2.4-1 and 6.2.4-2) is the operations and communications center for the launch complex. It contains all necessary control and monitoring equipment. The launch control, electrical, landline instrumentation, and ground computer systems are the major systems in this facility.

The launch control provides consoles and cabling for control of the launch complex systems. The landline instrumentation system (coupled with the CCTV system) monitors and records safety and performance data during test and launch operations. The ground computer system consists of redundant computer-controlled launch sets (CCLS) and a telemetry ground station. The CCLS provides control and monitoring of the vehicle guidance, navigation, and control systems and monitors vehicle instrumentation for potential anomalies during test and launch operations.

6.2.5 Launch Services Building (LSB)

The LSB provides the means for erecting the launch vehicle, interconnecting electrical wiring between the umbilical tower and the blockhouse, and locating spacecraft remote GSE. It also provides the launching platform. The LSB is a two-story structure (Fig. 6.2.5-1). LSBs at LC-36A and LC-36B are similar in function.

6.2.6 Customer Support Building

Lockheed Martin provides an Atlas launch services customer technical support building (Fig. 6.2.6-1). This building includes private office space, workspace, and conference rooms for customer management during spacecraft launch site processing. Reproduction machines, administrative telephones with long-distance access, and facsimile are available. The facility is conveniently located between the Astrotech and LC-36 facilities.

6.2.7 Recent LC-36 Enhancements

LC-36A—Launch Complex 36A has been upgraded to support Atlas IIAS launches. Modifications included a new launcher structure and an enhanced holddown release system (pyrovent valve instead of the current Barksdale valve).

LC-36B—Launch Complex 36B has been modified to support the Atlas III vehicle configuration. Modifications included a MST height increase of 6.1 m (20 ft), a new 20-ton bridge crane, UT changes for a taller Atlas, and launcher modifications for the new III vehicle to ground interfaces.

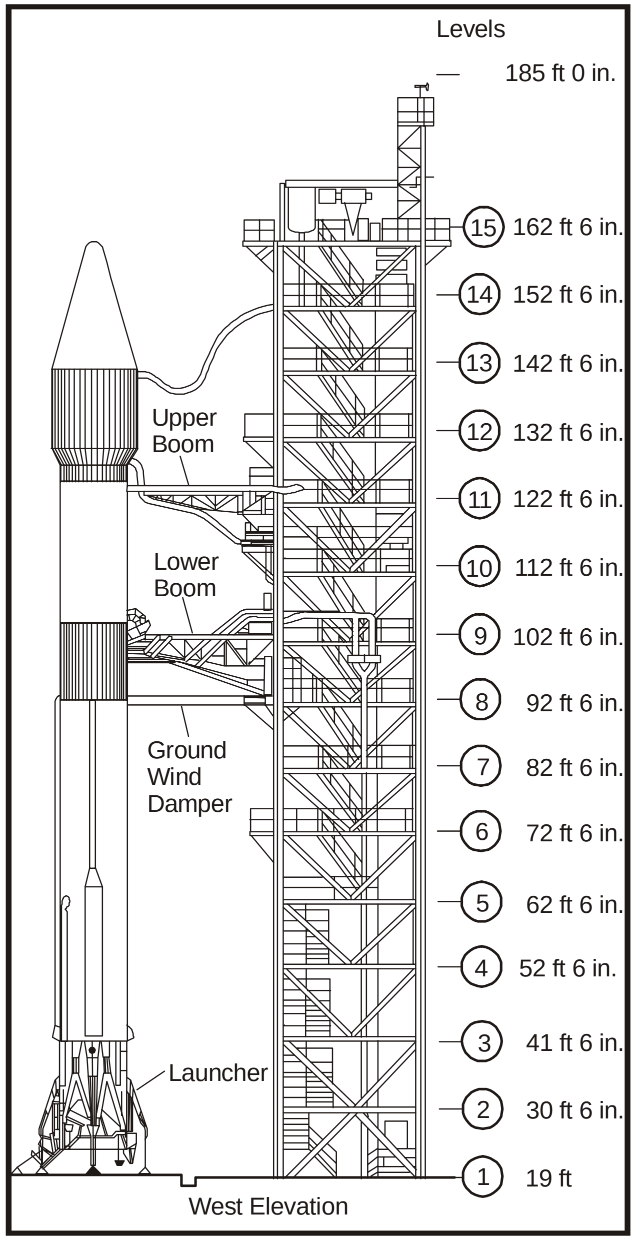


Figure 6.2.2-1 The LC-36B umbilical tower retractable service arms allow quick disconnect and boom retraction at vehicle launch.

6.3 ATLAS V LAUNCH SITE FACILITIES

In addition to the encapsulation facility (Section 6.1), Atlas V uses three primary facilities and integrated GSE that support spacecraft processing. The following section describes LC-41 consisting of the Vertical Integration Facility (VIF) and the launch pad, the Atlas V Spaceflight Operations Center (ASOC), the mobile launch platform (MLP), and payload van (PVan).

6.3.1 Vertical Integration Facility (VIF)

The VIF is a weather-enclosed steel structure, approximately 22.9-m (75-ft) square and 87.2-m (286-ft) tall, with a fabric roll-up door, a hammerhead bridge crane, platforms, and servicing provisions required for launch vehicle integration and checkout (Fig. 6.3.1-1). Launch vehicle processing in the VIF includes stacking booster(s) and upper stage, performing launch vehicle subsystem checks and

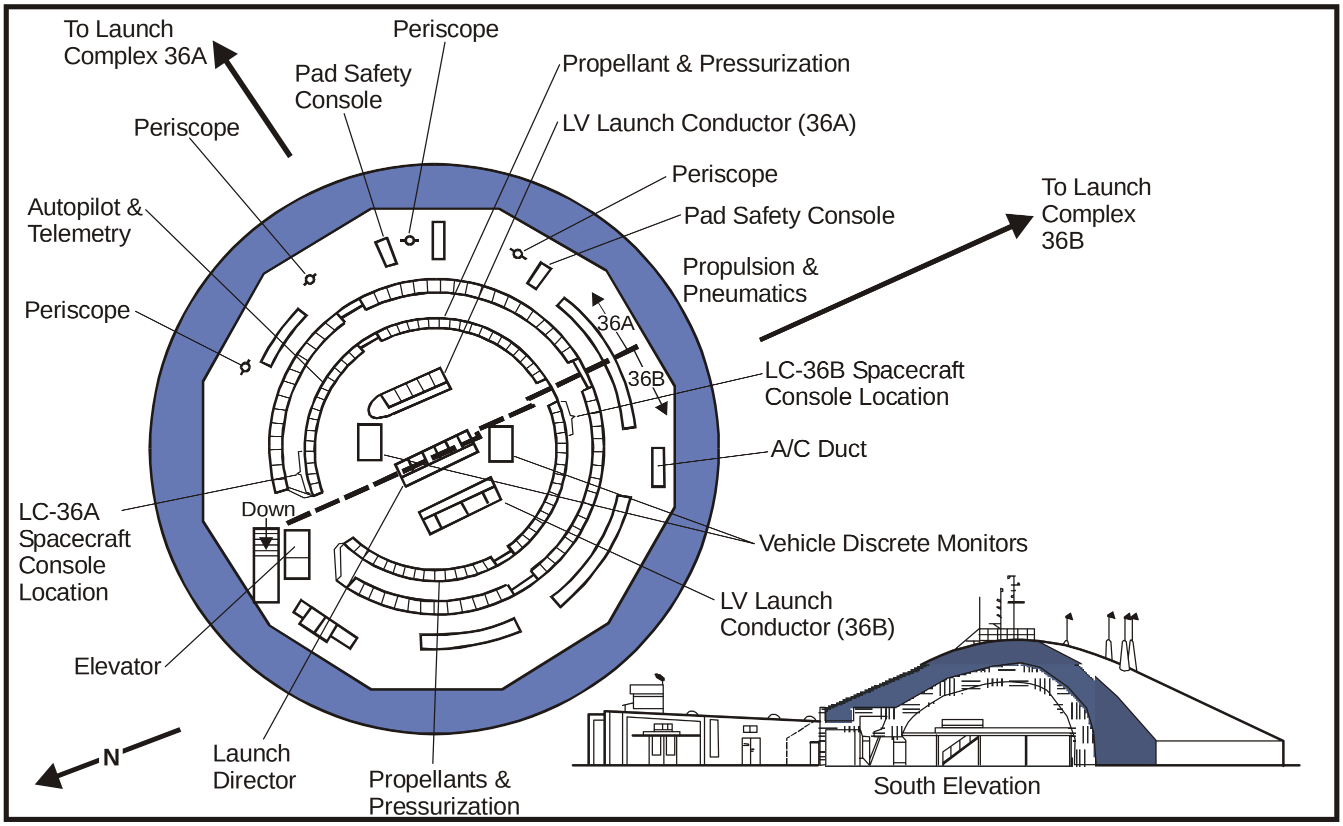


Figure 6.2.4-1 Test and launch operations are controlled and monitored from the blockhouse.

Figure 6.2.4-2 Aerial View of Common LC-36 Blockhouse

system verification, installing the encapsulated spacecraft, performing integrated system verification, final installations, and vehicle closeouts.

No provisions for spacecraft propellant loading are available at the launch complex, either on-pad or in the VIF. The spacecraft is fueled at the payload processing facility before encapsulation. If required, emergency spacecraft detanking on the launch complex will occur in the VIF.

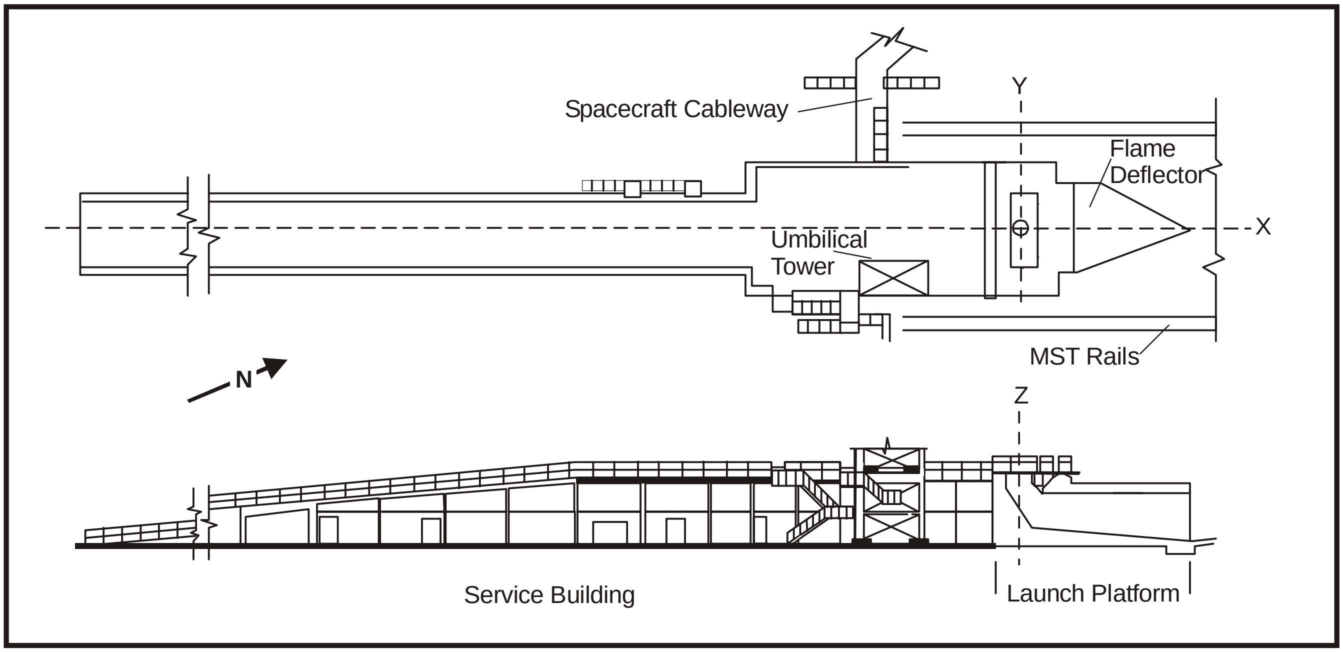


Figure 6.2.5-1 LC-36 Launch and Service Building (LSB)

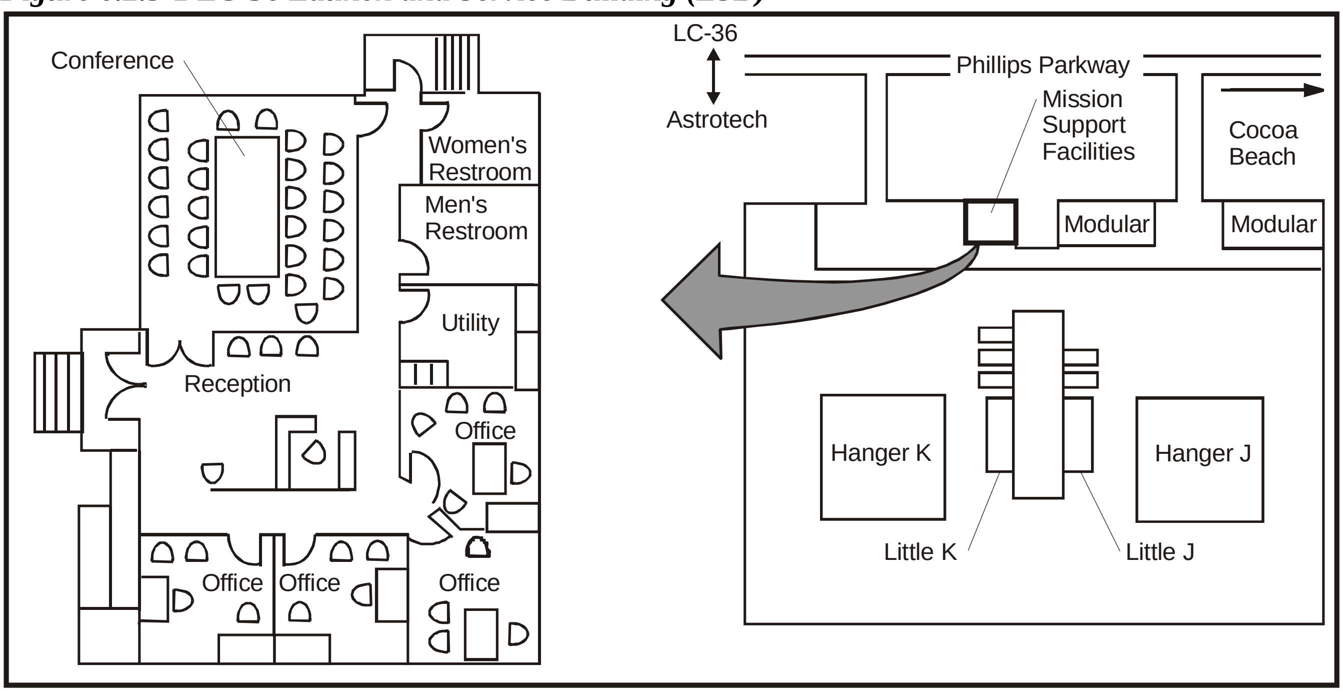


Figure 6.2.6-1 CCAFS Launch Services Customer Support Facility

The facility includes two stairways and one freight elevator for access to the platforms. Elevator capacity is 1,820 kg (4,000 lb), with maximum cargo dimensions of 1.7-m (5-ft 7-in.) width, 2.4-m (7-ft 9-in.) length, and 2.1-m (7-ft) height.

Spacecraft access through the PLF doors occurs on VIF Levels 5 to 7, depending on the launch vehicle and spacecraft configuration. Access to the spacecraft, if required, is provided using portable access stands. Class 5,000 conditioned air and localized GN_2 purges (if required) are provided to the PLF to maintain the spacecraft environment. Spacecraft conditioned air cleanliness, temperature, humidity and flow rate are as follows:

1) Cleanliness Class 5,000;
2) Temperature 10-29°C (50-85°F), selectable with tolerance ±2.8°C (±5°F);

3) Relative humidity 20-50%, maximum dewpoint 4.4°C (40°F) and 35-50% when required for sensitive operations;
4) Flow rate 0.38-1.21 kg/s (50-160 lb/min) (400 series), 0.38-2.27 kg/s (50-300 lb/min) (500 series), selectable with tolerance ±0.038 kg/s (±5 lb/min) (400 series) and 0.095 kg/s (±12.5 lb/min) (500 series).

Accommodations for spacecraft personnel and test equipment are provided on VIF Levels 5, 6, 6.5, and 7. Table 6.3.1-1 identifies power, commodities, and communications interfaces on these levels. Spacecraft customers are encouraged to contact Lockheed Martin for further details and capabilities.

The 54,000-kg (60-ton) facility crane used for lifting and mating the encapsulated spacecraft has creep speed controls down to 30 mm/min (1.2 in./min) on the bridge and trolley, and 12.5 mm/min (0.5 in./min) on hoist, with maximum acceleration/deceleration of 75 mm/s^2 (0.25 ft/s^2).

Figure 6.3.1-1 Vertical Integration Facility

Payload critical power (redundant facility power) and facility power (120 V) are provided to support spacecraft operations in the VIF. Communications on spacecraft access levels in the VIF include unsecure operational voice system, telephone, public address, user interfaces, and secure operational voice system. Lighting to 50 fc is provided on spacecraft access levels in the VIF. If it should be necessary to perform an emergency spacecraft propellant detank, the VIF includes fuel and oxidizer drains and vent lines, emergency propellant catch tanks, hazardous vapor exhaust, and breathing air support.

6.3.2 Launch Pad

The Atlas V launch pad is within LC-41 (Fig. 6.3.2-1) at CCAFS. The launch complex uses a “clean-pad” launch processing approach whereby the launch vehicle is fully integrated off-pad on the MLP in the VIF, and the launch pad is used only for launch-day propellant loads and launch countdown. There are no provisions for spacecraft access at the pad. (All final spacecraft access activities, including removal of ordnance safe and arm devices, are made in the VIF.) All spacecraft umbilicals needed at the pad are flyaway disconnects. Rollback from the pad to the VIF can be accomplished in 6 hours if launch vehicle propellants have not been loaded, and within 18 hours if launch vehicle propellants must be detanked.

The launch pad continues to provide MLP, and PVan interfaces as described in the following paragraphs. Payload compartment conditioned air is provided to the same parameters provided in the VIF. Spacecraft conditioned air is switched to GN_2 approximately 1 hour before upper-stage cryogenic propellant operations. GN_2 cleanliness, temperature, and flow rate are the same as conditioned air, with maximum dewpoint of -37°C (-35°F).

6.3.3 Atlas V Spaceflight Operations Center

The ASOC, located approximately 4 miles from LC-41, is a multifunctional facility supporting hardware receipt and inspection, horizontal testing, and launch control. The Launch Operations Center

Table 6.3.1-1 Vertical Integration Facility Payload Accommodations

Accommodation	Description	Qty
VIF Level 5 (Payload Mate)		
Payload Critical Power	120V, 20A, 60 Hz, 1Ø, 3W, 2P, Explosion Proof	3
	120/208V, 30A, 60 Hz, 3Ø, 5W, 4P, Explosion Proof	1
	120/208V, 60A, 60 Hz, 3Ø, 5W, 4P, Explosion Proof	1
Shop Air Outlets	GN_2 Service	2
Fuel & Oxidizer Lines	Emergency Propellant Detanking	1 Set
LMAO/Customer Work Area	Desk/Phone/User Interface	2
User Interface	Hazardous, Fiber Optic	2
Telephone		7
Facility Ground Plate		2
Technical Ground Plate		3
VIF Level 6 (Payload Access Through PLF Doors)		
Payload Critical Power	120V, 20A, 60 Hz, 1Ø, 3W, 2P, Explosion Proof	2
Shop Air Outlets	GN_2 Service	2
Fuel & Oxidizer Lines	Emergency Propellant Detanking	1 Set
Customer Work Area	Desk/Phone/User Interface	2
User Interface	1 Each Nonhazardous, 2 Each Hazardous, Fiber Optic	3
Telephone		7
Facility Ground Plate		2
Technical Ground Plate		2
VIF Level 7 (Access to Top of PLF)		
Payload Critical Power	120V, 20A, 60 Hz, 1Ø, 3W, 2P, Explosion Proof	4
Shop Air Outlets	GN_2 Service	2
Fuel & Oxidizer Lines	Emergency Propellant Detanking	1 Set
Customer Work Area	Desk/Phone/User Interface	2
User Interface	2 Each Nonhazardous, 2 Each Hazardous, Fiber Optic	4
Telephone		6
Facility Ground Plate		3
Technical Ground Plate		3
Freight Elevator		
Size	Length x Width: 2,438.4 mm x 1,828.8 mm (8 ft x 6 ft)	
Door Clearance	Width x Height: 1,828.8 mm x 2,133.6 mm (6 ft x 6 ft 11 in.)	
Capacity	2,267.9 kg (5,000 lb)	

(LOC) in the ASOC (Figs. 6.3.3-1 and 6.3.3-2) provides spacecraft interfaces for command, control, monitoring, readiness reviews, anomaly resolution, office areas, and launch viewing. The LOC design provides maximum flexibility to support varying customer requirements.

Spacecraft Control Center (SCC)—If desired the SCC can be used to provide connectivity between spacecraft EGSE located in the SCC and the PVan. The room includes 30 m^2 (323 ft^2), raised computer flooring; launch control console with voice, video, and phone; access to the Launch Control Center (LCC); and a private entrance.

Spacecraft Operations Center (SOC)—The SOC provides five each spacecraft customer and one each LMAO position. Each position includes access to launch vehicle and spacecraft data, voice, and video. The stadium seating provides excellent viewing of the LCC video wall.

Mission Operations Center (MOC)—The MOC provides eight each spacecraft customer positions combined with the LMAO launch management team. Each position includes access to launch vehicle data, voice, and video. The stadium seating provides excellent viewing of the LCC video wall.

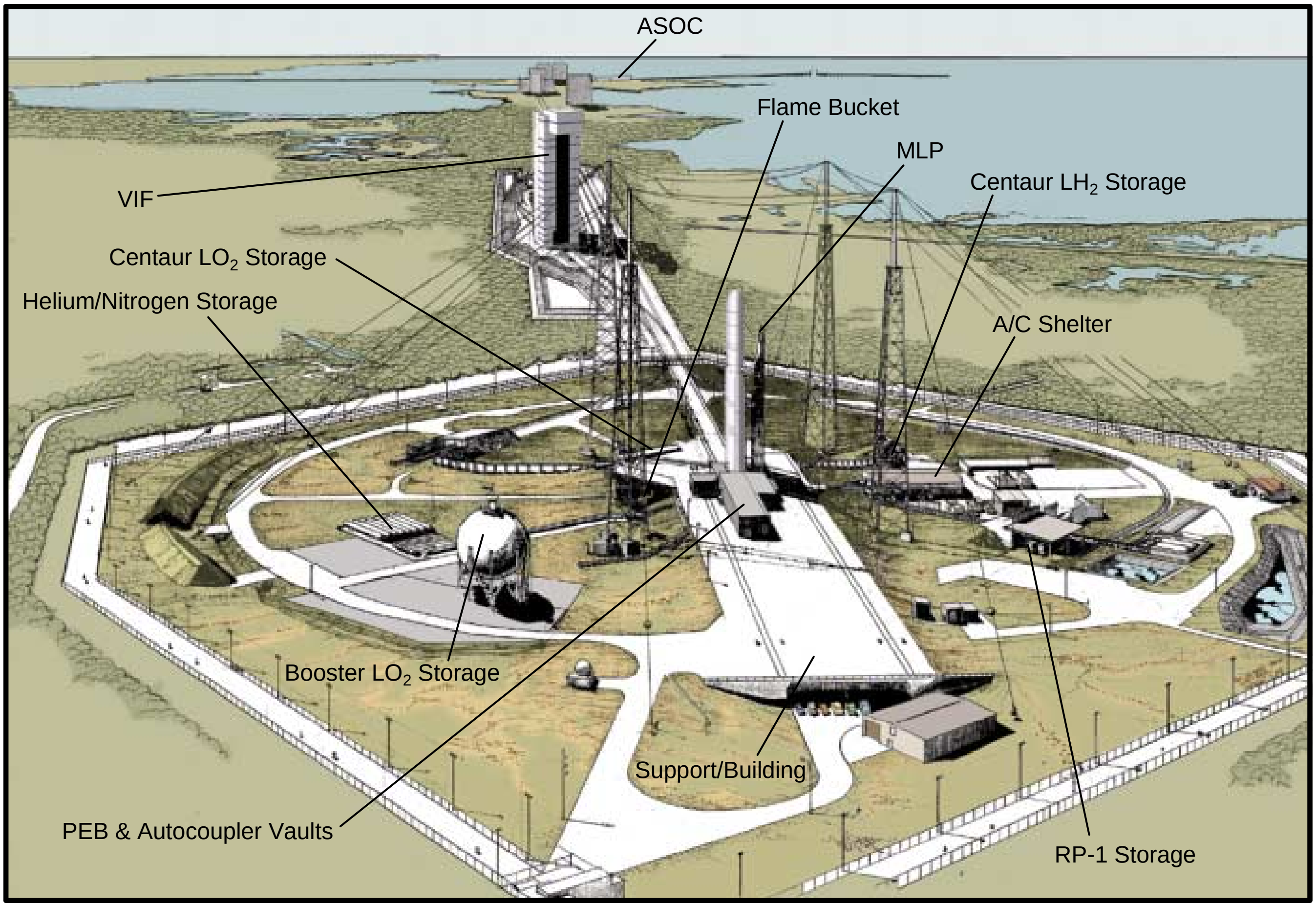


Figure 6.3.2-1 Launch Complex 41

Engineering Operations Center (EOC)—The EOC provides two each spacecraft customer positions combined with eighteen each LMAO and LMAO subcontractor launch vehicle engineers. Each position includes access to launch vehicle data, voice, and video. The stadium seating provides excellent viewing of the LCC video wall.

Spacecraft Customer Support Center (CSC)—The CSC is a multifunctional room that can be configured to support data, voice, and video, or be used for other spacecraft specific needs. The room includes 41.8 m^2 (450 ft^2) and provides a view of the LCC and video wall.

Customer Support Facility (CSF)—The CSF provides office space, briefing and anomaly resolution areas, data, voice, and video, hospitality area, and launch viewing. The CSF includes easy access to other areas within the LOC.

6.3.4 Payload Support Van (PVan)

The PVan (Fig. 6.3.4-1) provides electrical, gas, and communication interfaces between the spacecraft support equipment and the spacecraft, initially at the VIF for prelaunch testing, and subsequently at the pad during launch. The PVan consists of a rail car undercarriage and support container that houses the spacecraft ground support equipment. The PVan provides 23.2 m^2 (250 ft^2) of floor space for spacecraft mechanical, electrical, and support equipment. The PVan also provides power, air conditioning, lighting, and environmental protection.

The PVan provides the electrical interface between the spacecraft ground support equipment and the Atlas V T-0 umbilical that supplies the ground electrical services to the spacecraft. The PVan provides 20-kVA UPS power for spacecraft GSE racks. Power receptacles provided include eight 120-Vac, 15-A receptacles; two 120-Vac, 30-A receptacles; and four 120/208-Vac, three-phase, four-pole,

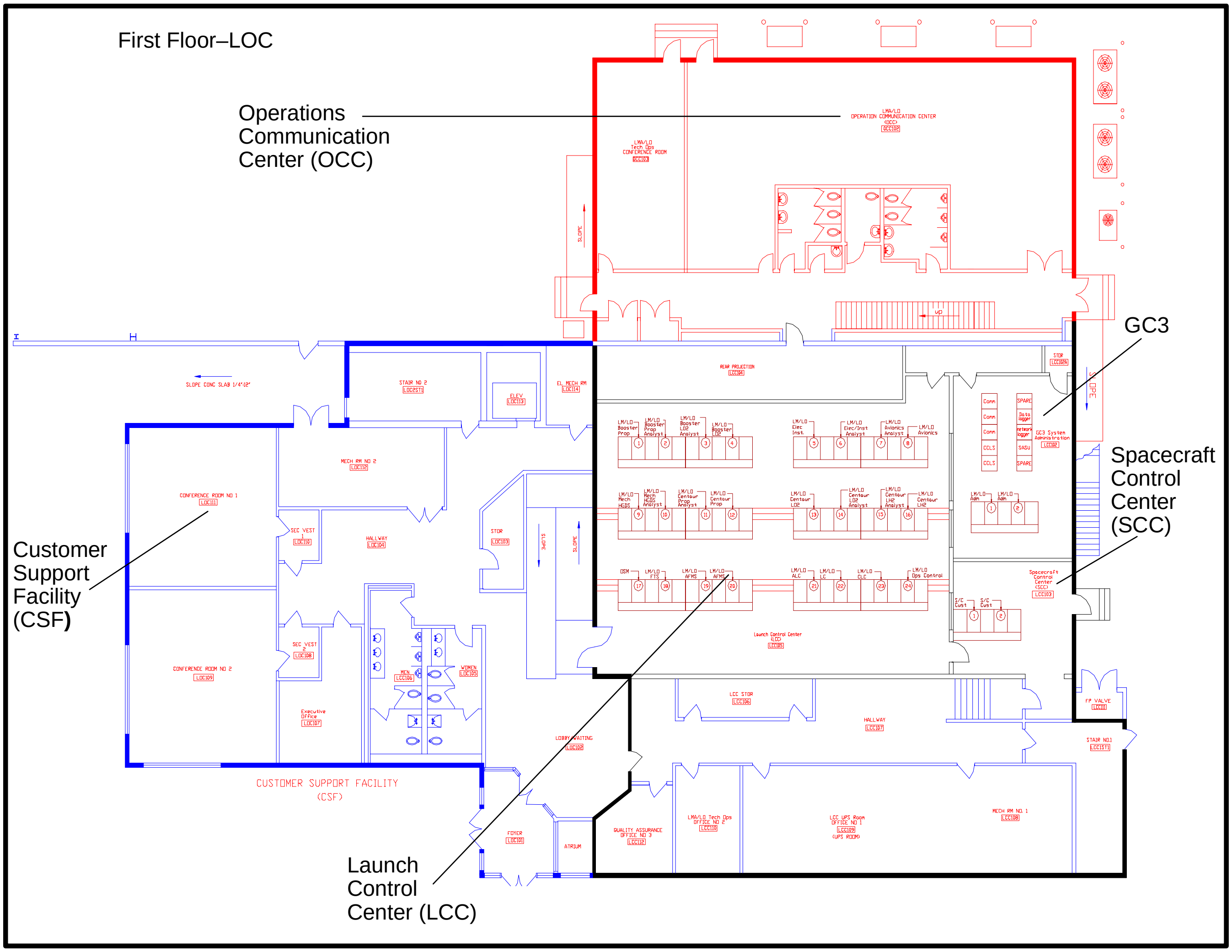


Figure 6.3.3-1 ASOC Launch Operations Center First Floor

five-wire receptacles. The PVan provides the supply and interface through the T-0 umbilical for Grade B or Grade C GN_2 at a flow rate up to 14.2 scmh (500 scfm) for spacecraft instrument purge.

The Atlas V communication system provides spacecraft communication connectivity from spacecraft ground support equipment in the PVan to the Atlas V fiber-optic network. The communication network (Fig. 6.3.4-2) provides interfaces to the spacecraft remote command and control station located at the ASOC or Astrotech. Additional remote spacecraft processing sites may also access spacecraft data by precoordinating with the Range for connectivity to the ASOC. Spacecraft RF communications are routed from the PLF RF window or reradiating antenna to the PVan and then through the Atlas V fiber-optic network. Options are also available for spacecraft to radiate directly from the RF window to the PPF. This connection is available while in the VIF or at the pad. During the move from the VIF to the pad, the spacecraft RF signal is available at the PVan for local spacecraft monitoring and recording only. The Atlas V system provides spacecraft RF uplink and downlink capability at the VIF and at the pad. During flight, spacecraft data can be interleaved with the LV telemetry stream.

The PVan is air-conditioned to maintain 69-77°F and 35-70% relatively humidity, assuming a spacecraft support equipment heat load of 21,000 Btu per hour. The PVan provides interior lighting of 50 fc in the personnel work area. The PVan limits dynamic loads to 1.5 g during transportation. When the PVan is housed within the Pad Equipment Building (PEB), spacecraft support equipment is

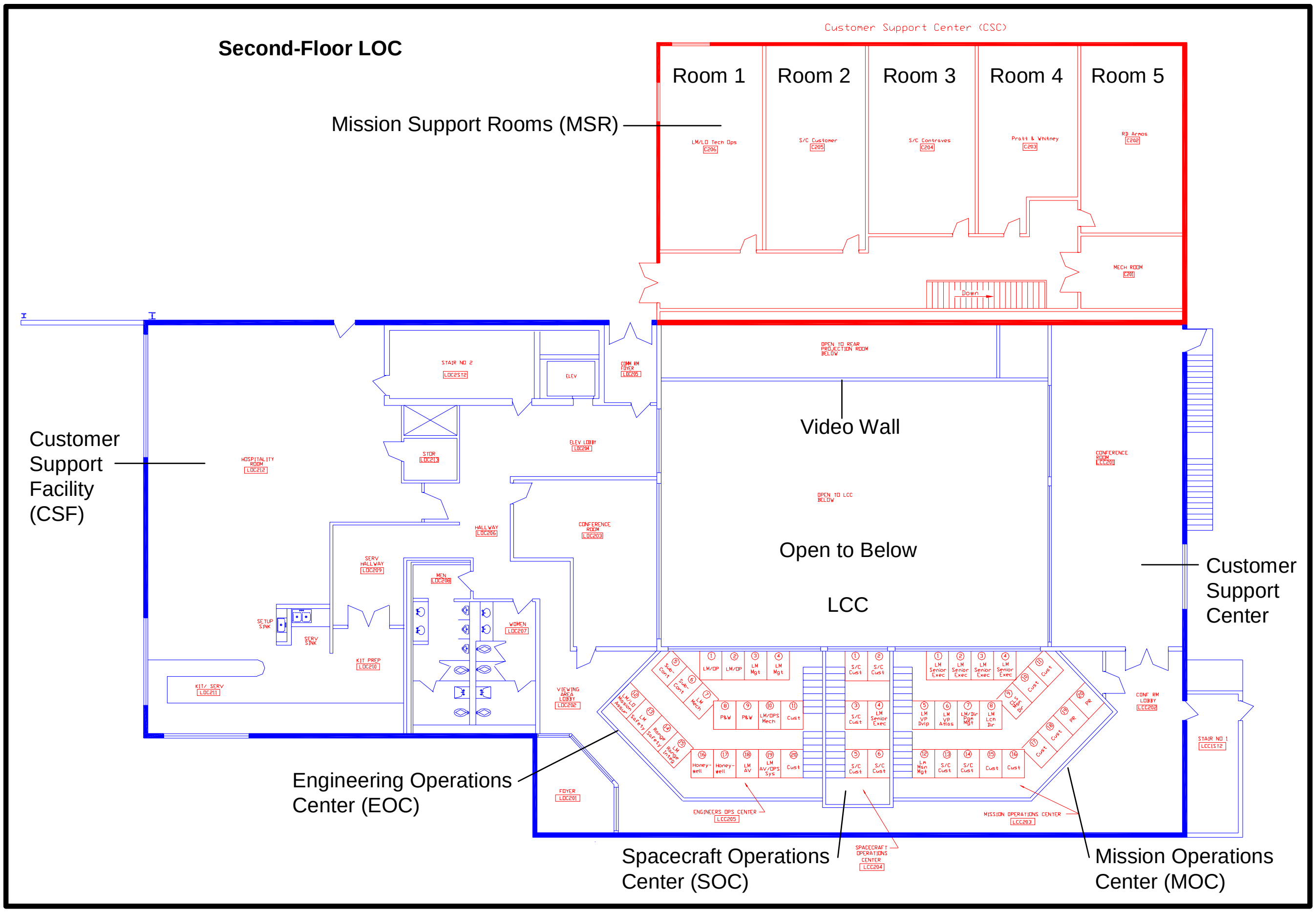


Figure 6.3.3-2 ASOC Launch Operations Center Second Floor

protected from the launch-induced environment, including overpressure, acoustics (<110 db), and thermal.

The PVan can be staffed during operations at the VIF, during transit to the pad, and on pad until final pad clear operations. Van reconfiguration for each launch consists of transporting the PVan from LC-41 to the VIF, removing the previous users equipment, performing baseline electrical checkout, and installing the spacecraft support equipment for the current user.

6.3.5 Mobile Launch Platform (MLP)

The MLP (Fig. 6.3.5-1) consists of a structural steel frame capable of supporting the Atlas V 400 and 500 series configurations during integration of the booster(s), upper stage, and spacecraft in the VIF, during transport to the launch pad, during launch vehicle fueling and final preparation for launch, and during thrust holddown and release of the launch vehicle at launch. This frame is supported underneath by piers at the VIF and at the launch pad. The frame is rolled to these locations using four 227,000-kg (250-ton) rail cars equipped with a hydraulic jacking system for raising the MLP for movement and lowering onto the piers for stability. The MLP is moved between the VIF and launch pad by two tugs that ride on a rail system. The MLP frame also supports the umbilical mast.

Mobile Launch Platform Umbilical Mast—The Atlas V MLP includes an umbilical mast for electrical, fluids, and gas servicing during final countdown, which eliminates the need for an on-pad umbilical tower. All umbilical interfaces are connected and checked out in the VIF before rollout to the pad. The T-0 umbilicals remain connected up to launch.

During transit from the VIF to the launch pad, launch vehicle and spacecraft conditioned air is provided using an air conditioning trailer connected to the MLP. The MLP includes common ductwork

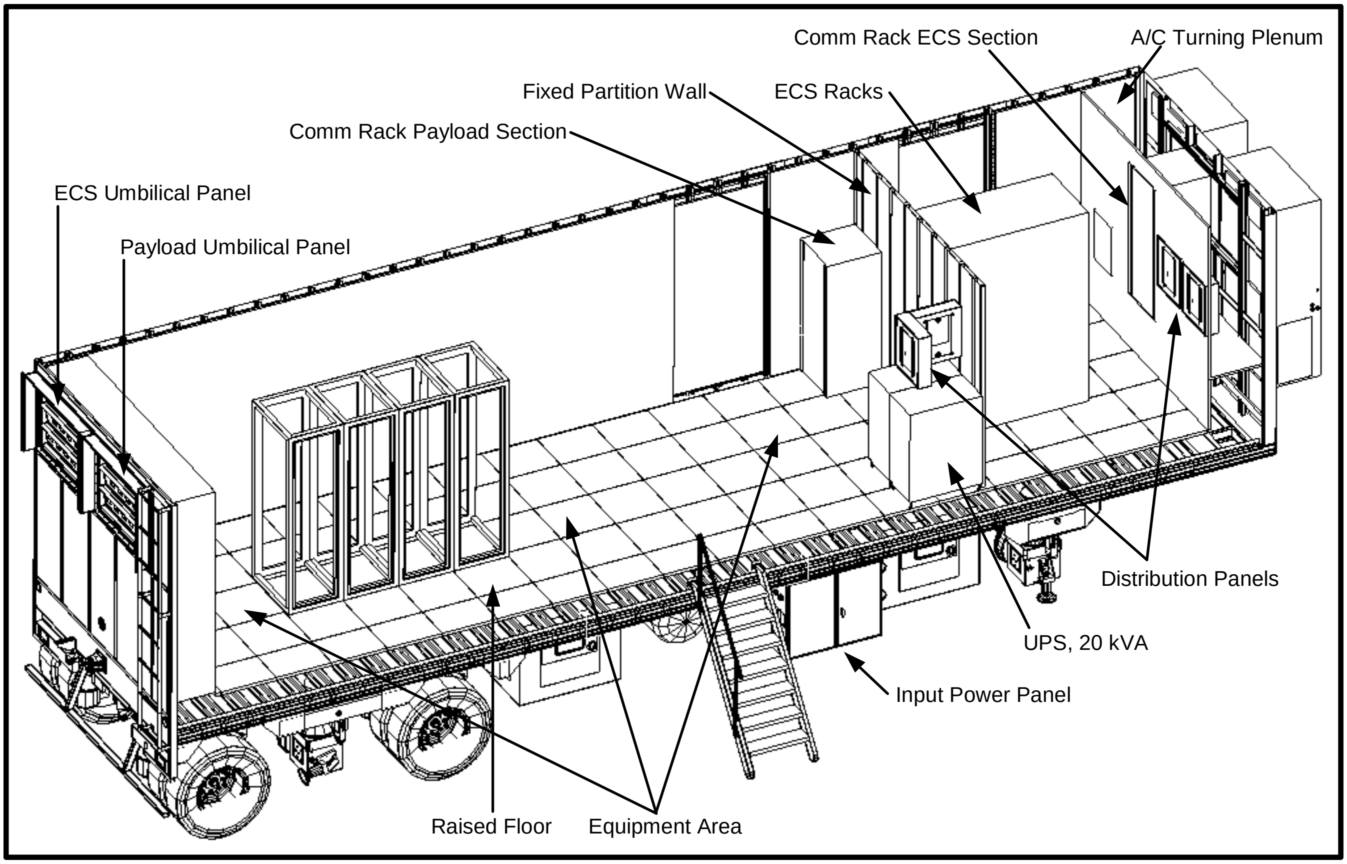


Figure 6.3.4-1 Payload Van

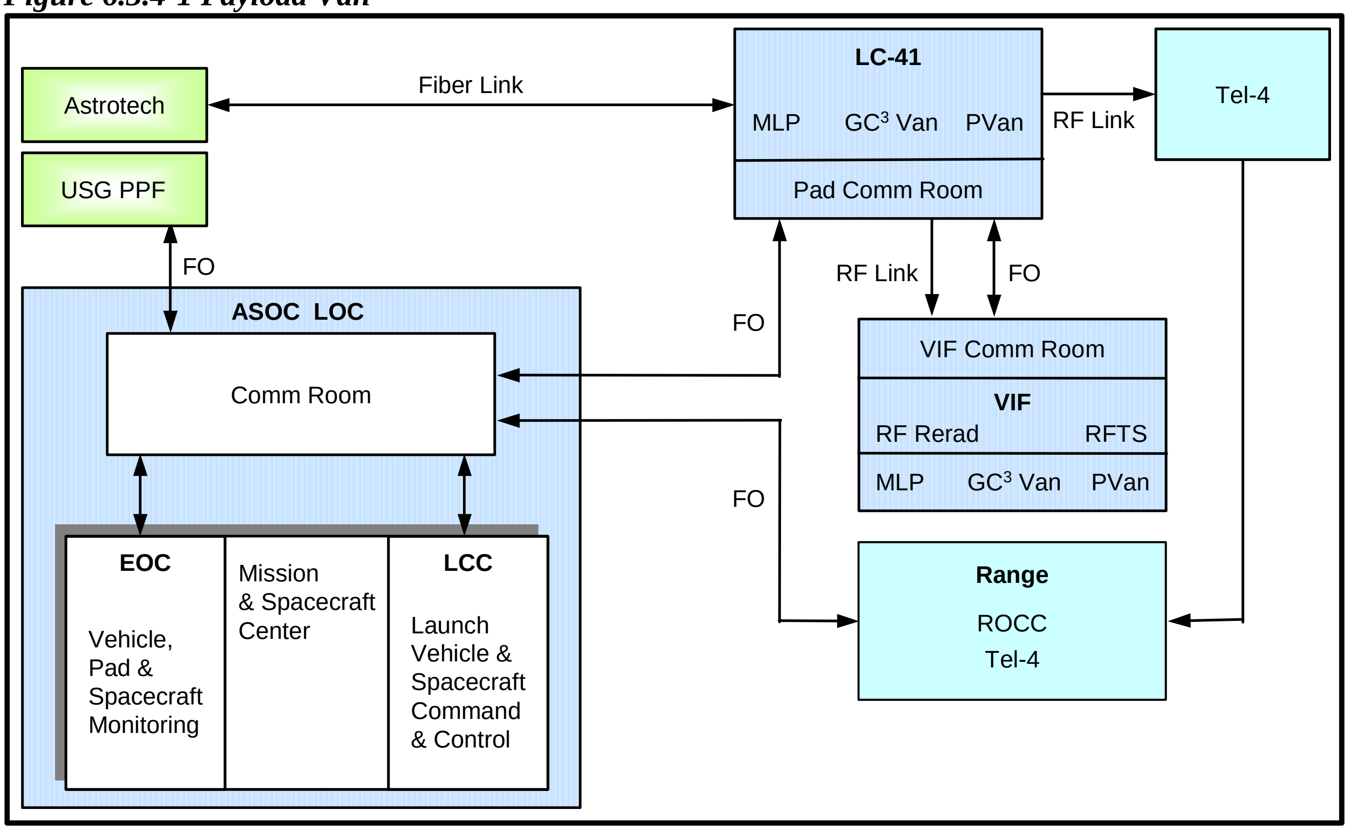


Figure 6.3.4-2 Communications Network

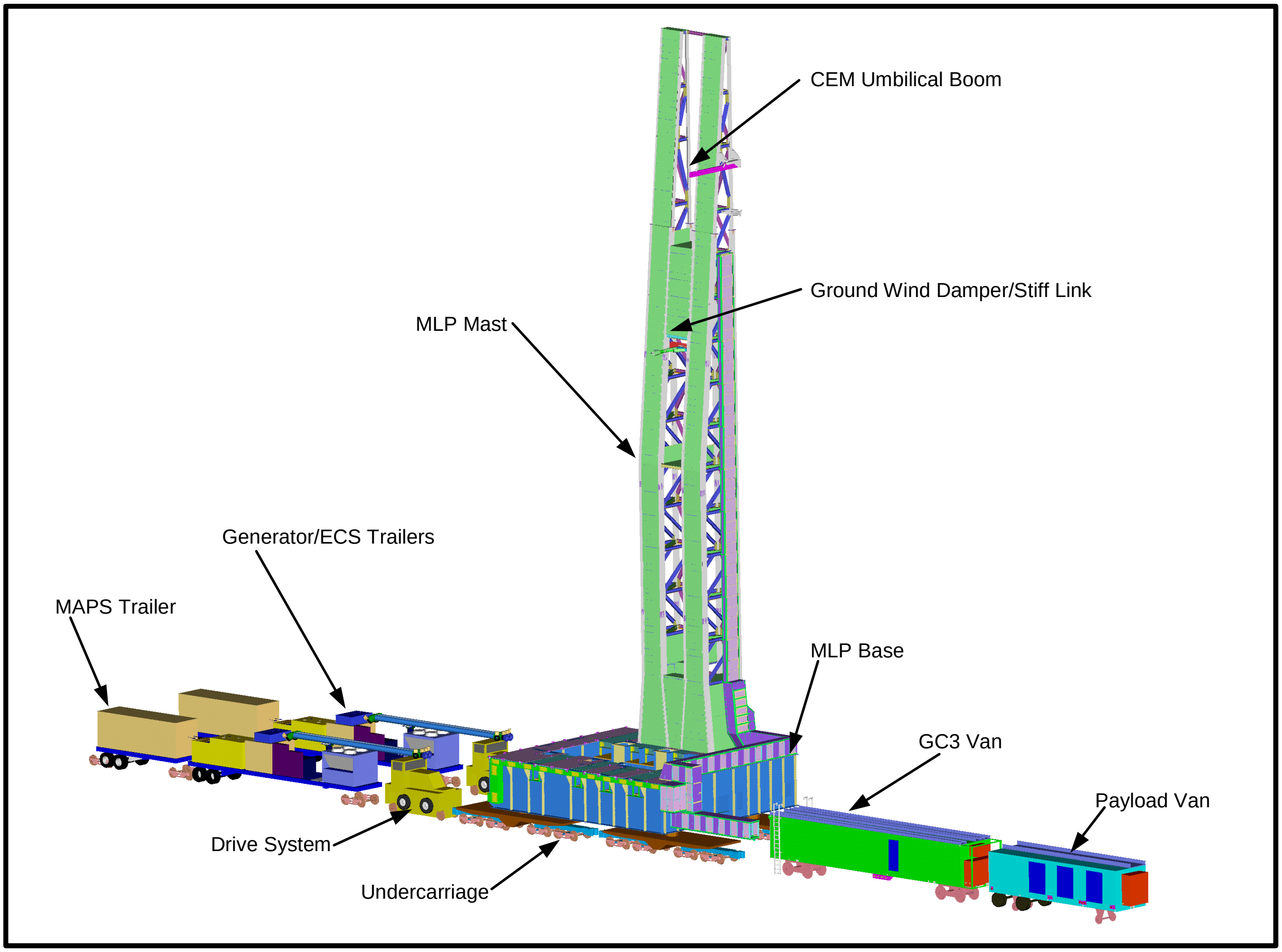


Figure 6.3.5-1 Mobile Launch Platform

allowing switching between facility-provided and trailer-provided conditioned air sources without interruption of conditioned air services.

6.4 VAFB SPACECRAFT FACILITIES

With the expansion of SLC-3 at VAFB, California, to support the launch of the Atlas IIAS vehicle, two payload-processing facilities capable of handling Atlas IIAS class payloads have been developed at VAFB.

The principal processing facility at VAFB is operated by Astrotech Space Operations, Inc., operators of our primary PPF at CCAFS. This facility supports final checkout of smaller class payloads and supported final checkout and encapsulation of the EOS TERRA satellite that flew on an Atlas IIAS from SLC-3E. Depending on VAFB launch interest and U.S. government payload processing requirements for Atlas/Centaur class payloads, the Astrotech/VAFB facility offers compatibility with spacecraft contractor and Lockheed Martin use requirements.

Figure 6.4-1 shows the location of various facilities at VAFB.

6.4.1 Astrotech PPF/VAFB

The Astrotech/VAFB PPF can be used for all payload preparation operations, including liquid propellant transfer, SRM and ordnance installations, and payload fairing encapsulation. The facility is near the VAFB airfield, approximately 12 km from SLC-3E. The PPF (Fig. 6.4.1-1) contains the following:

1) One airlock,

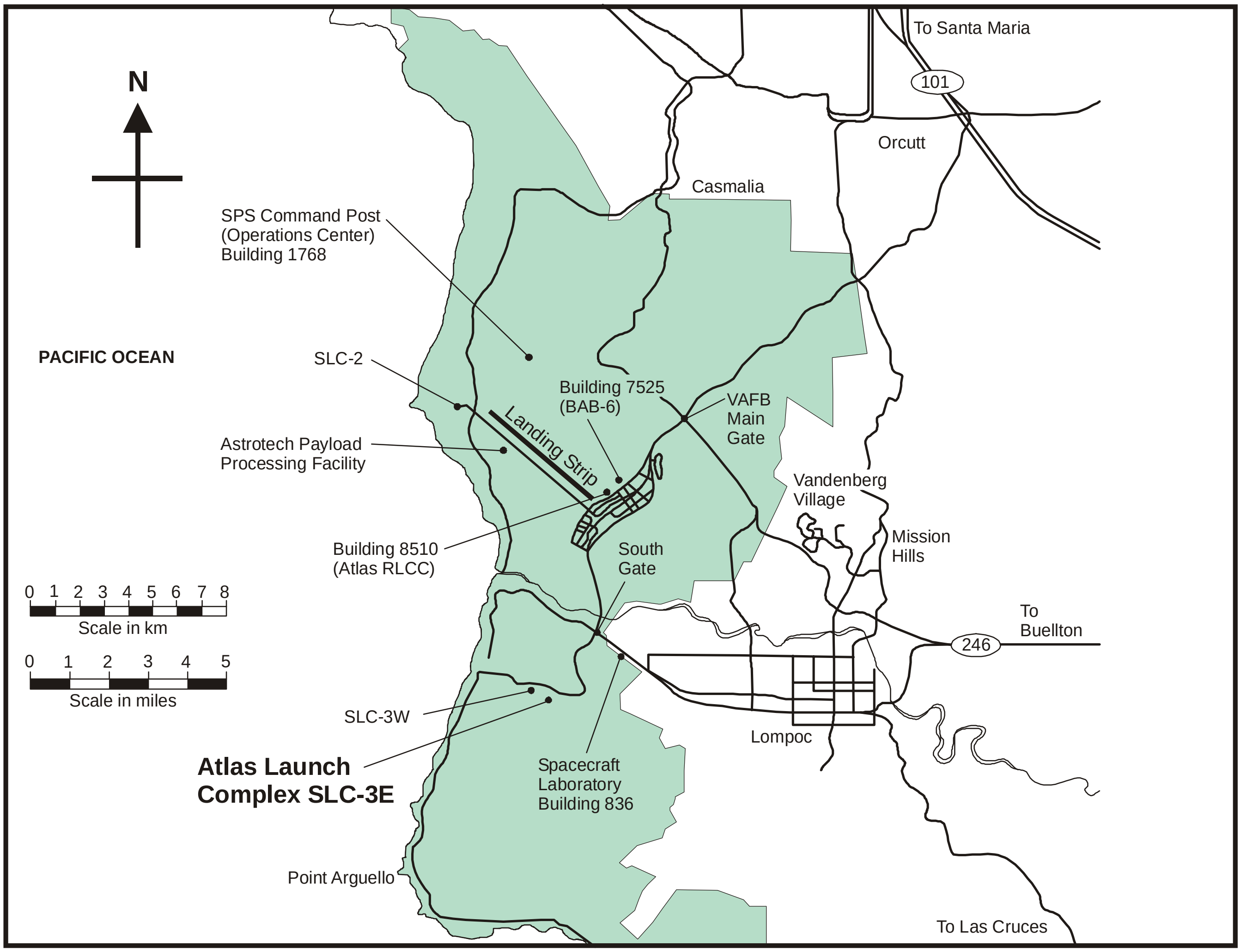


Figure 6.4-1 VAFB Facilities

2) Two high bays,
3) Three low bays,
4) Control rooms (one per high bay),
5) Auxiliary control room,
6) Two walk-in coolers.

In addition, a facility and customer support office (Fig. 6.4.1-2) is available and is shared by Astrotech resident professional and administrative staff and customer personnel. Shared support areas include office space, a conference room, a copier, a facsimile, and amenities.

Spacecraft Services—A full complement of services can be provided at Astrotech to support payload processing and integration.

Electrical Power—The Astrotech/VAFB facility is served by 480-Vac/three-phase commercial 60-Hz electrical power that can be redistributed as 480-Vac/three-phase/30-A, 120/208-Vac/three-phase/60-A, or 125-Vac/single-phase, 20-A power to all major areas within the facility. Standard power is backed up by a diesel generator during critical testing and launch periods.

Telephone and Facsimile—Astrotech provides all telephone equipment, local telephone service, and long-distance access. A Group 3 facsimile machine is available and commercial telex service can be arranged.

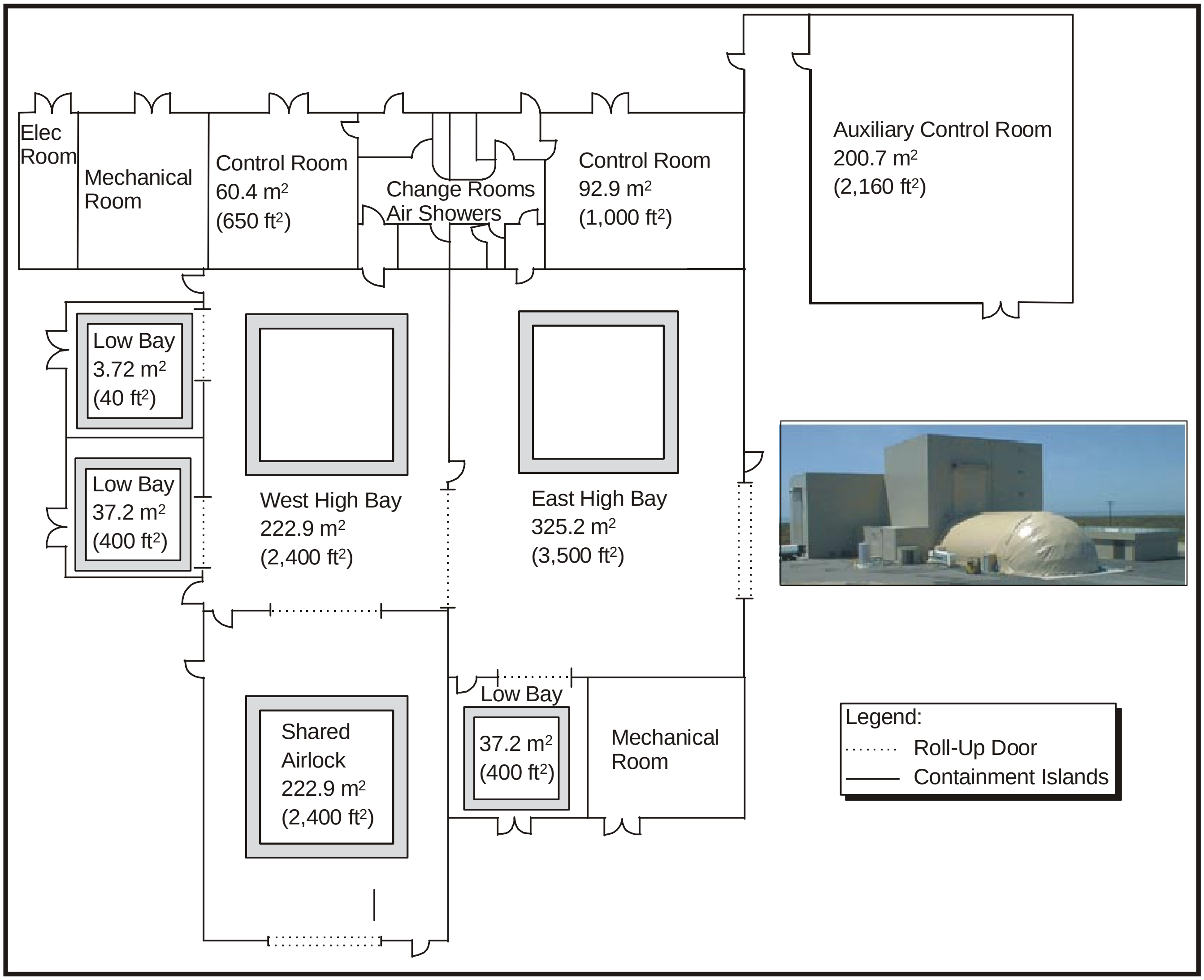


Figure 6.4.1-1 Astrotech/VAFB Payload Processing Facility Layout

Intercommunication Systems—The Operational Voice Intercommunication System provides internal intercom and a link to other facilities as VAFB. TOPS nets are available throughout the PPF. TOPS provides operational communications to other government facilities at VAFB. TOPS allows entrance into the government voice net for direct participation during flight readiness tests and launch countdowns. A paging system is also available throughout the complex.

Closed-Circuit Television—Five CCTV cameras are located in the PPF. Two are located in the processing high bay and one is in the airlock. CCTV can be distributed within the Astrotech/VAFB complex to any desired location, including the Auxiliary Control Room and Technical Support Building.

Remote Spacecraft Control Center—Astrotech has the capability to link remote ground stations (voice and data) between Astrotech and VAFB resources.

Temperature/Humidity Control—A 2,000-cubic feet per minute (cfm) humidity, ventilation, and air-conditioning (HVAC) control system provides reliable air conditioning for cleanroom operations and is capable of maintaining temperature at 21 ±1.1°C (70 ±2°F) with a relative humidity of 45 ±10%. Positive pressure is maintained in all cleanroom areas. Air is circulated through the high-efficiency particulate air (HEPA) filter bank at 3.5-4 room changes per hour. Differential pressure can be maintained between control rooms and cleanrooms to prevent toxic vapor leaks into adjacent areas.

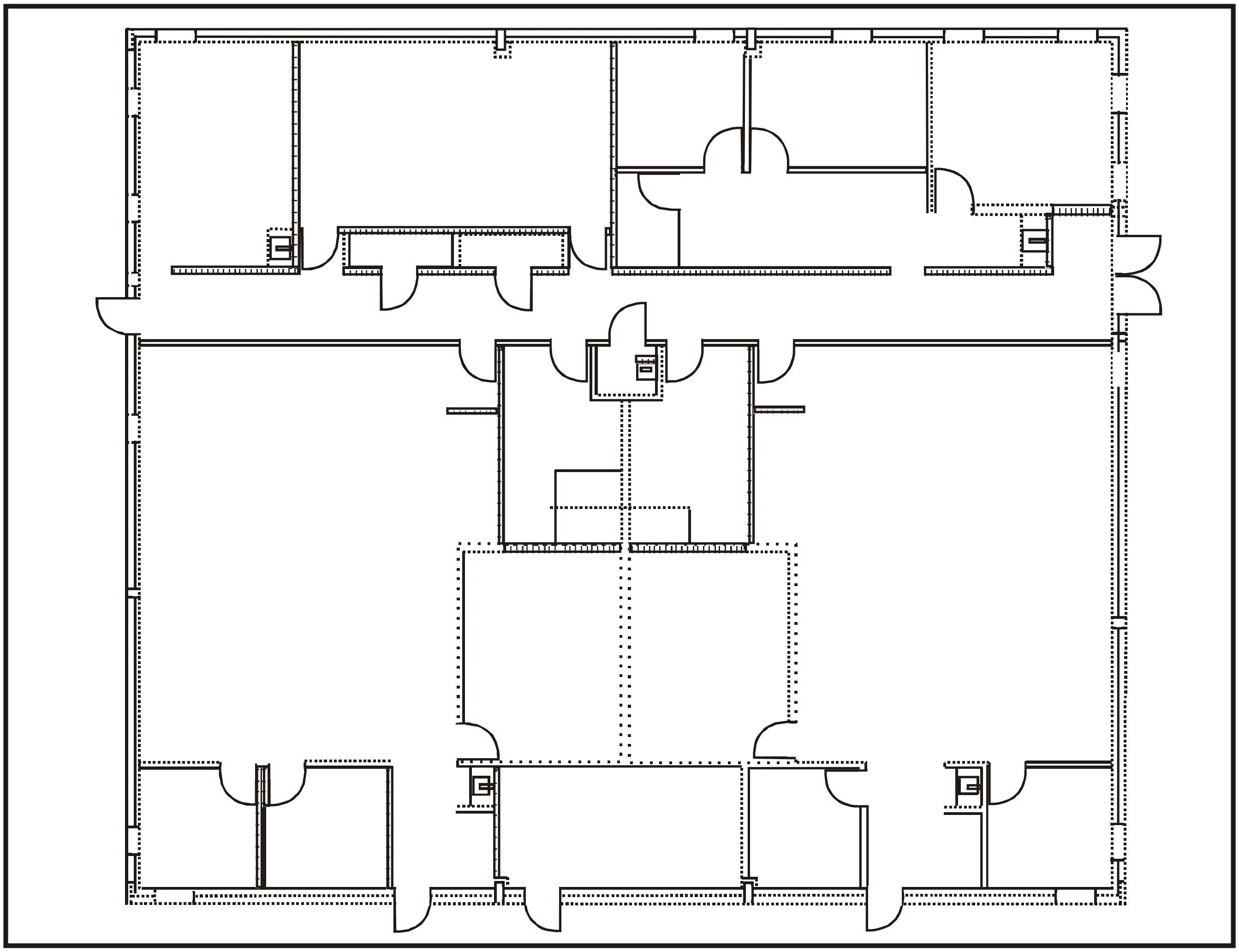


Figure 6.4.1-2 Technical Support Building

Compressed Air—A stationary, two-stage, rotary-tooth compressor supplies oil-free compressed air for breathing, shop air, and pallet air applications. The 30-horse power (hp) compressor provides 100 cfm at 125 pounds per in.^{2} (psi). Breathing air purifiers meet current Occupational Health and Safety Administration (OSHA), National Institute for Occupational Safety and Health (NIOSH), and Environmental Protection Agency (EPA) guidelines for production of Grade D breathing air.

Security and Emergency Support—Physical security is provided by a locked gate and two S&G locked entry doors. All doors providing access to closed areas are alarmed with remote readout at the VAFB Law Enforcement Desk. The alarm system is designed to allow completely segregated operations in the two processing high bays.

6.4.2 Other Spacecraft Facilities

Other USAF, NASA, or privately operated facilities may become available for Atlas class payload processing in the future. This document will be updated as more information becomes available.

6.4.3 Payload Encapsulation and Transportation to SLC-3

During final checkout and propellant loading of the spacecraft, Lockheed Martin will require use of the PPF for approximately 30 days to receive and verify cleanliness of the fairing and to encapsulate the spacecraft.

After encapsulation, Lockheed Martin will transport the encapsulated payload to SLC-3E and mate the spacecraft/fairing assembly to the launch vehicle. Transportation hardware and procedures will be similar to those used at our East Coast LC-36 launch site. Postmate spacecraft testing can be performed from GSE located on the MST, in the LSB payload user's room, or through connectivity to the VAFB fiber-optics transmission system (FOTS) from offsite locations.

6.5 ATLAS IIAS SPACE LAUNCH COMPLEX-3 (SLC-3)

SLC-3 at VAFB in California has supported the launch of Atlas vehicles since the 1970s. Through March 1995, the SLC-3W complex supported the launch of refurbished Atlas E space launch vehicles to deliver payloads such as the National Oceanic and Atmospheric Administration (NOAA) and Defense Meteorological Satellite Program (DMSP) polar weather satellites to low-Earth orbit (LEO). In 1992, the USAF contracted with Lockheed Martin to convert the inactive SLC-3E site to support the launch of Atlas/Centaur to orbits not attainable from the East Coast CCAFS launch site (Fig. 6.5-1). The initial operational capability of SLC-3E was late 1997. This section discusses facilities available to support spacecraft and Atlas IIAS launch vehicle integration and launch.

SLC-3 is located on South VAFB, 11 km (7 miles) from the base industrial area on North VAFB and approximately 6.5 km (4 miles) from NASA Building 836. For reference, locations of SLC-2, SLC-3W, and Buildings 8510 and 7525 are shown in Figure 6.3-1. Major facilities at SLC-3 include the MST, LSB, UT, and a Launch Operations Building (LOB).

The Remote Launch Control Center (RLCC) is located in Building 8510 on North VAFB. Building 7525, also located on North VAFB, is used for launch vehicle receiving and inspection. High-volume, high-pressure GN_2 is supplied to the SLC-3E site from the South Vandenberg nitrogen generation plant operated by the USAF.

Figure 6.5-2 provides a view of the SLC-3E launch complex.

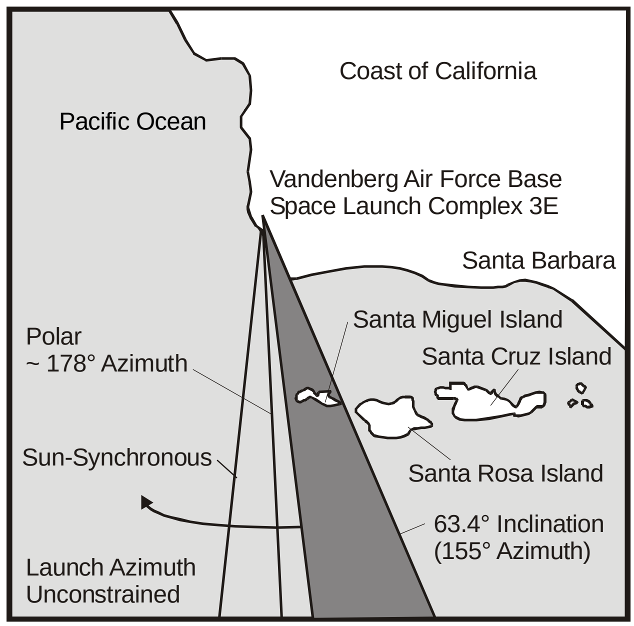


Figure 6.5-1 Available Launch Azimuths from Complex 3E

Figure 6.5-2 SLC-3E modifications for Atlas IIA/IIAS launches are complete.

6.5.1 Mobile Service Tower (MST)

The MST is a multilevel, movable, totally enclosed steel-braced frame structure for servicing of launch vehicles and payloads (Fig. 6.5.1-1). A truck system on rails is used for transporting the MST from its park position at a point approximately 76.2-m (250-ft) south of the LSB to its service position over the launcher. The tower is secured in place with a seismic tie-down system at both tower positions. The MST is normally in place over the launch pad except during major systems tests and before cryogenic tanking during the launch countdown sequence.

The MST has 19 levels. A hammerhead overhang is incorporated at the top of the structure on the north side to allow an 18,150-kg (20-ton) overhead bridge crane on Level 19 to move outside

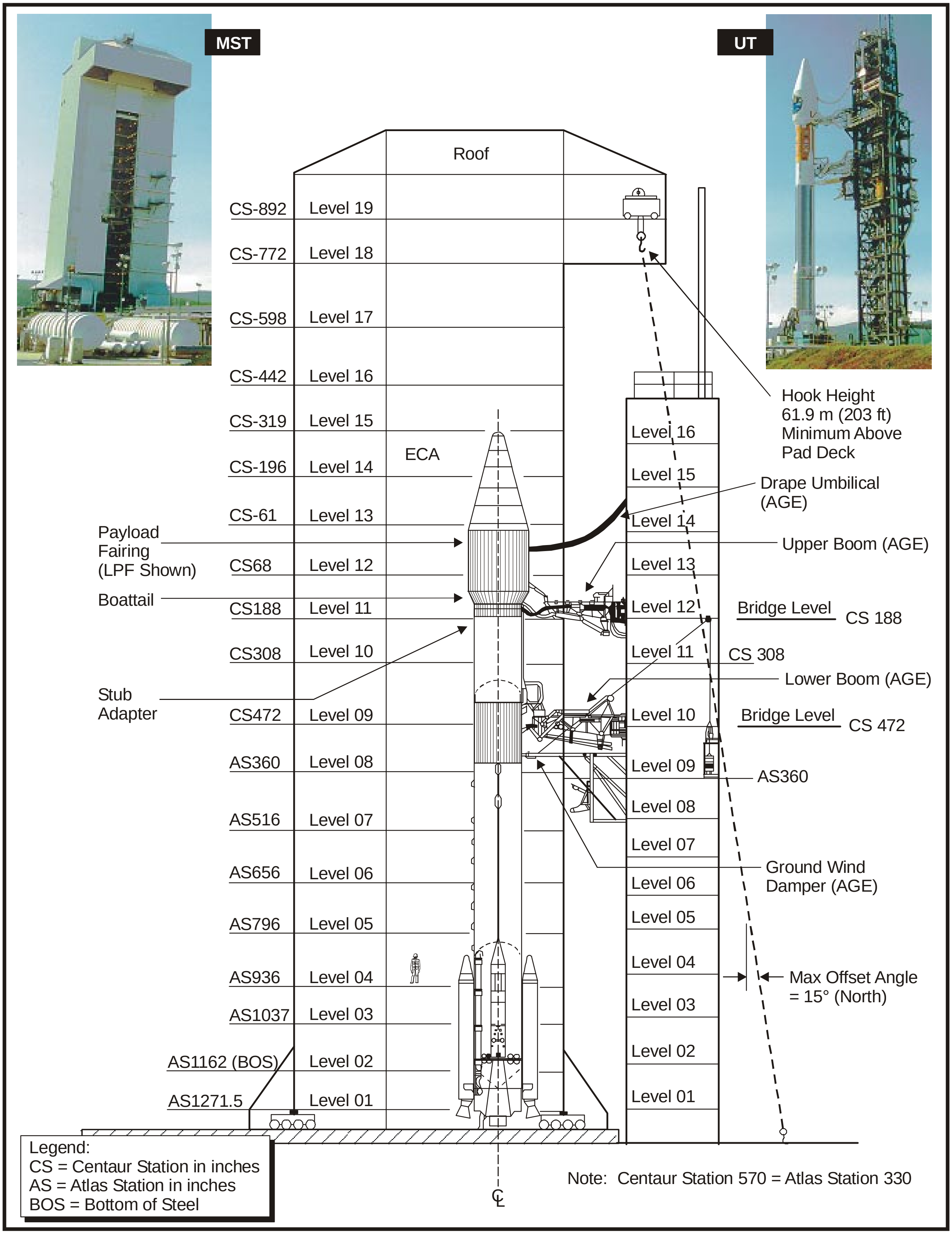


Figure 6.5.1-1 SLC-3E MST and UT

the MST for erection and mating of the Centaur and payload. A 31,750-kg (35-ton) torus crane under Level 8 is used for erecting and attaching solid rocket boosters (SRB) to the Atlas booster. An erection hoist is used to erect the Atlas IIAS vehicle. The MST provides access to the Atlas booster, the Centaur upper stage, and the payload. It also provides a lighted, weather-protected work area for erection, mating, and checkout of flight vehicle.

In addition to external siding, the MST incorporates an environmentally controlled area (ECA) around the vehicle on Levels 8 through 15 to protect the Centaur and payload.

6.5.2 Umbilical Tower (UT)

The UT is a steel structure with 16 levels (Fig. 6.5.1-1). The UT supports two retractable umbilical booms, a draped umbilical, and a ground wind damper. The draped umbilical is used to supply conditioned air to the payload via its connection to the PLF. The umbilical tower supports power cables, command and control cables, propellant and gas lines, monitoring cables, and air-conditioning ducts routed from the LSB pad deck to appropriate distribution points.

6.5.3 Launch Services Building

The LSB is a reinforced concrete and steel structure that is the platform on which the Atlas family of vehicles is assembled, tested, and launched. The top of the LSB, or LSB pad deck, provides support for the Atlas launcher and the MST while it is in the service position. The LSB pad deck also is a support structure for the UT that has supporting columns extending down through the upper-level pad deck and lower-level foundation into the ground. The LSB provides a protective shelter for shop areas, storage, locker rooms, air-conditioning equipment, electrical switch gear, instrumentation, fluid and gas transfer equipment, launch control equipment, and other launch-related service equipment.

LSB equipment is a front end for all aerospace ground equipment (AGE) and vehicle control functions. This equipment issues commands as requested by operators, provides safing when operator connections are broken, and acquires data for monitoring of all pad activities.

The LSB also contains a payload user's room, which is electrically interconnected to MST Level 11 (with capability to route cabling from Level 11 to Levels 12 through 15) and the T-0 umbilical. Capability also exists to connect the user's room to the FOTS for connectivity to offsite locations.

6.5.4 Launch Operations Building (LOB)

The LOB is an existing reinforced concrete and steel structure that provides 24-hr launch complex safety monitoring and control. The LOB provides 24-hr monitoring for critical systems and command and control capabilities except during hazardous operations, when responsibility is transferred to the RLCC. Systems that are monitored from the LOB include the environmental control system, the fire and vapor detection system, and the fire suppression and deluge system.

6.5.5 SLC-3E Payload Support Services

Electrical interfaces exist in the LSB payload user's room and the MST on Levels 11 and 15. This power is available in 120-V 20-A, 208-V 30-A, and 208-V 100-A technical power. Critical technical power circuits, 120-V 20-A and 208-V 30-A, are also provided and are backed up by uninterruptible power systems.

To support spacecraft testing while in the MST, GN_2 and GHe support services are supplied as part of the facility on MST Levels 11, 12, and 13. Type 1, Grade B, GN_2 per MIL-P-27401 is supplied through a 2-micron nominal 10-micron absolute filter in the pressure ranges of 0-100, 0-400, 1,500-3,600, and 2,500-5,000 psig. Type 1, Grade A, GHe per MIL-P-27407 is also supplied through a 2-micron nominal 10-micron absolute filter in the pressure ranges of 0-100 and 2,500-5,000 psig. Both clean gas and contaminated gas vent systems exist on MST Levels 11, 12, and 13. Contingency offload of spacecraft propellants is supported via the facility spacecraft propellant deservicing system.

Propellant deservicing interfaces exist on MST Levels 11, 12, and 13 with ground interfaces at the propellant deservicing pad, with portable fuel and oxidizer scrubbers connected to the MST contaminated vent system. A SLC-3E breathing air system is available in the MST on MST Levels 11, 12, and 13, at the propellant deservicing pad, and at the scrubber pad to support self-contained atmospheric-protective ensemble (SCAPE) operations required for spacecraft processing.

A payload user's room is provided in the LSB to support spacecraft testing. The user's room is electrically connected to MST Level 11 by an array of payload support cables. Support cables are terminated at a connector interface panel both in the user's room and in the ECA on MST Level 11. Cable trays and cable passage ways are provided on MST Levels 11, 12, 13, 14, and 15 to route cabling from the connector interface panel on Level 11 to Levels 12-15. In addition, the LSB payload user's room can be electrically connected to the VAFB FOTS to provide connectivity to offsite locations.

6.5.6 Remote Launch Control Center (RLCC)

The RLCC (Fig. 6.5.6-1) is the focal point for launch site test monitoring and recording. The RLCC supports routine daily vehicle and AGE processing activities and total monitor and control over hazardous operations requiring launch site evacuation (i.e., wet dress rehearsal and launch). AGE systems in the RLCC (with interconnectivity between the RLCC and the LSB and LOB at SLC-3E) provide command control and monitoring of the overall launch control system. The RLCC also provides RLCC-to-SLC-3E communications, launch site control interfaces between the CCLS and launch vehicle and AGE, and, through the safe/arm and securing unit (SASU), control of safety-critical functions independent of the CCLS and RLCC-to-SLC-3E AGE interface links.

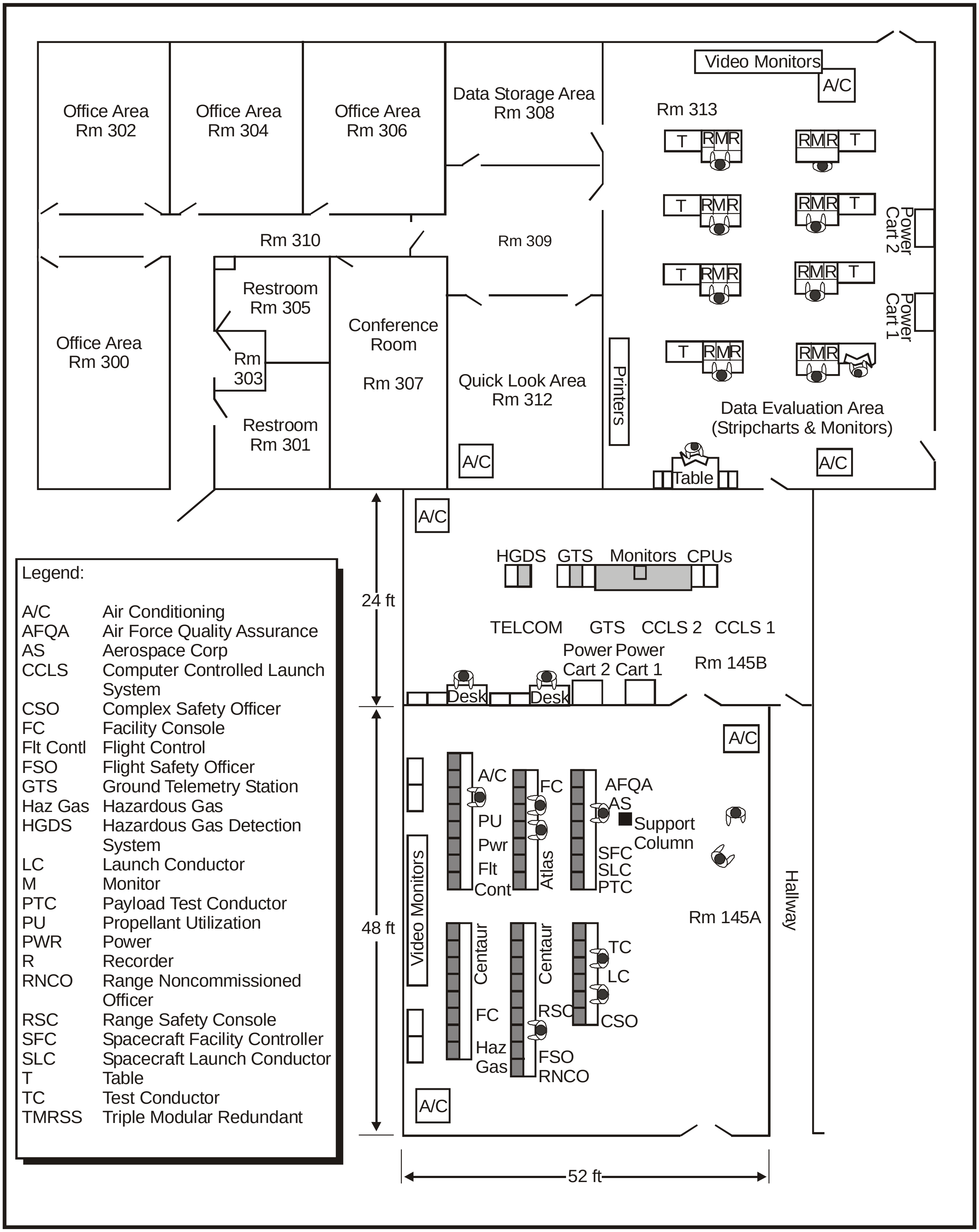


Figure 6.5.6-1 Atlas RLCC Area Within Building 8510

7.0 LAUNCH CAMPAIGN

7.1 VEHICLE INTEGRATION AND SITE PREPARATION

Lockheed Martin provides complete vehicle integration and launch services for its customers. A system of facilities, equipment, and personnel trained in launch vehicle/spacecraft integration and launch operations is in place. The following sections summarize the types of support and services available. Figure 7.1-1 shows a typical factory-to-launch operations flow.

7.1.1 Vehicle Spacecraft Integration

Lockheed Martin performs launch vehicle/spacecraft integration and interface verification testing. Testing includes:

1) Matchmate testing of interface hardware at the spacecraft contractor’s facility;
   a) Prototype items:
      i) For early verification of design;
      ii) For accessibility to install equipment;
      iii) For development of handling/installation procedures;
   b) Flight items:
      i) For verification of critical mating interfaces before hardware delivery to launch site;
      ii) Separation system installation;
      iii) Bolt hole pattern alignments and indexing;
      iv) Mating surface flatness checks;

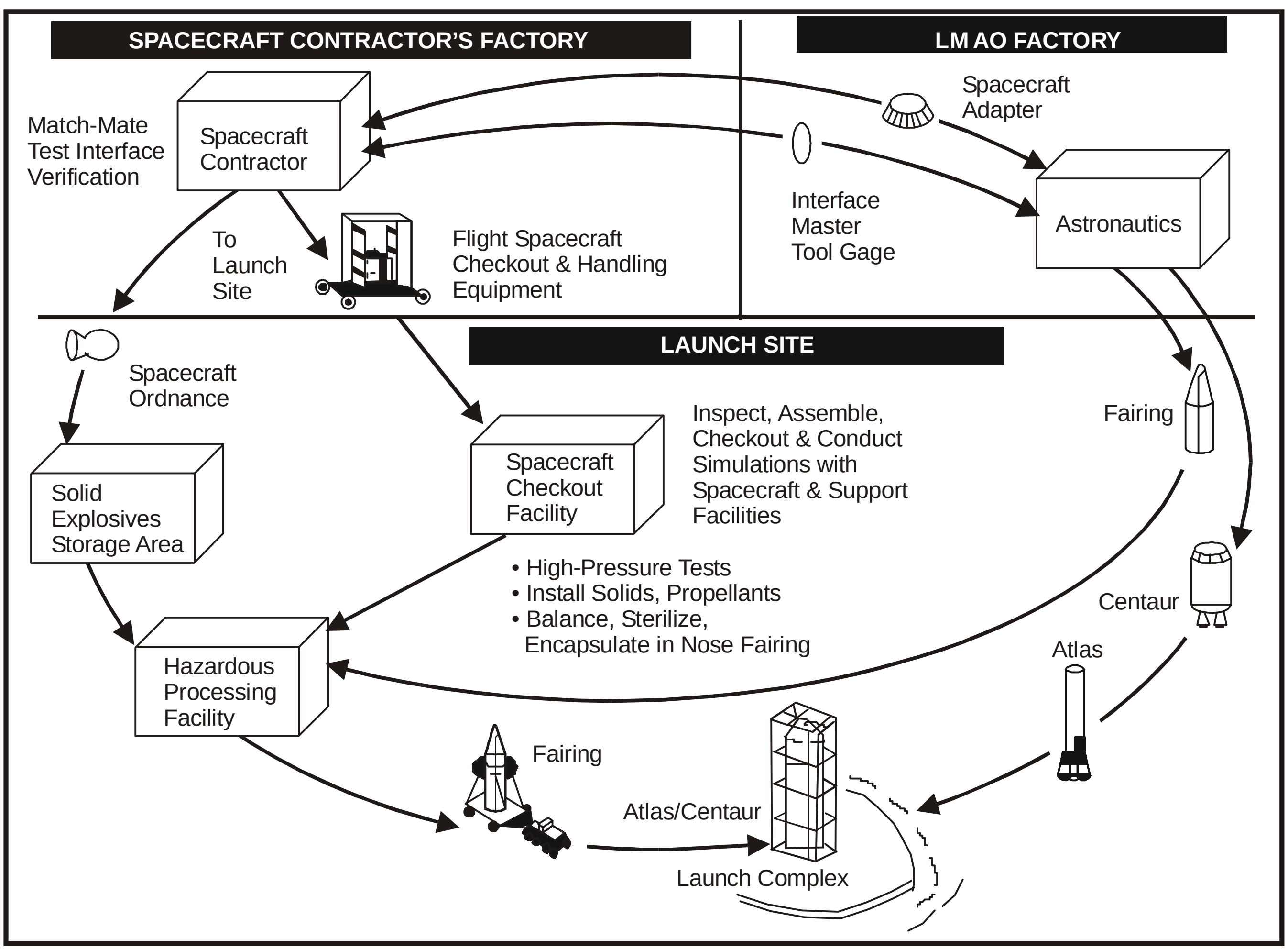


Figure 7.1-1 Typical Factory-to-Launch Operations

v) Electrical conductivity checks;
vi) Electrical harness cable lengths;
vii) Electrical connector mechanical interface compatibilities.

A matchmate at the spacecraft contractor's facility is required for all first-of-a-kind spacecraft. For follow-on and second-of-a-kind spacecraft, matchmates are optional based on experience with the spacecraft and may be performed at the launch site if required.

2) Avionics/electrical system interface testing in the Systems Integration Laboratory (SIL), using a spacecraft simulator or prototype test items for verifying functional compatibility:
   a) Data/instrumentation interfaces;
   b) Flight control signal interfaces;
   c) Pyrotechnic signal interfaces.
3) Special development tests at the launch site;
   a) Spacecraft data flow tests at launch pad (to verify spacecraft mission-peculiar command, control, and/or data return circuits, both hardline and/or radio frequency [RF]);
   b) Electromagnetic compatibility (EMC) testing at the launch pad (to verify spacecraft, launch vehicle, and launch pad combined EMC compatibility).

In addition to integration and interface verification test capabilities, Lockheed Martin uses test facilities to perform system development and qualification testing. Facilities include an integrated acoustic and thermal cycling test facility capable of performing tests on large space vehicles. Other test facilities include the vibration test laboratory, the hydraulic test laboratory, the pneumatic high-pressure and gas flow laboratories, and our propellant tanking test stands.

7.1.2 Launch Services

In addition to its basic responsibilities for Atlas design, manufacture, checkout, and launch, Lockheed Martin offers the following operations integration and documentation services for prelaunch and launch operations:

1) Launch site operations support;
   a) Prelaunch preparation of the Lockheed Martin-supplied payload adapter, nose fairing, and other spacecraft support hardware;
   b) Transport of the encapsulated spacecraft and mating of the encapsulated assembly to the launch vehicle;
   c) Support of launch vehicle/spacecraft interface tests;
   d) Support of spacecraft on-stand launch readiness tests (if requested);
   e) Prepare for and conduct the joint launch countdown.
2) Provide basic facility services and assistance in installation of spacecraft ground support equipment at the launch site:
   a) Installation of spacecraft power, instrumentation, and control equipment in the launch services building and blockhouse or payload van (PVan) and launch operations center (LOC);
   b) Provision of electrical power, water, gaseous helium (GHe) and gaseous nitrogen (GN_2) long-run cable circuits, and on-stand communications;
   c) Supply of on-stand payload air conditioning;
   d) Provision of a spacecraft RF reradiate system in the umbilical tower/mobile launch platform (MLP) mast (permitting on-stand spacecraft RF testing).
3) Coordination, preparation, and maintenance of required range support documents:
   a) Air Force System Command documents required whenever support by any element of the Air Force Satellite Control Facility (AFSCF) is requested (includes Operations Requirements

Document [ORD], which details all requirements for support from the AFSCF remote tracking stations [RTS] and/or satellite test center [STC] during on-orbit flight operations);

b) Range ground safety and flight safety documentation as required by the launch site Range Safety regulation and the Federal Aviation Authority (FAA);
   i) Missile system prelaunch safety package (MSPSP), which provides detailed technical data on all launch vehicle and spacecraft hazardous items, forming the basis for launch site approval of hazardous ground operations at the launch site;
   ii) Flight data safety package, which compiles detailed trajectory and vehicle performance data (nominal and dispersed trajectories, instantaneous impact data, 3-sigma maximum turn rate data, etc), forming the basis for launch site approval of mission-unique targeted trajectory.
   iii) FAA launch license (for commercial missions), which includes items 3.b.i and 3.b.ii above as well as overviews of hazardous spacecraft commodities (propellants, pressure systems, etc.) The baseline FAA license is updated to address each commercial mission.

4) Flight status reporting during launch ascent, which is real-time data processing of upper-stage flight telemetry data:
   a) Mark event voice callouts of major flight events throughout launch ascent;
   b) Orbital parameters of attained parking and transfer orbits (from upper-stage guidance data);
   c) Confirmation of spacecraft separation, time of separation, and spacecraft altitude at separation.
5) Transmission of spacecraft data via upper-stage telemetry (an option), which interleaves a limited amount of spacecraft data into the upper-stage telemetry format and downlinks it as part of the upper-stage flight data stream (Reference Section 4.1.3.5.2 for requirements).
6) Postflight processing of launch vehicle flight data, which provide quick-look and final flight evaluation reports of selected flight data on a timeline and quantitative basis, as negotiated with the customer.

7.1.3 Propellants, Gases, and Ordnance

All chemicals used will be in compliance with the requirements restricting ozone-depleting chemicals. Minor quantities of GN_2, liquid nitrogen (LN_2), GHe, isopropyl alcohol, and deionized water are provided before propellant loading. A hazardous materials disposal service is also provided. Spacecraft propellants are available at the Cape Canaveral Air Force Station (CCAFS) fuel storage depot. The U.S. national aerospace standards and U.S. military specification that they meet are described in Table 7.1.3-1. Similar services are available at Vandenberg Air Force Base (VAFB). All propellants required by the spacecraft must comply and be handled in compliance with these standards:

1) **Sampling and Handling—**Analysis of fluid and gas samples is provided as specified in the interface control document (ICD);
2) **Propellant Handing and Storage—**Short-term storage and delivery to the Hazardous Processing Facility (HPF) of spacecraft propellants;
3) **Ordnance Storage, Handling, and Test—** Spacecraft ordnance and solid motors receiving inspection, bridge wire check, leak test, motor

Table 7.1.3-1 Hypergolic Propellants Available at CCAFS Fuel Storage Depot

1)	Propellant, Hydrazine, Standard Grade, MIL-P-26536
2)	Propellant, Hydrazine, Monopropellant Grade, MIL-P-26536
3)	Propellant, Hydrazine/Uns-Dimethylhydrazine, MIL-P-27402
4)	Monopropellant, High Purity Hydrazine, MIL-P-26536
5)	Propellant, Monomethylhydrazine, MIL-P-27404
6)	Propellant, Uns-Dimethylhydrazine, MIL-P-25604
7)	Propellant, Nitrogen Tetroxide (NTO), NAS3620
8)	Propellant, Nitrogen Tetroxide (MON-1), NAS3620
9)	Propellant, Nitrogen Tetroxide (MON-3), NAS3620
10)	Propellant, Mixed Oxides of Nitrogen (MON-10), MIL-P-27408
11)	Propellant, Nitrogen Tetroxide (MON-3, Low Iron), NAS3620

buildup, motor cold soak safe and arm check, x-ray, and delivery to HPF. Flight units may be stored for about 3 months and spares may be stored for up to 6 months. Other long-term storage is provided on a space-available basis and must be arranged in advance. In addition, a safe facility is available for test and checkout (receiving, inspection, and lot verification testing) of ordnance devices.

7.2 INTEGRATED TEST PLAN (ITP)

All testing performed during Atlas design, development, manufacture, launch site checkout, and launch operations is planned and controlled through the Atlas Integrated Test Plan (ITP). This encompasses all launch vehicle testing, including spacecraft mission-peculiar equipment and launch vehicle/spacecraft integrated tests. A document titled Test and Evaluation Master Plan (TEMP), with the same function, structure, and use as the ITP, is used for SLC-3E missions.

The ITP/TEMP documents all phases of testing in an organized, structured format. It provides the visibility necessary to formulate an integrated test program that satisfies overall technical requirements and provides a management tool to control test program implementation.

The ITP/TEMP consists of an introductory section (defining test concepts, philosophy, and management policies), a summary section (providing a system-by-system listing of all tests, requirements, and constraints for hardware development), and seven sections designated for seven different phases of testing (e.g., design evaluation, qualification, components, flight acceptance, launch site) (Fig. 7.2-1).

Subsections within these headings consist of the individual test plans for each Atlas component, system, and integrated system, and provided detailed test requirements and parameters necessary to achieve desired test objectives. Each subsection is issued as a unique standalone document, permitting its review, approval, and implementation to be accomplished independently from the parent document.

7.3 TEST PROCEDURES

All test operations are performed according to documented test procedures prepared by test operations personnel using either the approved ITP or TEMP subsections together with engineering drawings and specifications. The procedures for testing of Atlas flight hardware are formally reviewed, approved, and released before testing. The procedures are verified as properly performed by inspection and made a part of each vehicle’s permanent history file for determining acceptance for flight and final launch readiness.

Test procedures are documents for spacecraft mission-peculiar hardware and joint launch vehicle/spacecraft integrated tests and operations. Customers are urged to discuss their needs with Lockheed Martin early in the mission-planning phase so that the various interface and hardware tests can be identified and planned. Customer personnel review and approve mission-peculiar test procedures and participate as required in launch vehicle/spacecraft integrated tests.

7.4 ATLAS IIAS/III LAUNCH VEHICLE VERIFICATION TASKS

The following paragraphs provide an overview of the typical sequence of tests and activities performed during manufacture, prelaunch checkout, major launch readiness operations, and launch countdown of the Atlas IIAS/III launch vehicle.

7.4.1 Factory Tests

Flight vehicle acceptance (or factory) tests are performed after final assembly is complete. Functional testing is typically performed at the system level: low-pressure and leak checks of propellant tanks and intermediate bulkhead, checkout of propellant-level sensing probes, verification of electrical harnesses, and high-pressure pneumatic checks.

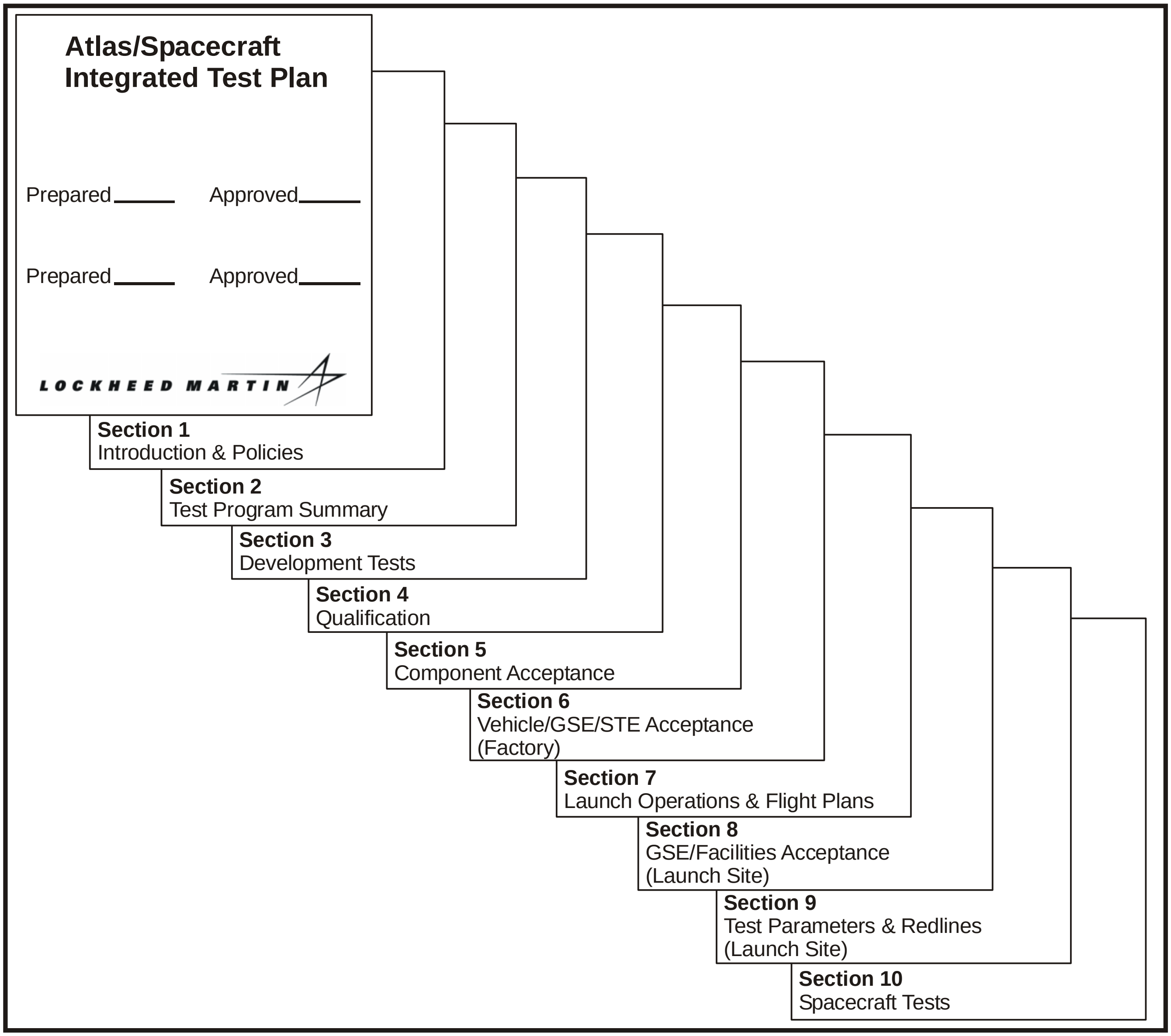


Figure 7.2-1 Integrated Test Plan Organization

7.4.2 Atlas IIAS/III Launch Site Prelaunch Operations

Figure 7.4.2-1 shows a typical Atlas IIAS/III checkout and launch operations sequence. After arrival at the launch site, all launch vehicle items are inspected before erection on the launch pad.

After erection of the Atlas and connection of ground umbilical lines, subsystem and system-level tests are performed to verify compatibility between airborne systems and associated ground support equipment in preparation for subsequent integrated system tests.

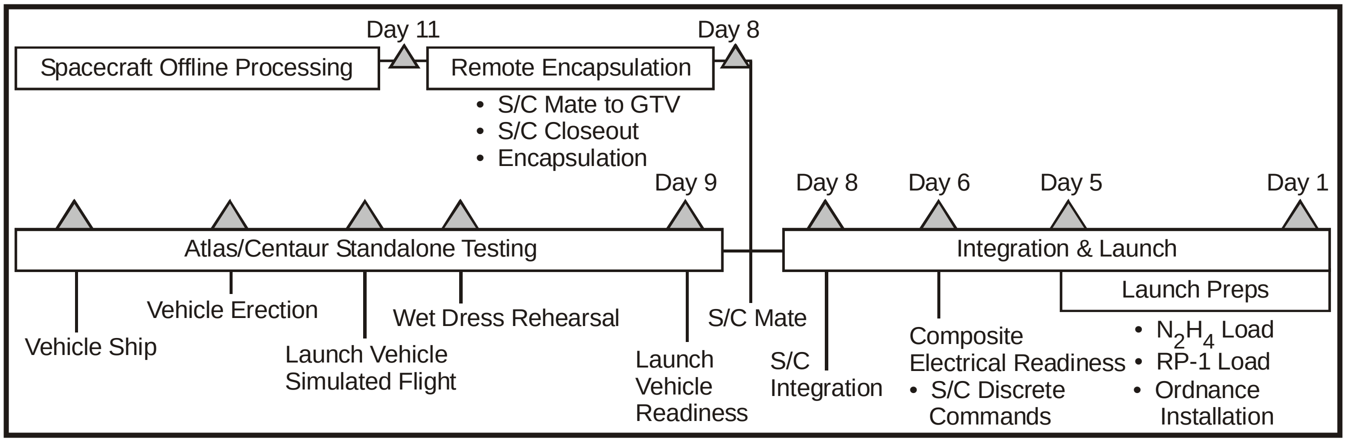


Figure 7.4.2-1 Typical Atlas IIAS/III Launch Operations Sequence

The payload fairing (PLF) halves and payload adapter are prepared for spacecraft encapsulation in the HPF for CCAFS operations and in the Payload Processing Facility (PPF) for VAFB operations (Fig. 7.4.2-2, the 4-m PLF is shown and the 5-m PLF processing is similar). Major tests are performed before the launch vehicle and launch pad are prepared to accept the spacecraft and start integrated operations, as follows.

Launch Vehicle Simulated Flight—This first major launch vehicle test verifies that all integrated Atlas/Centaur ground and airborne electrical systems are compatible and capable of proper integrated system operation throughout a simulated launch countdown and plus-count flight sequence.

Wet Dress Rehearsal (WDR)—The WDR is a tanking test to verify readiness of all ground/airborne hardware, support functions, the launch countdown procedure, and Atlas and spacecraft system launch operations personnel assigned launch countdown responsibilities. Although pad operations and selected system responses are simulated, the WDR demonstrates that the integrated Atlas ground, airborne, and associated launch support functions, including range operations, are ready to support launch operations.

The customer may monitor launch vehicle simulated flight and WDR operations. Although spacecraft and launch vehicle activities are simulated, participation enhances efficiency of personnel and procedures in subsequent integrated tests and the launch operation.

7.4.3 Atlas IIAS/III Integrated Operations

After successful WDR, the launch vehicle and launch pad are prepared to accept the spacecraft and commencement of integrated operations. Major prelaunch integrated test and operations are discussed in the following paragraphs.

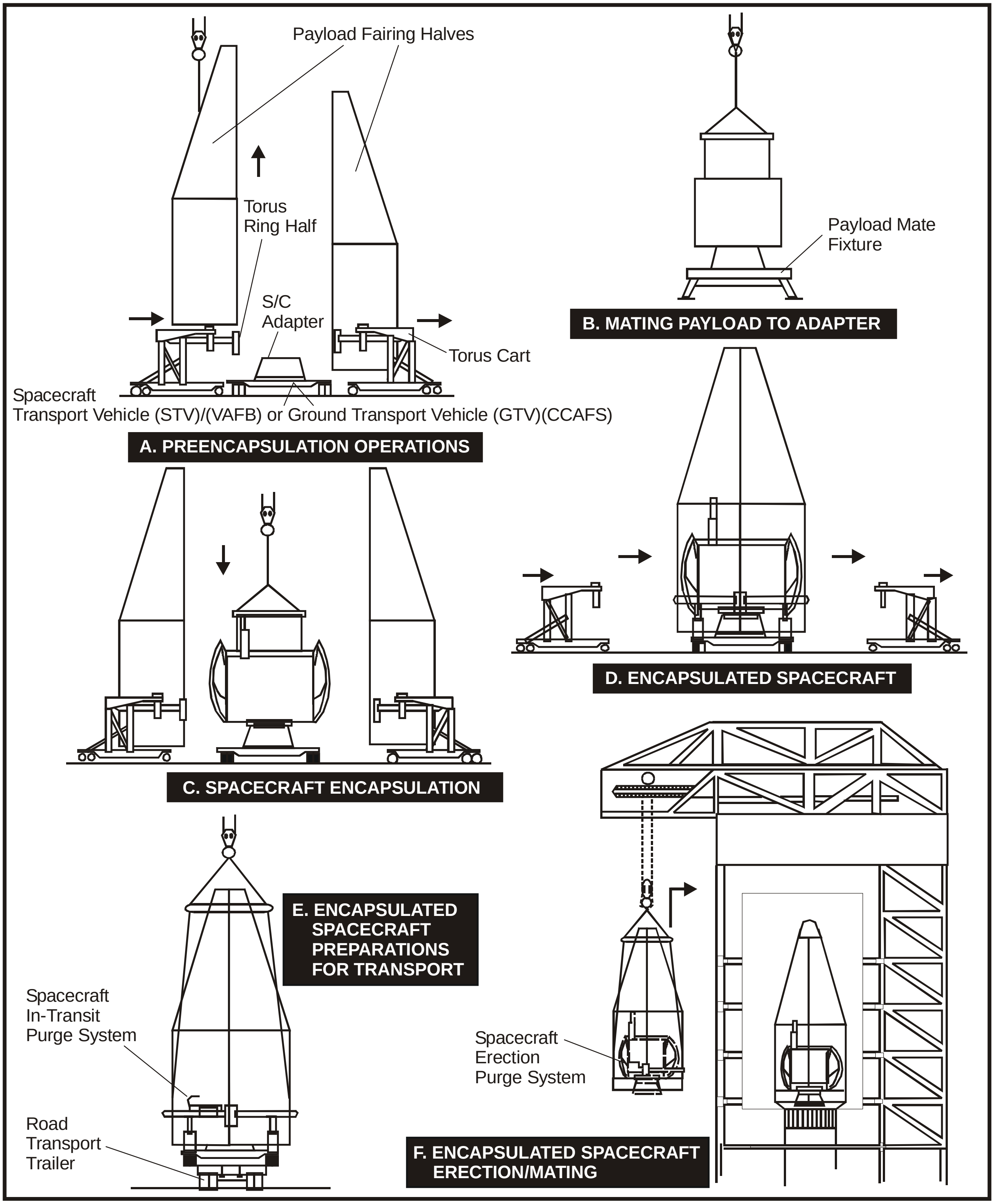


Figure 7.4.2-2 Payload fairing and spacecraft processing are identical to previous Atlas/Centaur operations.

Erection and Mating of Spacecraft to Centaur—Figure 7.4.3-1 depicts the integrated launch operations tasks to be performed. After arrival at the launch pad, the encapsulated spacecraft assembly is positioned atop the launch pad ramp in front of the erected Atlas/Centaur vehicle. Gas conditioning is available for the encapsulated spacecraft on the ramp and during hoisting. Next, a hoisting sling is fastened to the encapsulated assembly-lifting fixture (torus ring) and tie-downs to the ground transport vehicle are released.

The assembly is then hoisted into the service tower by the gantry crane (Fig. 7.4.3-2) and lowered onto Centaur. During final lowering, the payload gas conditioning launch umbilical is attached to the PLF, airflow is initiated, and the payload adapter protective cover is removed. After mechanical attachment of the spacecraft PLF to the Centaur equipment module, the torus ring is removed, necessitating a temporary detachment of the gas-conditioning duct. The 4-m nose cap contamination cover is removed and the flight upper cone/cap assembly is installed.

Spacecraft Functional Tests—Spacecraft functional tests are performed by the spacecraft contractor shortly after spacecraft mating. These tests verify spacecraft/launch vehicle/launch complex/RF interfaces before initiation of more extensive spacecraft on-stand testing. Included as an integral part of these checks is a verification of the spacecraft launch umbilical and the spacecraft flight harness routed through the Centaur vehicle. The main operations performed during spacecraft testing at the service tower are:

1) Umbilical and RF S-band, C-band, and/or K-band link checks (without spacecraft);
2) Spacecraft batteries trickle charge;
3) TM/TC operations in video configuration;
4) TM/TC operations in RF configuration (via reradiation system);

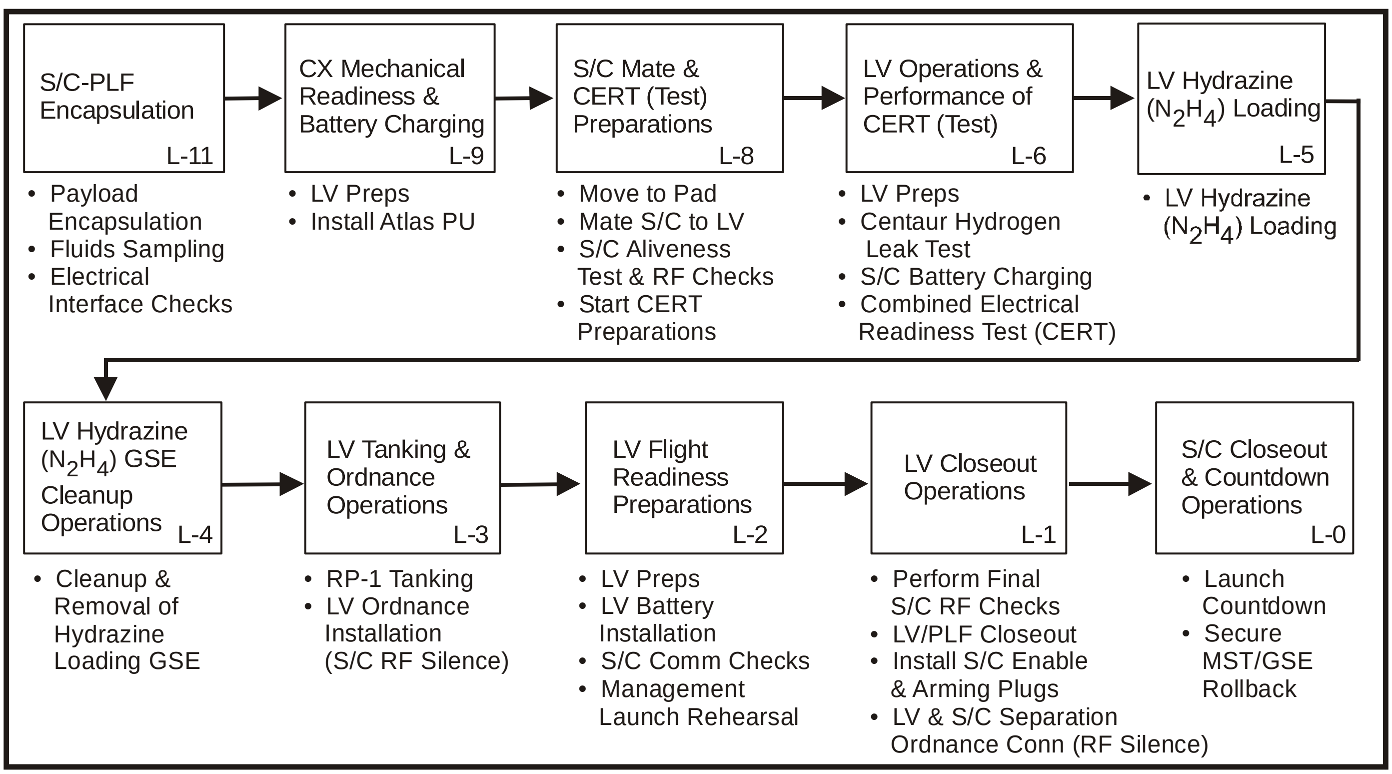


Figure 7.4.3-1 Atlas IIAS/III Launch Site Integrated Launch Operations

5) Spacecraft flight configuration verifications;
6) Spacecraft/ground stations end-to-end test;
7) Stray voltage test with launch vehicle.

Test Equipment:

1) Spacecraft mechanical and electrical ground support equipment (GSE) at the launch complex;
2) Command and telemetry hardlines among launch pad, blockhouse/remote launch control center (RLCC), and PPF;
3) RF consoles equipment at the blockhouse;
4) Spacecraft electrical simulator (as required).

Test Description—These checks are performed with the spacecraft mechanically and electrically mated to the Centaur as for launch. The spacecraft contractor performs a spacecraft-to-blockhouse interface test to verify the total integrated hardline links (i.e., facility hardlines, umbilical cabling, and Centaur flight harness). From the time the spacecraft is connected to the blockhouse test setup to liftoff, routine operations are conducted, such as battery charging, pressures monitoring, launch configuration check, and temperatures monitoring.

Figure 7.4.3-2 Encapsulated spacecraft is hoisted into the MST.

Special Spacecraft Tests—Special spacecraft tests may be necessary to investigate anomalies in planned tests or operations; reverify equipment operation or performance following changes that were made to correct anomalies; or accommodate spacecraft launch site schedules or operations planning. All special tests are conducted according to written and approved test procedures. For example, the spacecraft contractor may perform an operational spacecraft practice countdown to verify spacecraft timelines and to coordinate Lockheed Martin support during final launch preparations. In some instances a spacecraft simulator may be required to validate electrical interfaces, spacecraft communications, and spacecraft GSE before mating the encapsulated payload with the launch vehicle.

Composite Electrical Readiness Test (CERT)—Lockheed Martin performs this test as the final integrated launch vehicle readiness test before start of the launch countdown. The spacecraft contractor provides an input to the test procedure and participates in the test. The test typically is performed on L-6 day.

Purpose—To provide a launch readiness verification of the Atlas/Centaur ground and airborne electrical systems (with a minimum of systems violation) after reconnection of the ground umbilicals, after mating of the flight spacecraft assembly, and before pyrotechnic installation for launch. This is the final EMI compatibility test between the spacecraft and launch vehicle.

Test Configuration—The Atlas vehicle is in a flight configuration except for pyrotechnics, propellants, and batteries. All batteries are simulated by the gantry test rack (GTR) battery simulator system. The umbilicals remain connected for this test with the service tower in place and the GTR monitoring vehicle loads. The spacecraft is mated in the launch configuration, including pyrotechnics, propellants, and batteries.

Test Conditions—All Atlas and Centaur umbilicals remain connected throughout the test. The battery simulator provides vehicle power to Atlas and Centaur. Additional battery simulators are used for Centaur pyrotechnic batteries. The vehicle is armed and on internal power. The FTS is not tested. All launch vehicle electroexplosive devices (EED) will be simulated by squib simulators. PLF pyrotechnic wiring is used and the spacecraft pyrotechnic circuits are simulated. Atlas/Centaur telemetry and C-band radiate. Landline instrumentation, launch control GSE, the computer-controlled launch set (CCLS), telemetry, and the GTR are used for event monitoring. The spacecraft will be powered and a monitor mode established.

Test Description—Atlas electronic systems operate through an abbreviated launch countdown, which includes a vehicle flight control end-to-end steering test. A simulated flight sequence test is performed. All pyrotechnic signals are generated and each associated airborne pyrotechnic circuit is monitored in the low current mode for proper response. Centaur tank pressurization, N_2H_4 engine valve actuation, prestart, and start phases are monitored for proper vehicle responses. A posttest critique is performed at the completion of the test.

L-4 Day Operations—Between CERT and the start of L-3 day final launch preparations, spacecraft and launch vehicle readiness operations are performed as described in the following paragraph.

Atlas operations consist of removing CERT test equipment and performing ground and airborne systems readiness and early vehicle closeout tasks. Typically, these Lockheed Martin activities are performed on L-4 day, but final scheduling may vary at the discretion of the Lockheed Martin test conductor. Tasks include:

1) Complex electrical readiness;
2) Airborne electrical readiness;
3) Propellant loading control unit (PLCU) readiness;
4) Fluids sampling;
5) Complex mechanical readiness;
6) Atlas propulsion readiness;
7) Centaur propulsion readiness;
8) Boom lanyards installation;
9) Telemetry system readiness;
10) Landline instrumentation readiness;
11) Atlas hydraulic readiness;
12) Centaur hydraulic readiness;
13) RP-1 tanking preparation;
14) Closeout tasks (to be scheduled by launch site).

Hydrazine Tanking Operations—Hydrazine tanking operations of the Centaur reaction control system (RCS) and Atlas roll control module (ARCM) storage spheres occur between L-5 and L-4 days' operations. Because of the safety requirements for N_2H_4 tanking, the launch pad is cleared. No further testing is required for the N_2H_4 system.

L-3 Day Operations—Activities to be performed on L-3 day (and continuing through L-1 day) consist of final preparations necessary to ready the launch vehicle, spacecraft, and launch complex for start of the L-0 day launch countdown. Because many tasks are hazardous (e.g., limiting pad access, RF transmissions) and/or are prerequisites to others, they are organized on an integrated basis with their sequence and timeliness controlled by a launch precountdown operations procedure. Major operations planned for L-3 day are discussed below; those for L-2 and L-1 days are discussed in subsequent paragraphs. This division of tasks versus L-day is typical, but may be varied if required.

Atlas RP-1 Tanking—RP-1 fuel is tanked aboard the first stage. Together with associated preparations and securing, the task requires approximately 4.5 hours to accomplish and requires limited personnel access in the pad area during the period RP-1 is being transferred.

Installation of Atlas/Centaur Pyrotechnics—After completion of the above, an RF-silence period will be imposed during which mechanical installation of launch vehicle pyrotechnics will be performed. This takes approximately 8 hours.

The electrical harnesses are connected to the pyrotechnic devices and shielding caps installed on the pyrotechnic initiator end of the harness. In the period from L-3 through L-1 days, additional EEDs are installed and all EEDs are connected to the appropriate wiring. At L-1 day, the EEDs' connections are in flight configuration.

L-2 Day Operations—L-2 day operations involve a continuation of launch vehicle readiness activities consisting of approximately 8 hours of launch preparations followed by 7 hours of selected closeout tasks. Included are:

1) Atlas propulsion launch preparations (to include trichloroethylene flush [CCAFS] or hot GN_2 purge [VAFB] and hypergolic igniters installation);
2) Umbilical boom lanyard connections;
3) Spacecraft separation pyrotechnic battery installation;
4) Atlas and Centaur battery installations;
5) Closeout tasks (to be scheduled by launch site).

Launch Countdown Rehearsal—A joint spacecraft/launch vehicle countdown rehearsal is typically performed on L-2 day. This procedure uses key elements from the respective countdowns arranged on an abbreviated timeline. The objective of this integration test is to acquaint launch team operations personnel with communications systems, reporting, and status procedures that are used in the launch countdown. Simulated "holds" are included to rehearse hold-and-recycle procedures. The operation is critiqued and recommendations are incorporated as required to improve overall communications procedures. Spacecraft personnel participating in this rehearsal use the actual operating stations they will use for the countdown operation.

L-1 Day Operations—L-1 day operations consist of approximately 15 hours of integrated testing and final readiness tasks. Final scheduling of these activities will be formulated at the launch site and documented in the L-1 day section of the launch precountdown operations procedure. Major operations are summarized in the following paragraphs.

Spacecraft Ordnance Connect—45 SW/30 SW Safety may approve spacecraft ordnance connection on L-1 day. The spacecraft contractor performs this task is performed with Lockheed Martin support. Supporting data must be submitted with the request. During the ordnance connect procedure, power-on and power-off measurements are required on ordnance circuits.

Launch Vehicle Final Readiness Tasks—Early on L-1 day, approximately 9.5 hours of launch vehicle operations are performed:

1) Centaur propulsion final preparations;
2) Centaur hydraulic readiness tests;
3) Atlas and Centaur battery connections;
4) Centaur transfer line purge;
5) Atlas RP-l and liquid oxygen (LO_2) final connections;
6) Centaur C-band and telemetry early tests;
7) Atlas RP-1 system securing.

Launch Vehicle Ordnance Tasks—After completion of the above, launch vehicle final ordnance tasks, lasting approximately 3 hours, are performed by Lockheed Martin. Due to the hazardous nature of these tasks, RF silence is required throughout most of this period. One exception is the period during FTS command tests with the range (before FTS destructor installations), during which the launch pad is in an “area red” condition. Specific ordnance tasks include:

1) Mechanical installation and electrical connection;
    a) Atlas destructor charge, Centaur destructor charge, and Centaur safe/arm initiator;
    b) Atlas/Centaur shaped charge detonators and staging systems.
2) Electrical connection of all previously installed Atlas, Centaur, and PLF pyrotechnics.

Launch Vehicle Closeout Tasks—After completion of ordnance operations, RF silence conditions are lifted. Final L-1 day operations consist of approximately 3.5 hours of PLF, umbilical lanyard, and ground wind damper closeout tasks, and the PLF isolation diaphragm is removed. PLF access doors for the equipment module are closed for flight.

Test and Checkout Responsibilities—Lockheed Martin operations personnel are responsible for launch vehicle standalone operations and for the overall conduct of integrated spacecraft/launch vehicle tests and operations. The spacecraft contractor is responsible for the conduct of spacecraft standalone tests and operations.

Specifically, the Lockheed Martin launch site function performs the following major tasks in the test conduct arena:

1) Launch vehicle integration with the spacecraft contractor activities associated with the launch vehicle and the launch complex;
2) Provide overall launch pad operation integration and conduct the final countdown and launch;
3) Organize and chair ground operations working groups (GOWG) and technical interchange meetings between contractors, as necessary, to provide an integrated spacecraft/launch vehicle operation;
4) Represent the launch vehicle operations at the spacecraft/launch vehicle launch base-related working group meetings, joint procedure critiques, and readiness reviews;
5) Prepare and control all launch vehicle-prepared integrated spacecraft/launch vehicle test procedures and launch vehicle-unique test procedures;
6) Provide integrated safety documentation and monitoring for compliance with the applicable Range Safety requirements;
7) Provide integrated launch complex schedule;
8) Support spacecraft prelaunch activities;
9) Provide launch complex security;
10) Provide launch complex safety;
11) Conduct readiness reviews at key points during launch vehicle processing.

The spacecraft contractor is responsible for providing spacecraft-associated test equipment and the logistical and technical support required to support spacecraft launch operations at the launch site. The spacecraft contractor is responsible for the following major tasks in the test conduct arena:

1) Prepare and control spacecraft standalone procedures;
2) Prepare, input, and approve spacecraft/launch vehicle integrated procedures;
3) Attend all launch base-related working group meetings, joint procedure critiques, and readiness reviews;
4) Provide safety documentation (e.g., procedures plans) and monitor compliance with the applicable range safety requirements;
5) Plan and, if necessary, implement spacecraft abort activities.

The Lockheed Martin program manager continues to manage the mission-peculiar engineering aspects of the program by performing the following major tasks during the launch campaign:

1) Provide technical management overview of the launch vehicle/spacecraft launch campaign to ensure compliance to mission requirements and resolution of requirements issues;
2) Ensure that launch vehicle/spacecraft interface requirements to be verified during launch operations are flowed down into appropriate site checkout and test procedures and parameter/redline documents;
3) Act as engineering focal point, responsible for timely and thorough communication of all issues and decisions within Lockheed Martin engineering and spacecraft community and initiate engineering review board (ERB) activity as required;
4) Integrate ICD waiver/deviation requests and disposition with spacecraft community and integrate ICD change activity as appropriate;
5) Manage mission integration schedule to ensure timely completion of Lockheed Martin engineering milestones for the launch campaign and review and approve mission-peculiar technical documentation;
6) Ensure closeout of all mission-peculiar action items.

Tasks versus their approximate launch-day schedule (launch days are calendar days before launch) are listed in Table 7.4.3-1.

7.4.4 Atlas IIAS/III Launch Countdown Operations

The Atlas IIAS/III launch countdown consists of an approximate 9- to 10-hour count, which includes two built-in holds (one at T-105 minutes [for 30 minutes] and the second at T-5 minutes [for 15 minutes]) to enhance the launch-on-time capability (Fig. 7.4.4-1).

Lockheed Martin's launch conductor performs the overall launch countdown for the total vehicle. The launch management is designed for customers and Lockheed Martin efficiencies and control elements (Fig. 7.4.4-2).

Spacecraft operations during the countdown should be controlled by a spacecraft test conductor located either in the launch control facility or at some other spacecraft control center (e.g., in the spacecraft checkout facility) at the option of the spacecraft customer.

As the launch pad integrator, Lockheed Martin prepares the overall countdown procedure for launch of the vehicle. However, the spacecraft customer prepares its own launch countdown procedure for controlling spacecraft operations. The two procedures are then integrated in a manner that satisfies the operations and safety requirements of both. This integration permits task synchronization through status checks at predetermined times early in the count and a complete mesh of operations during the final steps to launch.

7.5 ATLAS V LAUNCH VEHICLE VERIFICATION TASKS

The following paragraphs provide an overview of the typical sequence of tests and activities performed during manufacture, prelaunch checkout, major launch readiness operations, and launch countdown of the Atlas V launch vehicle.

Table 7.4.3-1 Launch-Day Task Schedule

Tasks	L-Day
1) Ground & Airborne Systems Readiness Tests	L-4
2) Hydrazine Loading of Centaur Reaction Control System	L-4/L-3
3) Atlas RP-1 Tasking	L-3
4) Pyrotechnic Installations	L-3
5) Atlas & Centaur Battery Installations	L-2
6) Ordnance Final Installations & Hookups	L-1
7) Launch Vehicle Closeout Tasks	L-1

7.5.1 Factory Tests

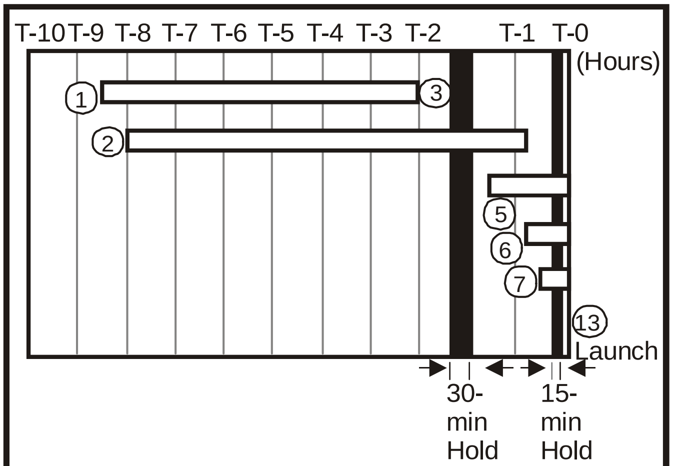
Flight vehicle acceptance (or factory) tests are performed after final assembly is complete. Functional testing is typically performed at the system level: low-pressure and leak checks of propellant tanks and intermediate bulkhead, checkout of propellant-level sensing probes, verification of electrical harnesses, and high-pressure pneumatic checks.

7.5.2 Atlas V Launch Site Prelaunch Operations

After erection of the Atlas V and connection of ground umbilical lines, subsystem and system-level tests are performed to verify compatibility between airborne systems and associated ground support equipment in preparation for subsequent integrated system tests.

The payload fairing (PLF) halves and payload adapter are prepared for spacecraft encapsulation in the HPF. Major tests are performed before the launch vehicle and launch pad are prepared to accept the spacecraft and start integrated operations.

Launch Vehicle Readiness Test (LVRT)—This first major launch vehicle test within the VIF verifies that the launch vehicle (complete, less the spacecraft and PLF) ground and airborne systems



1. Start Launch Complex Mechanical & Electrical Preps — T-515
2. Start S/C Initialization Preps — T-480
3. Start Tower Removal — T-135
4. Built-In Hold for 30 min — T-105
5. Start Centaur LO_2 Tanking — T-91
6. Start Atlas LO_2 Tanking — T-65
7. Start Centaur LH_2 Tanking — T-51
8. T-5 & Holding for 15 min — T-5
9. Atlas to Internal Power — T-4:00
10. Centaur to Internal Power — T-2:00
11. Pressurization to Internal — T-70
12. Start Launch Automatic Sequence — T-31
13. Liftoff — T-0

Figure 7.4.4-1 Atlas IIAS/III Launch Countdown Summary

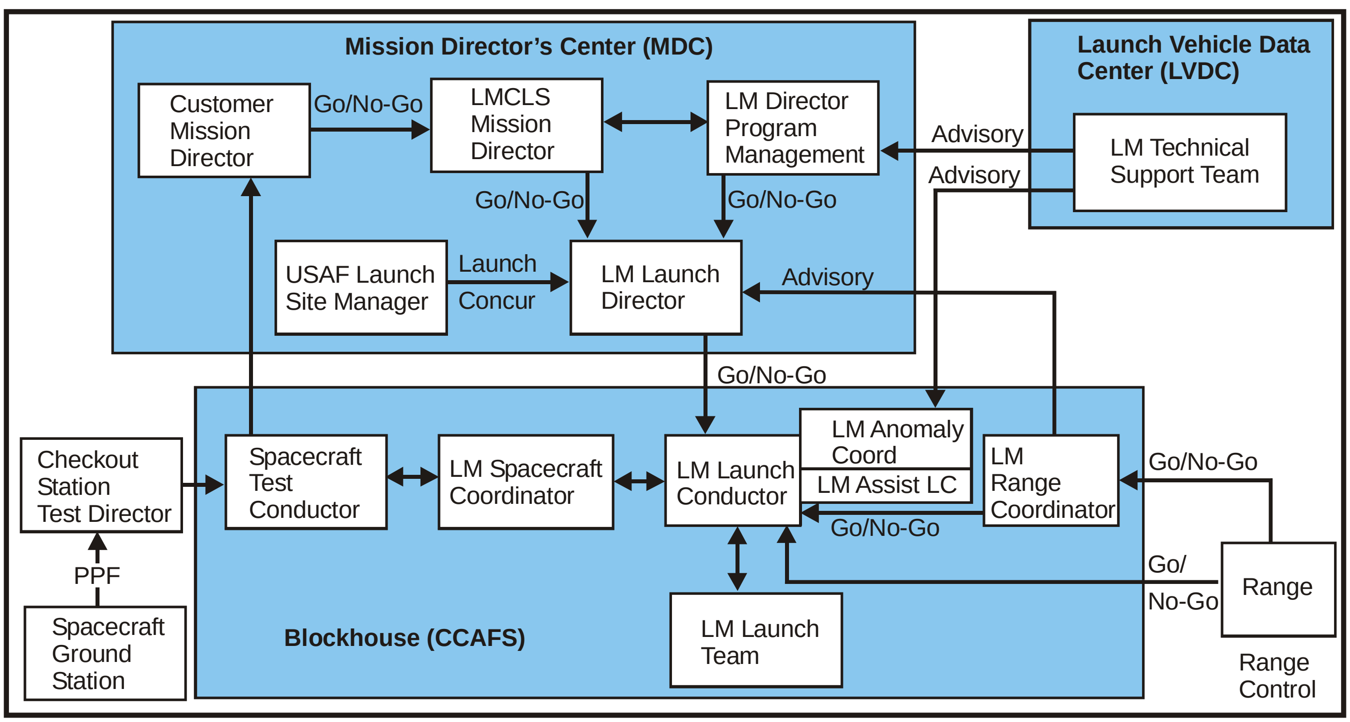


Figure 7.4.4-2 Typical Atlas IIAS/III Launch Day Management Flow Diagram

are compatible and capable of proper integrated system operation throughout a simulated launch countdown and plus-count flight sequence. Systems verified include flight termination system (FTS), ordnance staging, flight controls, RF, and engine hydraulics/alignment.

7.5.3 Atlas V Integrated Operations

L-3 Day, Payload Hoist/Mate—After a successful LVRT, the launch vehicle and the complex are prepared to accept the encapsulated spacecraft and start integrated operations. Spacecraft mating to the launch vehicle occurs according to the schedule coordinated between Atlas V and the spacecraft customer, typically at L-3 days. After arrival at the launch complex, the encapsulated spacecraft assembly is positioned for hoisting onto the Atlas V. Gas conditioning is transferred from the portable unit used during transport to the VIF, to a facility source and is available during hoisting operations. A hoisting sling is fastened to the encapsulated assembly-lifting fixture and tie-downs to the ground transport vehicle are released. The encapsulated payload assembly is then lifted into the VIF and lowered onto the launch vehicle inside the VIF. After mechanical attachment of the spacecraft and PLF to the launch vehicle, the lifting fixture is removed, possibly necessitating a temporary detachment of the gas-conditioning duct. Electrical connections between the payload adapter (PLA) and the launch vehicle are mated to complete the operation.

Postmate Functional Tests—The spacecraft customer may perform limited spacecraft functional tests shortly after spacecraft mating. These tests verify spacecraft/launch vehicle/launch complex/RF interfaces before initiation of more extensive spacecraft testing. Included as an integral part of these checks is a verification of the spacecraft launch umbilical and the spacecraft flight harness routed to the Atlas V standard electrical interface panel. The main operations performed during spacecraft testing at the VIF are:

1) Umbilical and RF S-band, C-band, and/or K-band link checks (without spacecraft);
2) Spacecraft batteries trickle charge;
3) Telemetry/telecommand (TM/TC) operations in hardline telemetry configuration;
4) TM/TC operations in RF configuration (using a reradiation system);
5) Spacecraft flight configuration verifications;
6) Spacecraft/ground stations end-to-end test;
7) Stray voltage test with the launch vehicle.

Postmate Special Tests—Special spacecraft tests may be necessary to investigate anomalies occurring in planned tests or operations, to reverify equipment operation or performance after changes were made to correct anomalies, or to accommodate spacecraft launch site schedules or operations planning. All special tests are conducted according to written and approved test procedures. For example, the spacecraft contractor may perform an operational spacecraft practice countdown to verify spacecraft timelines and coordinate Lockheed Martin support during final launch preparations. In some instances, a customer-provided spacecraft simulator may be required to validate electrical interfaces, spacecraft communications, and spacecraft GSE before mating the encapsulated spacecraft with the launch vehicle.

L-2 Day, Integrated System Test (IST)—Lockheed Martin performs this integrated test with the spacecraft before moving to launch configuration and launch countdown. The spacecraft contractor provides input to the test procedure and participates in the test. An IST flow is depicted in Figure 7.5.3-1. This test exercises key elements of the countdown sequence arranged on an abbreviated timeline.

Purpose—To provide a launch readiness verification of launch vehicle ground and airborne electrical systems (with a minimum of systems violations) after reconnection of ground umbilicals, after mating of the flight spacecraft assembly, and before pyrotechnic installation for launch. This is the final electromagnetic interference (EMI) compatibility test between the spacecraft and launch vehicle. A

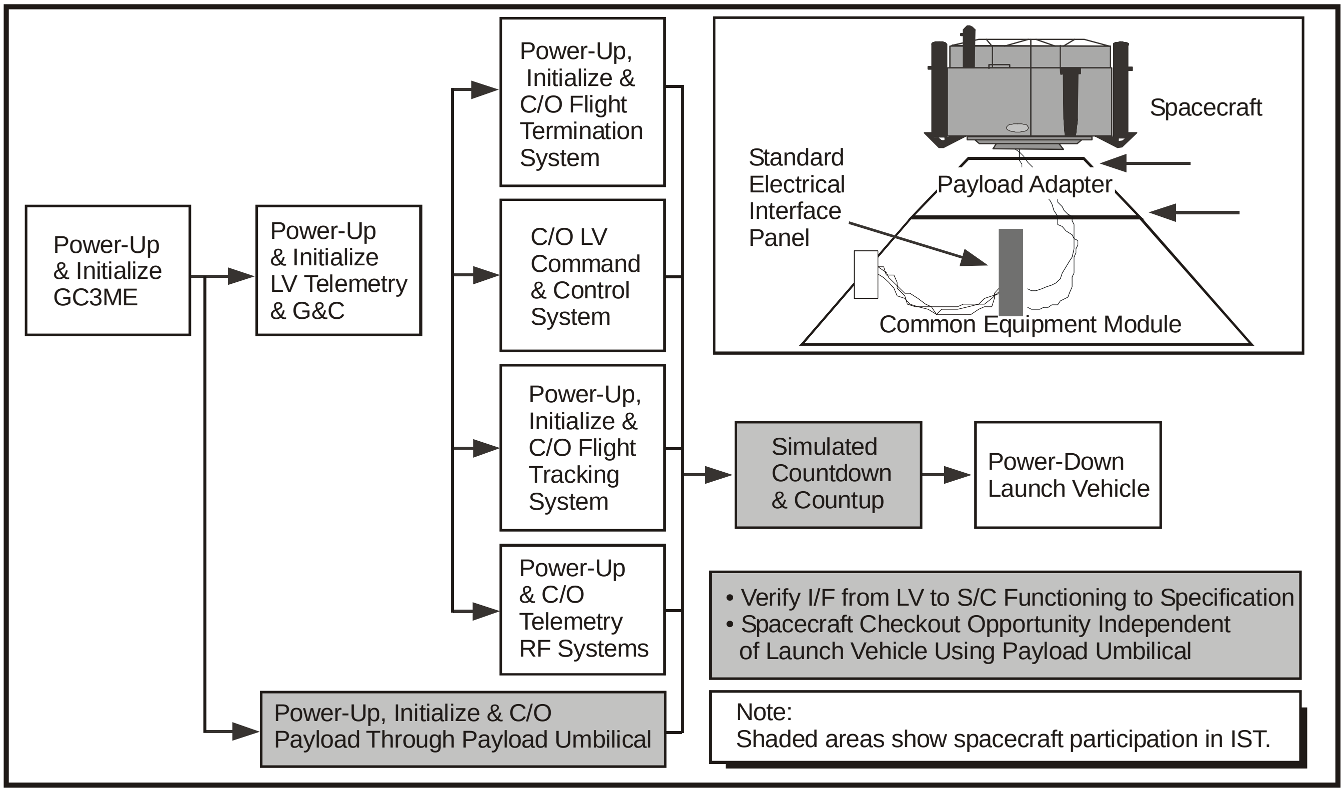


Figure 7.5.3-1 Integrated System Test (IST) Flow

secondary objective of this integration test is to acquaint launch team operations personnel with communications systems, reporting, and status procedures used in the launch countdown. Simulated "holds" are included to rehearse hold and recycle procedures. The operation is critiqued and recommendations are incorporated as required to improve overall communications procedures. Spacecraft personnel participating in this rehearsal use the actual operating stations they will use for the countdown operation.

Test Configuration—The Atlas vehicle is in the VIF and in flight configuration, except for pyrotechnics, main propellants, and batteries. All batteries are simulated by support equipment. The spacecraft is mated in the launch configuration, including pyrotechnics, propellants, and batteries. Spacecraft test and control support equipment is installed in the PVan.

Test Conditions—All launch vehicle umbilicals remain connected throughout the test. Vehicle power to the booster and upper stage is provided by airborne power. All launch vehicle electroexplosive devices are simulated by squib simulators. Atlas/Centaur telemetry, flight termination system (FTS), and C-band RF systems are open-loop tested. The spacecraft is afforded the opportunity to radiate open loop during this test, if required. This test is conducted from the launch control center (LCC), and uses the full set of ground, command, control, and communications (GC^3) systems.

Test Description—Atlas electronic systems operate through an abbreviated launch countdown that includes a vehicle flight control end-to-end steering test. A simulated flight sequence test is performed. All pyrotechnic signals are generated and each associated airborne pyrotechnic circuit is monitored for proper response. Centaur tank pressurization, engine valve actuation, prestart, and start phases are monitored for proper vehicle responses. Spacecraft participation is not required during the initial portions of the test, and no restrictions are imposed on spacecraft activity. Spacecraft and spacecraft launch team participation begins approximately 4 hours into the test. The spacecraft should be configured, as it will be on launch day, at the T-5 hour point. The test continues through an abbreviated countdown, and a nominal "plus" count, terminating at the separation command, and contamination and collision avoidance maneuvers (CCAM). Following this sequence, spacecraft participation in the IST is no longer required, and spacecraft testing and launch preparations may continue. A posttest critique is performed at the completion of the test.

L-1 Day, Final Closeouts—Activities to be performed during final closeouts consist of final preparations necessary to ready the launch vehicle, spacecraft, and launch complex for start of launch day activities. Because many tasks are hazardous (e.g., limiting pad access, RF transmissions) and/or are prerequisites to others, they are organized on an integrated basis with their sequence and timeliness controlled by a launch precountdown operations procedure.

Installation of Pyrotechnics—An RF-silence period will be imposed during which mechanical installation/connection of launch vehicle pyrotechnics will be performed.

Vehicle Compartment Closeout—After connection of pyrotechnics, vehicle compartments are readied for launch. Activities include final visual checks, final hardware configurations/remove-before-flight items, internal platform removal, closeout photographs, and airborne door installations.

Vehicle and Facility Preparations for Roll to Pad—As vehicle compartments are readied for launch, the launch vehicle-to-MLP mast ground umbilicals/lanyards are configured for roll/flight. The VIF platforms, pad, and transport GSE are configured for transport. L-1 day (L-14 hours) is the last opportunity for access to the spacecraft.

Launch Day Operations—Day-of-launch activities include transport of the launch vehicle/MLP from the VIF to the pad, securing the payload van and GC^3 vans in the Payload Equipment Building (PEB), propellant loading, systems verification test, and launch/plus count. The launch countdown timeline is shown in Figure 7.5.3-2.

7.5.4 Atlas V Launch Countdown Operations

Countdown Operations—The launch countdown consists of an approximate 12- to 14-hour count, which includes a built-in hold at T-180 minutes (30-minute hold) and T-5 minutes (10-minute hold) to enhance the launch-on-time capability.

Lockheed Martin's launch conductor performs the overall launch countdown for the total vehicle. The launch management is designed for customers and Lockheed Martin efficiencies and control elements (Fig. 7.5.4-1).

Spacecraft operations during the countdown should be controlled by a spacecraft test conductor located either in the launch control facility or at some other spacecraft control center (e.g., in the spacecraft checkout facility) at the option of the spacecraft customer.

Lockheed Martin prepares the overall countdown procedure for launch of the vehicle. However, the spacecraft customer prepares their own launch countdown procedure for controlling spacecraft operations. The two procedures are then integrated in a manner that satisfies the operations and safety requirements of both. This integration permits task synchronization through status checks at predetermined times early in the count and a complete mesh of operations during the final steps to launch.

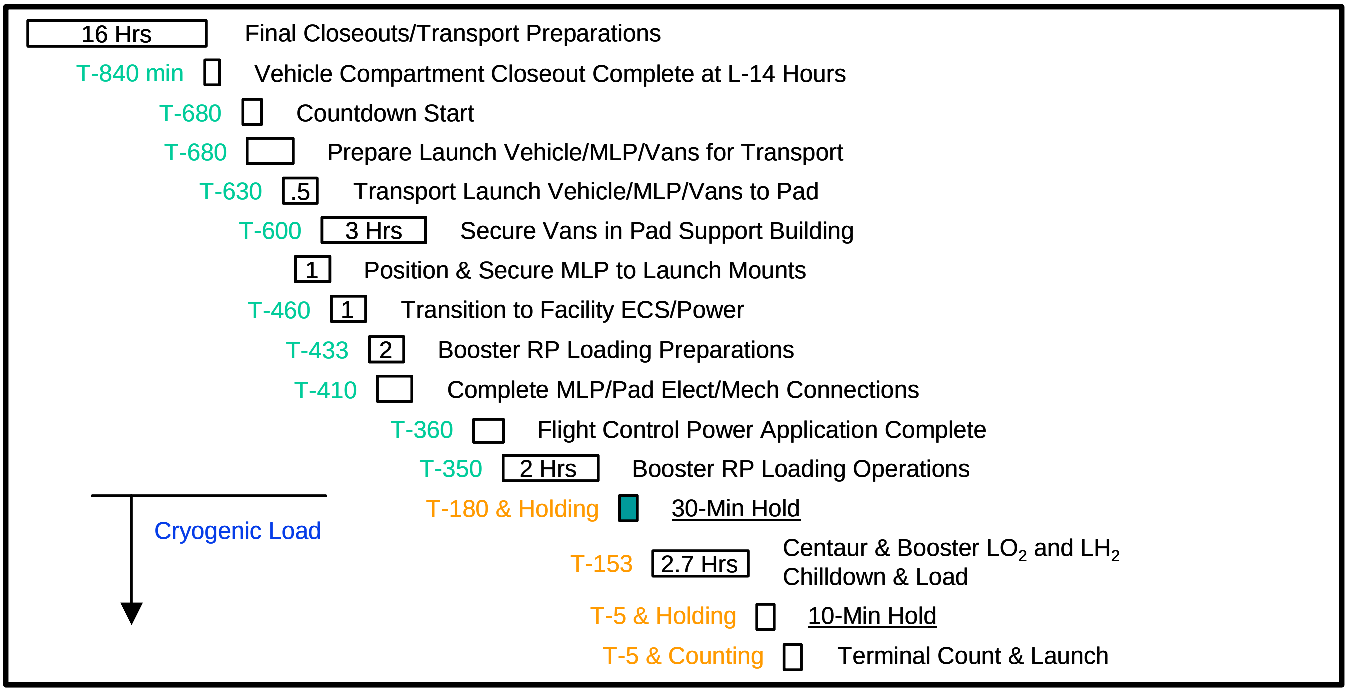


Figure 7.5.3-2 Atlas V Launch Countdown

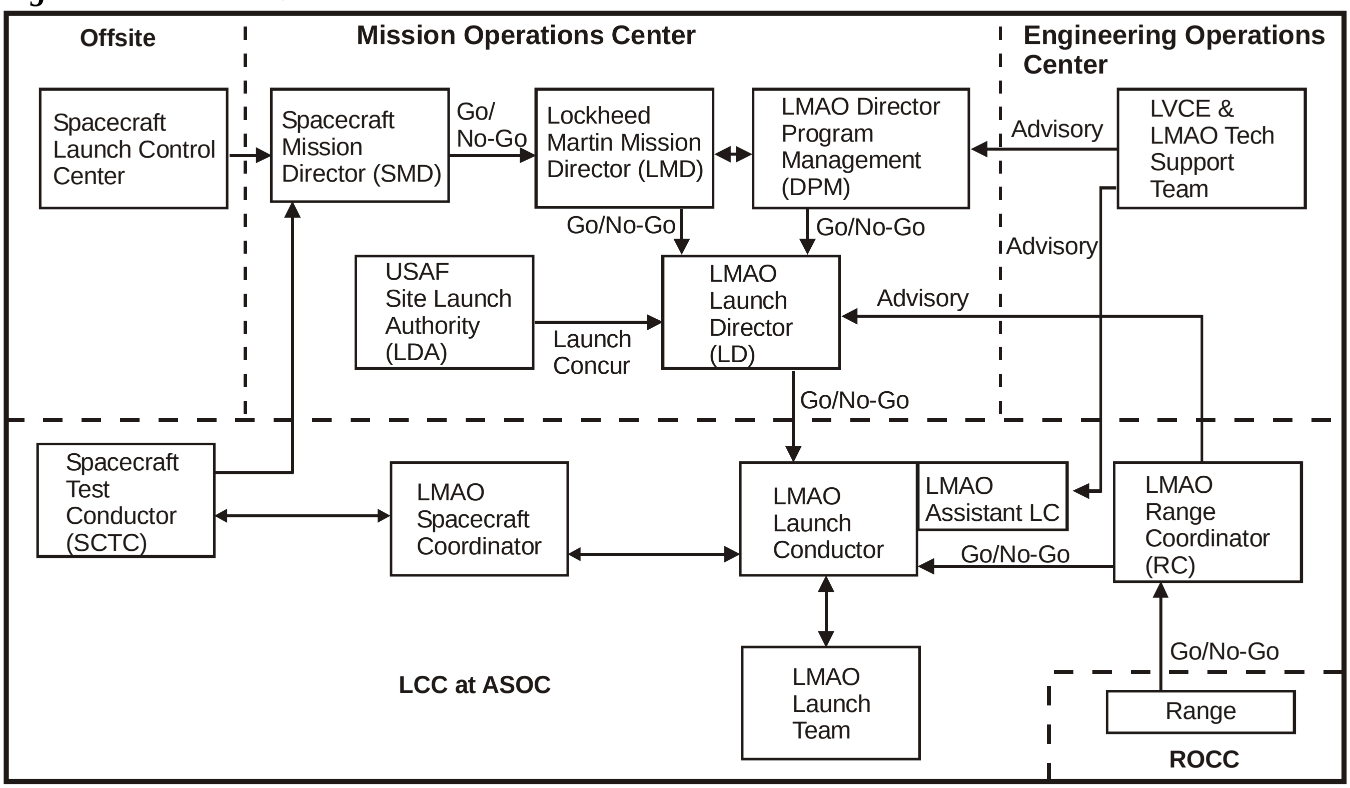


Figure 7.5.4-1 Typical Atlas V Launch Day Management Flow Diagram

7.6 LAUNCH CAPABILITY

In addition to the scheduled holds, additional hold time can be scheduled for up to 2 hours under normal environmental conditions or until the end of the scheduled launch window, whichever comes first.

Launch window restrictions have typically been determined by the spacecraft mission requirements. The Atlas launch vehicle essentially does not have launch window constraints beyond those of the mission.

7.7 WEATHER LAUNCH CONSTRAINTS

In addition to mission-dependent launch window restrictions, the decision to launch depends on weather launch constraints. Weather launch constraints include cloud conditions, lightning, thunderstorms, and ground and upper atmosphere winds. Excessive winds during launch may cause overloading of the vehicle structure and control system. Limiting conditions have been well defined and operational approaches developed to ensure launch within safe limits.

The decision to launch may be constrained if significant weather and/or thunderstorm conditions exist in the proximity of the launch site or the planned vehicle flight path at the time of liftoff. A go/no-go decision to launch is made by the Lockheed Martin launch director based on the following information.

7.7.1 Avoidance of Lightning

Lightning or equivalent weather conditions within a certain distance from the launch complex require a halt to all pad operations. A complete list of launch vehicle lightning constraints is in the appropriate generic launch vehicle program requirements document (PRD). Several conditions, as identified below, are used in determining operations criteria:

1) Do not launch for 30 minutes after any type of lightning occurs in a thunderstorm if the flight path will carry the vehicle within 18 km (10 nmi) of that thunderstorm, unless the cloud that produced the lightning has moved more than 18 km (10 nmi) away from the planned flight path.
2) Do not launch if the planned flight path will carry the vehicle:
    a) Within 18 km (10 nmi) of cumulus clouds with tops higher than the -20°C level;
    b) Within 9 km (5 nmi) of cumulus clouds with tops higher than the -10°C level;
    c) Through cumulus clouds with tops higher than the -5°C level;
    d) Within 18 km (10 nmi) of the nearest edge of any thunderstorm cloud, including its associated anvil;
    e) Through any cumulus cloud that has developed from a smoke plume while the cloud is attached to the smoke plume, or for the first 60 minutes after the cumulus cloud is observed to have detached from the smoke plume.
3) Do not launch if at any time during the 15 minutes before launch time, the absolute electric field intensity at the ground exceeds 1 kV/m within 9 km (5 nmi) of the planned flight path, unless there are no clouds within 18 km (10 nmi) of the launch site. This rule applies for ranges equipped with a surface electric field mill network.
4) Do not launch if the planned flight path is through a vertically continuous layer of clouds with an overall depth of 1,370 m (4,500 ft) or greater, where any part of the clouds is located between the 0 and -20°C temperature levels.
5) Do not launch if the planned flight path is through any cloud types that extend to altitudes at or above the 0°C temperature level and are associated with the disturbed weather within 9 km (5 nmi) of the flight path.
6) Do not launch if the planned flight path will carry the vehicle through thunderstorm debris clouds or within 9 km (5 nmi) of thunderstorm debris clouds not monitored by a field network or producing radar returns greater than or equal to l0 dBz.

7) Good Sense Rule—Even when constraints are not violated, if any other hazardous conditions exist, the launch weather team reports the threat to the launch director. The launch director may hold at any time based on the instability of the weather.

7.7.2 Ground Winds Monitoring

The Atlas launch vehicle is subject to ground wind restrictions during vehicle erection and assembly, after tower rollback up to the time of launch, and at launch. Lockheed Martin has an established ground winds restriction procedure that provides limiting wind speeds for all ground winds critical conditions. In addition, the document provides insight into the nature of ground winds loadings and possible courses of action should the wind speed limits be attained. The ground winds restriction procedure also contains limiting wind speeds during Atlas/Centaur erection, hoisting, and payload hoisting.

The ground winds monitoring system is designed to monitor vehicle loads after mobile service tower (MST) rollback and before launch. This is accomplished by sampling flight rate gyro rotational velocities (pitch and yaw signals), ground winds anemometer speed, ground wind directional azimuth, tanking levels, and tank ullage pressures. Data are processed, providing the ground winds monitor with a ground winds load ratio (LR) that represents the maximum load-to-limit allowable ratio in the vehicle or launcher at any given time. In addition, the LR is presented from the computer-controlled launch set (CCLS) using a present ground winds monitoring station strip chart. This system requires the presence of a ground winds monitor (one person) to evaluate the plotted and printed output data and immediately inform the launch conductor whenever the LR is approaching an out-of-tolerance condition.

7.7.3 Flight Wind Restrictions

Most loads experienced by the Atlas vehicle in flight can be calculated well before the vehicle's launch date. However, one major loading condition induced by the prevailing atmospheric winds (called flight wind profile) must be accounted for just before launch if maximum launch availability and mission success are to be ensured during marginal weather conditions.

On each mission, the pitch and yaw program is designed on launch day based on the actual launch day winds as determined from launch site weather balloon soundings. This capability is provided by a computer software program called Automatic Determination and Dissemination of Just Updated Steering Terms (ADDJUST) performed on Lockheed Martin Denver-based computer systems. Specifically, ADDJUST makes it possible to accomplish the following automatically:

1) Design an Atlas booster phase pitch/yaw program pair based on wind data measured at the launch site during the launch countdown;
2) Determine whether the wind profile loads and engine angles violate the vehicle's structural and control constraints;
3) Transmit the designed programs to the CCLS computer at the launch site launch control facility (for subsequent loading into the flight computer) with verification of correct transmittal of data.

Wind Sounding Procedure—Operations begin with the release of weather balloons from the launch range at specific intervals before launch. Raw wind data obtained from each balloon sounding are computer-reduced by range weather personnel to wind speed and direction data.

ADDJUST Program Procedure—Launch site wind data are received by computers at Lockheed Martin in Denver, CO, via data phone and automatically verified. The ADDJUST design and verification sequence is then executed. The resulting pitch and yaw program pair designed by ADDJUST is available for transmission back to the launch site approximately 10 minutes after Denver completes reception of the wind data. Pitch and yaw data transfer occurs directly from the Denver computer to the launch site backup CCLS computer via standard telephone lines. Simultaneously with this transmission, ADDJUST will proceed with loads validation computations, checking predicted loads and engine angles resulting from the pitch and yaw program design versus vehicle structural and control allowables. This will be followed by an engineering trajectory simulation run to check all trajectory-related parameters.

Launch Recommendations—With the ADDJUST-generated programs, all constraints must be satisfied before a "go" recommendation for launch may be made. The ADDJUST designer was developed so that the trajectory related constraints resulting from the engineering trajectory simulation would be satisfied. While the designer minimizes angle of attack, it cannot design pitch and yaw programs for a specific chosen set of loads.

7.8 LAUNCH POSTPONEMENTS

7.8.1 Launch Abort and Launch Vehicle 24-hour Recycle Capability

Before T-4 seconds (when the upper stage aft panel is ejected), the launch vehicle has a 24-hour turnaround capability after a launch abort due to a nonlaunch vehicle/GSE problem.

7.8.2 Launch Abort and Vehicle 48-hour Recycle Requirements

A launch abort after T-4 seconds and before T-0.7 seconds requires a 48-hour recycle. The principal reason for a 48-hour recycle versus a 24-hour recycle is the added time requirement for replacing the upper stage aft panel (ejected at T-4 seconds) and removal and replacement of the propellant pressurization line pyrovalves (fired upon aft panel ejection).

8.0 ATLAS SYSTEM ENHANCEMENTS

Maximizing on the combined strengths of the engineering, development, and production functions of Lockheed Martin, several Atlas launch systems enhancements are being implemented or planned to maintain the competitiveness of Atlas in the launch services market. Several initiatives are being supported to expand the operational capability of the Atlas/Centaur vehicle. And finally, looking toward 21st century requirements, several major Atlas modifications are planned and initial engineering is beginning. This section describes the initiatives being pursued by the Atlas program.

8.1 MISSION-UNIQUE ENHANCEMENTS

8.1.1 Spacecraft Dispensers and Universal Spacecraft Separation Node

To support multiple payload launches on Atlas, the program has predesigned several different configurations of payload "dispensers." Dispensers for both radial and longitudinal spacecraft separation have been investigated. Figure 8.1.1-1 illustrates a radial release dispenser that accommodates four spacecraft. In addition to primary structure design, significant emphasis has been placed on the design of an efficient spacecraft attach point. The result of these efforts is illustrated in Figure 8.1.1-2. The Universal Spacecraft Separation Node (USSN) is designed to maximize the capabilities of a given separation device by using unique load carrying features. It has the flexibility to be adapted to a variety of spacecraft and dispenser configurations and can accommodate a fast-acting shockless separation nut (FASSN).

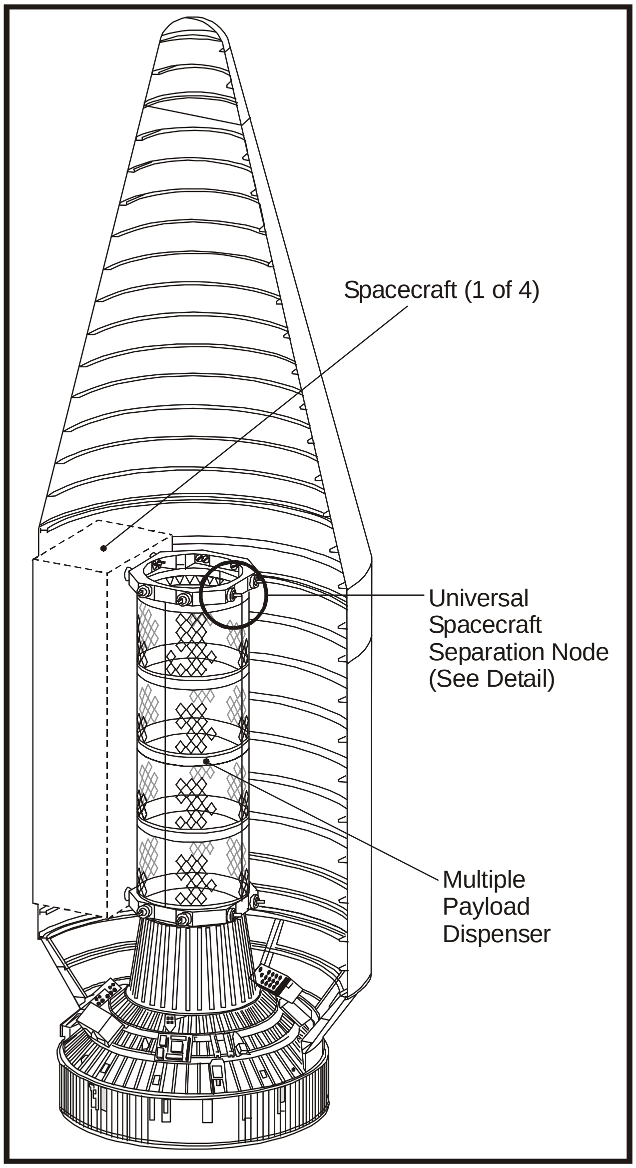


Figure 8.1.1-1 Multiple Payload Dispenser for four spacecraft—shown in the Atlas EPF.

The FASSN provides virtually shockless and instantaneous release of the spacecraft by converting separation bolt strain energy into rotational kinetic energy and by the elimination of pyrotechnic devices. A 12.7-mm (½-in.) diameter version of this system has been qualified for flight, including more than 450 successful activations, and successfully flew on a Lockheed Martin-built spacecraft in 1998. A 19.05-mm (¾-in.) diameter version of the FASSN is under development. Although the spacecraft interface shown in these figures is the baseline used in prototype design, the node can be modified to optimize individual spacecraft needs. The USSN allows a simple bolted interface between the launch vehicle and spacecraft; this significantly simplifies mechanical interface integration efforts.

8.1.2 Dual Payload Carrier

Lockheed Martin has completed the preliminary design for a dual payload carrier (DPC) for use on the Atlas V with the 5-m Medium payload fairing. The DPC will give the Atlas V the capability to simultaneously carry two medium- or intermediate-class payloads. The DPC will be adjustable in height to accommodate payloads of different heights.

The DPC fits entirely within the 5-m Medium fairing and divides the fairing volume into two payload compartments (Fig. 8.1.2-1). The forward compartment payload static envelope has a diameter of 4,572 mm (180.0 in.) at the base, and conforms to the ogive shape of the fairing as it extends forward. The forward payload mates to an Atlas or customer-provided adapter that in turn mates to the 1,575-mm (62.01-in.) diameter forward interface ring of the DPC.

The aft payload is encapsulated by the DPC and mates to an Atlas or customer-provided payload adapter that in turn mates to the 1,575-mm (62.01-in.) diameter forward interface ring of a C-type adapter. This C-type adapter mates to the top of the Centaur forward adapter. This aft compartment will have a static payload envelope with a minimum 4,000-mm (157.5-in.) diameter. Preliminary forward and aft payload static envelopes are shown in Figure 8.1.2-2.

The DPC provides access to the aft payload through standard 600-mm (23.6-in.) diameter doors, and can provide accommodations for a reradiating system when required. Ports in the DPC structure will ensure adequate conditioned air passes through the aft compartment to maintain the required thermal environment for the aft payload. The dispenser forward load reactor, near the top of the DPC cylinder section, controls relative motion between the DPC and payload fairing before fairing jettison.

The DPC structure, a lightweight, carbon fiber reinforced composite sandwich structure, attaches to the forward interface of the 4,394-mm (173-in.) truss adapter.

To facilitate DPC jettison after forward payload deployment, the DPC cylinder section contains a separation ring near its aft end. This pyrotechnic, frangible joint-type separation system contains all combustion byproducts and debris. After separation system actuation, the DPC forward portion is pushed away from the aft payload by a set of force-balanced springs, ensuring adequate clearance is maintained between the DPC and aft payload. The Centaur then turns to the required separation attitude and commands aft payload separation.

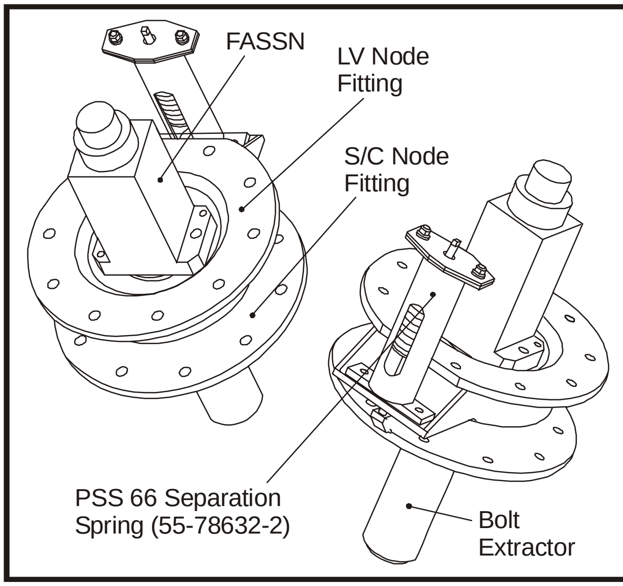


Figure 8.1.1-2 The Universal Spacecraft Separation Node (USSN)—for Use with "Hard-point" Interface Spacecraft

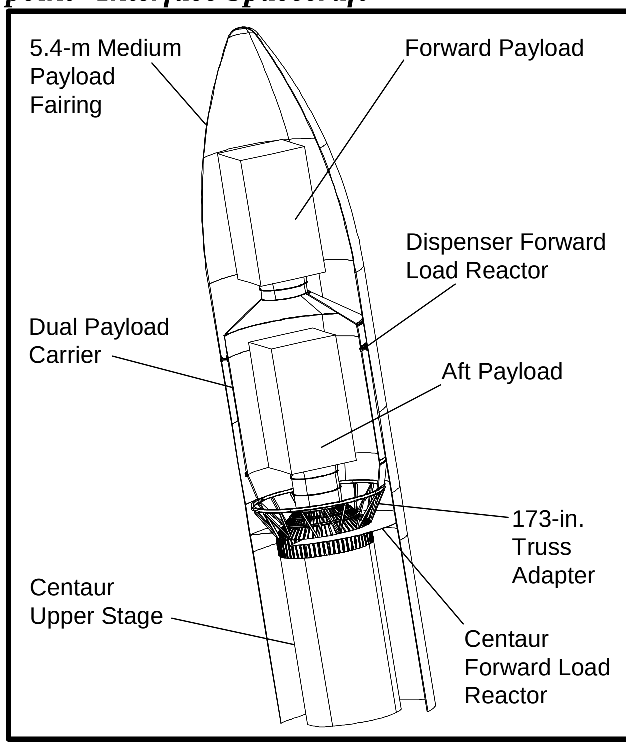


Figure 8.1.2-1 Atlas V 500 Series Dual Payload Carrier

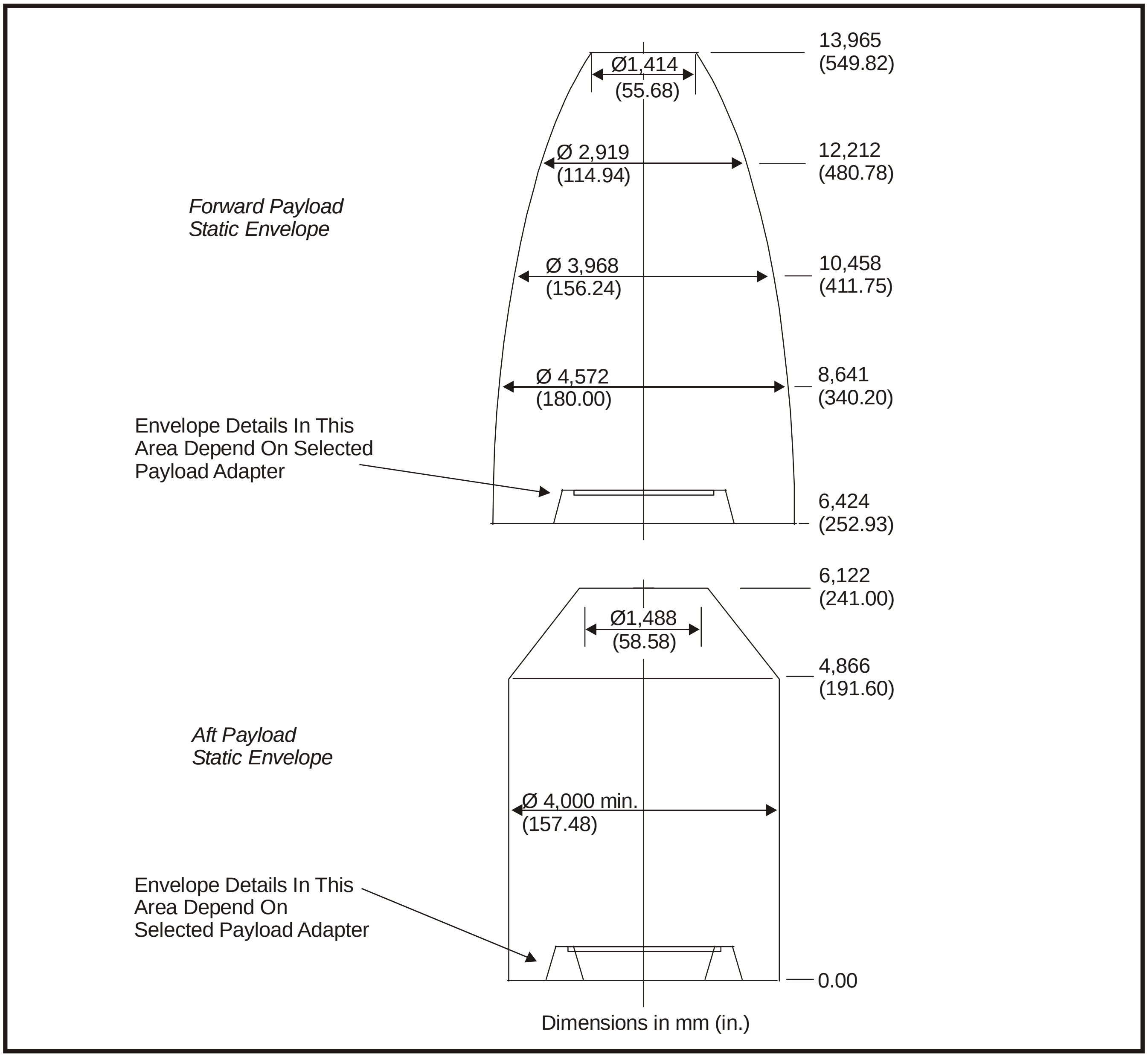


Figure 8.1.2-2 Preliminary Payload Envelopes

8.1.3 Atlas V Auxiliary Payload Service

Atlas now has a standard capability to integrate and launch auxiliary payloads to GTO and other potential orbits. This is accomplished through a unique partnership with our auxiliary satellite broker. This technical and managerial partnership will deliver customer-focused launch services to the auxiliary payload community while maintaining schedule and mission success. Please contact ILS for detailed information (contact information is in the foreword section of this document).

8.2 ATLAS EVOLUTIONARY ENHANCEMENTS

8.2.1 Atlas V 441 Vehicle

The Atlas V launch vehicle includes the provisions to accommodate the addition of up to five solid rocket boosters (SRB). Up to three SRBs are part of the current Atlas V 400 baseline. The Atlas V 441 vehicle may be available for use in response to launch service market requirements, however, no system analyses of this configuration have been performed. The Atlas V 441 performance capability for commercial missions (2.33-sigma performance) to GTO is approximately 6% over the Atlas V 431 payload systems weight (PSW) capability.

8.2.2 Atlas V Heavy Lift Vehicle (HLV)

The Atlas V family of launch vehicles—the Atlas V 400, Atlas V 500, and Atlas V Heavy Lift Vehicle (HLV)—is based on the newly developed 0.32-m (12.5-ft) diameter Common Core Booster™, which is powered by the RD-180 engine. The Atlas V maximizes commonality between launch vehicles by using a single booster for the core and as strap-on boosters to make the Atlas V HLV configuration. Each booster is powered by a single RD-180 engine, the same engine that flies on the Atlas III launch vehicle. Commonality continues with a Centaur III upper stage for the Atlas V 400, 500, and HLV configurations. Common interstage adapters, a common avionics suite, and payload fairings provide the remaining building blocks needed for the Lockheed Martin Atlas V family configurations.

The HLV booster uses three CCBs (Fig. 8.2.2-1). All common-element design criteria envelop HLV requirements. The HLV-unique hardware is minimal: (1) liquid rocket booster (LRB) nose cones and attach hardware; (2) LRB staging rockets; (3) 5-m Long PLF; and (4) ATS and intertank modifications.

The Atlas V HLV performance capability for commercial missions (2.33 sigma performance) is: GSO—6,350-kg PSW capability; GTO—12,650-kg PSW capability; and LEO—25,000-kg PSW capability. Spacecraft exceeding 19,050-kg in mass may require mission-unique accommodations.

8.3 ATLAS V SHOCK ENHANCEMENTS

8.3.1 Atlas V 500 Shock Levels

Figure 8.3.1-1 shows maximum allowable spacecraft-produced shock levels at the forward interface to the payload truss adapter. This capability is planned to be available by mid-2002.

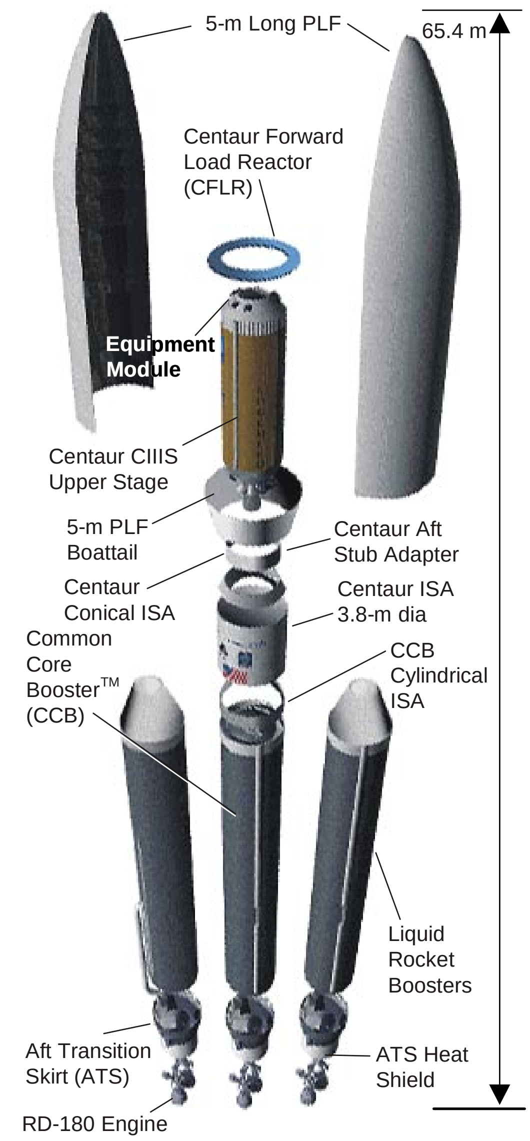
8.4 ATLAS V HEAVY-LIFT ENHANCEMENTS

8.4.1 Heavy Lift Payload Truss

A 3,302-mm (130-in.) truss has been designed and built for the upper stage acoustic, modal and static load testing. Figure 8.4.1-1 shows this truss attached to the Centaur. The truss was designed as a load fixture to input loads to the Centaur similar to a flight truss. Figure 8.4.1-1 shows the struts, which were made from aluminum tubes. A flight truss would have struts made from graphite/epoxy tubes and the forward ring would be an aluminum structural shape. A truss similar to this one with a spacecraft interface from 2,972 mm (117 in.) to 4,394 mm (173 in.) could be available as a mission-unique option for heavy payload needs (Fig. 8.4.1-2). This mission-unique interface is being designed to carry a heavy-lift spacecraft up to 20,400 kg (45,000 lb). The 914-mm (36-in.) high truss interfaces with the Centaur forward adapter at the same 12 mounting locations as the 4,394-mm truss. Similar to the 4,394-mm truss, the struts would be graphite epoxy with titanium end fittings, and the forward and aft brackets and the forward ring would be aluminum alloy. Spherical bearings are in each end of each strut to allow only tension or compression loads in each strut. This truss would be used inside the 5-m PLF on either the Atlas V 500 or Atlas V HLV configurations for heavy payloads (payloads over 9,072 kg [20,000 lb]).

8.4.2 Atlas V 5-m Long Payload Fairing (PLF)

Concurrently with the Atlas V HLV, the Atlas program is developing the 5-m Long PLF (Fig. 8.2.2-1). This fairing is derived from the 5-m Medium PLF described in Appendix D with the lower module stretched by 2.96 m (9.7 ft), giving the fairing an overall length of 26.5 m (87 ft). This results in a useable payload volume 10.588-m (416.86-in.) high in the cylindrical section of the fairing (11.43 m [450 in.] for a 4.572-m [180-in.] diameter cylinder) and a total envelope length of 15.882 m (625.27 in.), including the volume available in the ogive section of the fairing. The design of this fairing is being



PAYLOAD FAIRING

Features	**5-m Long**
Diameter:	5.4 m
Length:	26.5 m
Mass:	4,775 kg
Subsystems	
Fairing:	Bisector; Sandwich Construction with Graphite Epoxy Face Sheets & an Aluminum Honeycomb Core
Separation:	Vertical Separation by a Linear Piston & Cylinder Activated by a Pyrotechnic Cord; Horizontal Separation by an Expanding Tube Shearing a Notched Frame, Activated by a Pyrotechnic Cord

CENTAUR UPPER STAGE (CIII)

Features	
Size:	3.05-m dia x 12.68-m Length with Extended Nozzle
Inert Mass:	2,316 kg
Propellant:	20,860-kg LH_2 & LO_2
Guidance:	Inertial
Subsystems	
Structure:	Pressure Stabilized Stainless Steel Tanks Separated by Common Ellipsoidal Bulkhead
Propulsion:	One Pratt & Whitney Restartable Engine
— Model:	RL10A-4-2
— Thrust:	99.2 kN (SEC)
— I_{SP}:	450.5 s
(SEC)	One Electromechanically Actuated 51-cm Columbium Fixed Nozzle Four 27-N Hydrazine Thrusters Eight 40.5-N Lateral Hydrazine Thrusters
(DEC)	Two Hydraulically Actuated 51-cm Columbium Extendible Nozzles
(12)	40.5-N Lateral Hydrazine Thrusters
Pneumatics:	Common with Atlas V 40X
Avionics:	Common with Atlas V 40X
Insulation:	Radiation Shields

LIQUID ROCKET BOOSTERS (LRB)

Features	
Size:	3.81-m dia x 35.99-m Length
Propellant:	286,795-kg LO_2 & RP-1 (Each)
Tank Mass:	21,902 kg
Thrust:	4,152-kN VAC 100% Thrust (Each)
I_{SP}:	337.8 s
Structure:	Same as Common Core Booster

CENTAUR INTERSTAGE ADAPTER (C-ISA LARGE)

Features	
Size:	3.81-m dia x 3.53-m Length
Mass:	1,096 kg
Subsystems	
Structure:	Composite Sandwich (Aluminum Core/Graphite Epoxy Face Sheets)

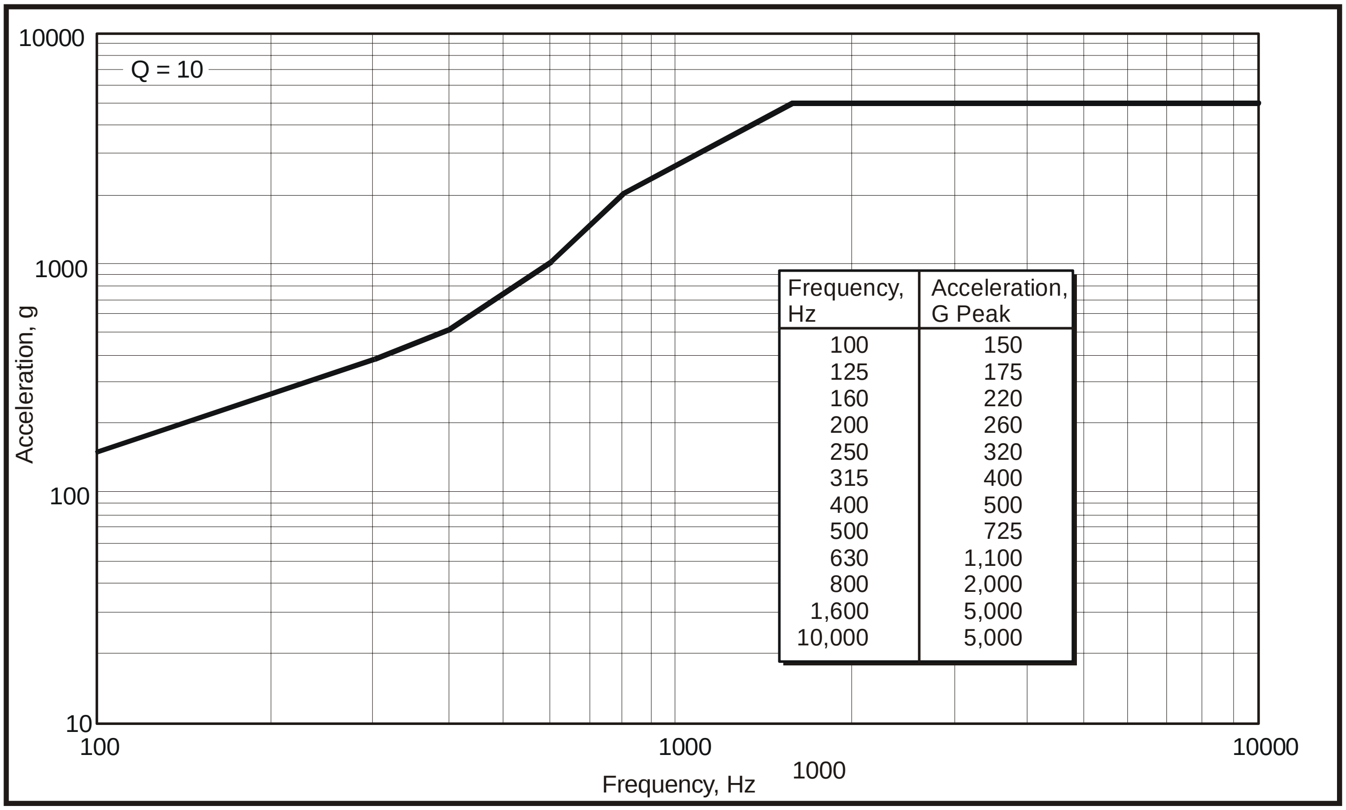
CCB CYLINDRICAL INTERSTAGE ADAPTER

Features	
Size:	3.81-m dia x 0.60-m Length
Mass:	278 kg
Subsystems	
Structure:	Aluminum Machined Rolled-Ring Forging

COMMON CORE BOOSTER™ (CCB)

Features	
Size:	3.81-m dia x 32.46-m Length
Propellant:	286,795-kg LO_2 & RP-1
Inert Mass:	23,648 kg
Guidance:	From Upper Stage
Subsystems	
Structure:	Structurally Stable Aluminum Isogrid Tanks; Integrally Machined Aft Transition Structure; Composite Heat Shield
Separation:	8 Retro Rockets
Propulsion:	Pratt & Whitney/NPO Energomash RD-180 Booster Engine (2 Chambers) SL 100% Thrust = 3,827 kN, I_{SP} = 311.8 s Vac 100% Thrust = 1,951 kN, I_{SP} = 337.8 s
Pneumatics:	Helium (Pressure Tanks), Computer-Controlled Pressurization System
Hydraulics:	Integral with Engine Provides Gimbal Control
Avionics:	Flight Control, Flight Termination, Telemetry, Redundant Rate Gyros, Electrical Power

Figure 8.2.2-1 Atlas V HLV Characteristics



Frequency, Hz	Acceleration, G Peak
100	150
125	175
160	220
200	260
250	320
315	400
400	500
500	725
630	1,100
800	2,000
1,600	5,000
10,000	5,000

Figure 8.3.1-1 Maximum Allowable Spacecraft-Produced Shock at Forward Interface to Payload Truss Adapter

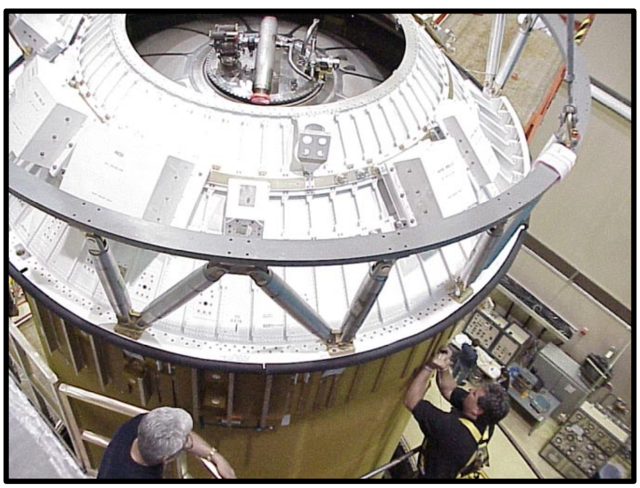


Figure 8.4.1-1 3,302-mm Test Truss

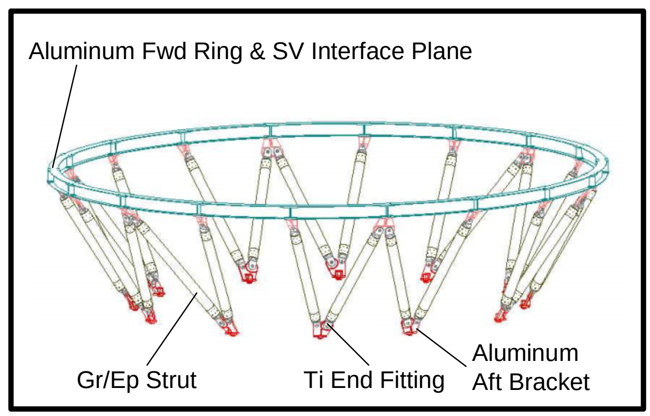


Figure 8.4.1-2 4,394-mm Diameter Payload Truss

completed through a critical design review stage with further development dependent on market demand. Final development and qualification would take approximately 36 months from an initial order. The 5-m Long PLF on the Atlas V 500 series may be available for use in response to launch service market requirements, however, no system analyses of this configuration have been performed.

APPENDIX A—ATLAS HISTORY, VEHICLE DESIGN, AND PRODUCTION

The Atlas Centaur launch vehicle is manufactured and operated by Lockheed Martin to meet commercial and government medium, intermediate, and heavy space lift needs.

A.1 VEHICLE DEVELOPMENT

Atlas Booster—The Atlas program began in the mid-1940s with studies exploring the feasibility of long-range ballistic missiles. The Atlas launch vehicle family has evolved through various United States Air Force (USAF), National Aeronautics and Space Administration (NASA), and commercial programs from the first research and development (R&D) launch in 1957 to the current Atlas II, III, and V configurations (Fig. A.1-1). To date, more than 565 Atlas vehicles have flown.

Versions of Atlas boosters were built specifically for manned and unmanned space missions, including the pioneering Project Mercury manned launches that paved the way toward the Apollo lunar program. The addition of the high-energy Centaur upper stage in the early 1960s made lunar and planetary missions possible. In 1981, the Atlas G booster improved Atlas Centaur performance by increasing propellant capacity and upgrading engine thrust. This baseline was developed into the successful Atlas I, II, Atlas IIA, IIAS, and IIIA launch vehicles.

Atlas IIIB, with an initial launch capability (ILC) in 2001, and Atlas V, with an ILC in 2002, continue the evolution of the Atlas family. Today, as the world's most successful launch vehicle, the Atlas is offered in a comprehensive family of configurations that efficiently meet payload requirements (Fig. A.1-2).

Atlas V, the nation's next-generation space launch vehicle, is the most flexible, robust, and reliable launch vehicle system offered by Lockheed Martin. The Atlas V system is capable of delivering a diverse array of satellite payloads, including all projected government missions to low Earth orbit (LEO),

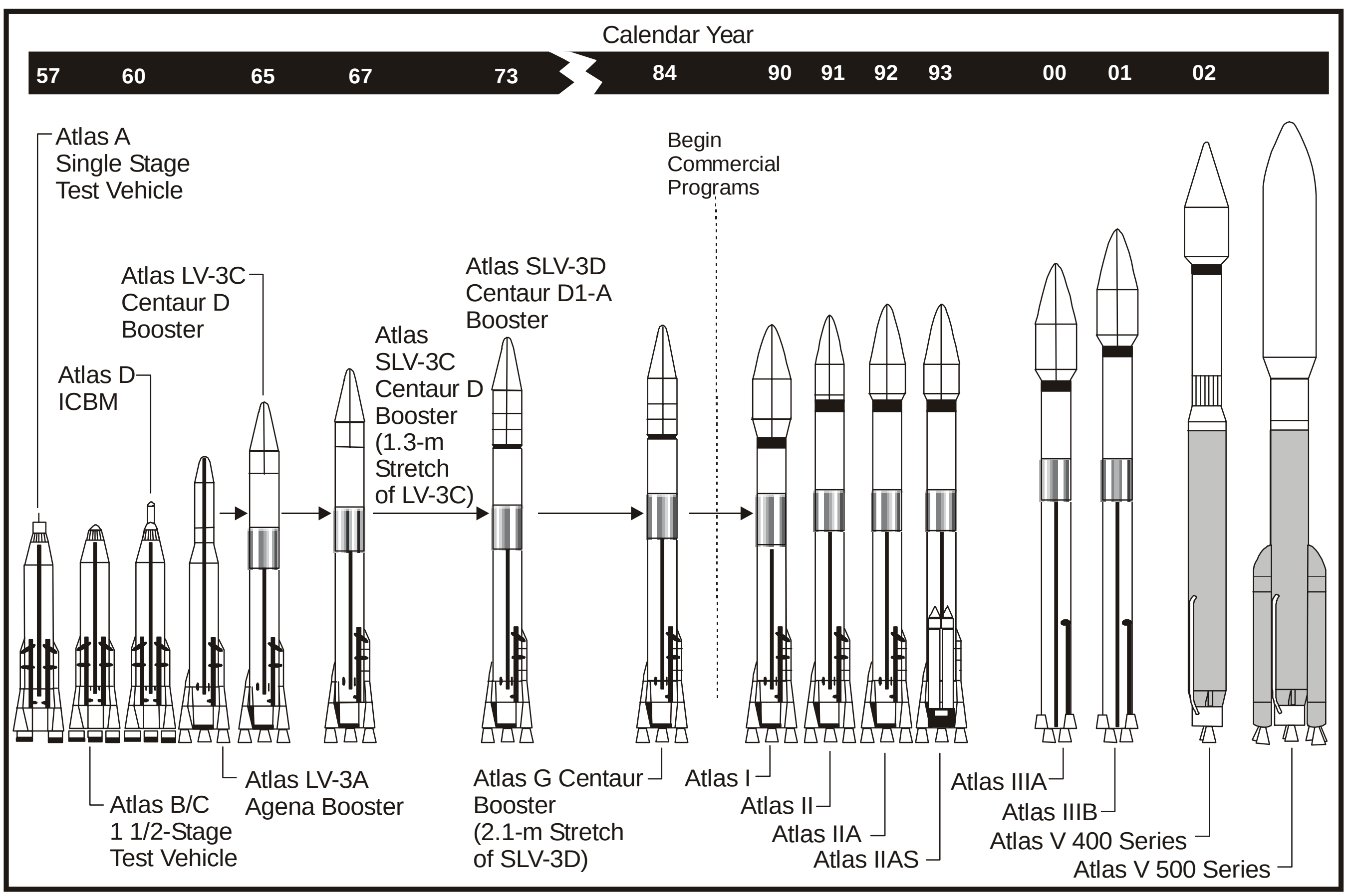


Figure A.1-1 Atlas has successfully evolved to satisfy requirements of many missions.

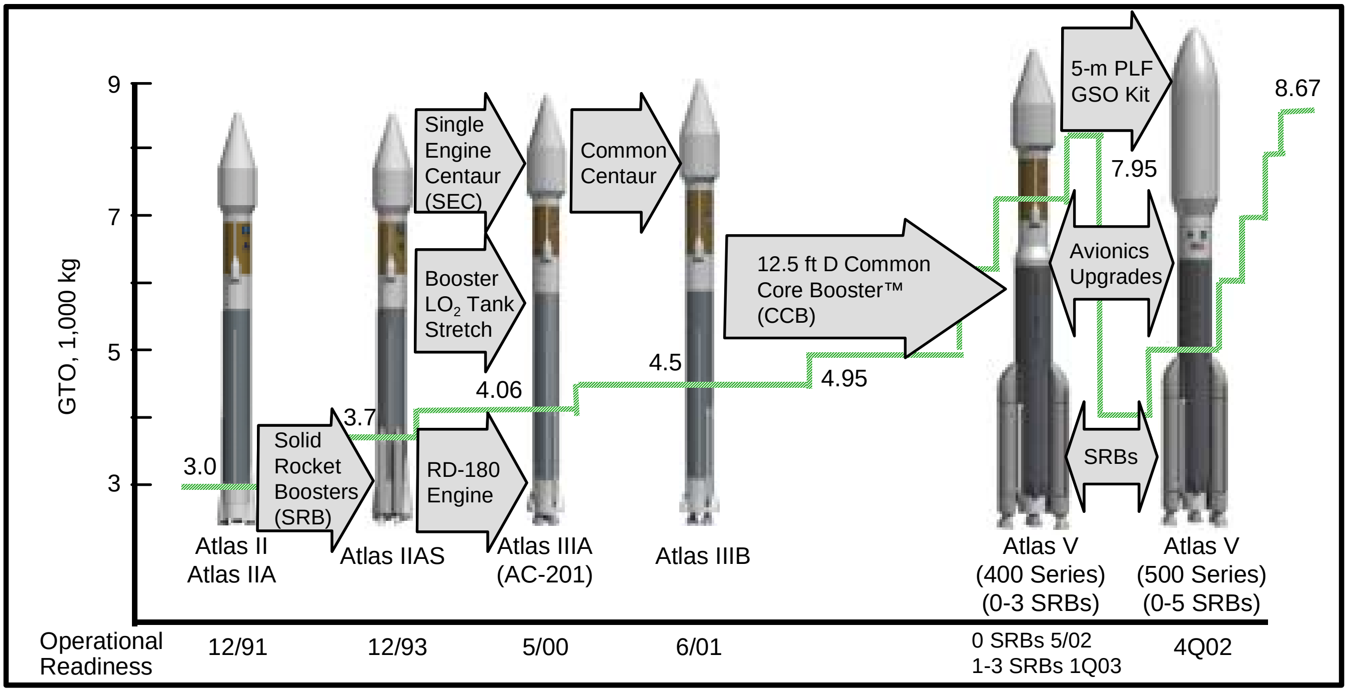


Figure A.1-2 Current Atlas Options and Performance Evolution

heavy lift geosynchronous orbits (GSO), and numerous geostationary transfer orbits (GTO). To perform this variety of missions, Lockheed Martin combines a Common Core Booster™ (CCB) powered by a single RD-180 engine with a standard Atlas large payload fairing or extended length large payload fairing to create the Atlas V 400 series. For larger and heavier payloads, the Atlas V 500 series combines the CCB with a 5-m diameter payload fairing available in three lengths. The Atlas V Heavy Lift Vehicle (HLV) has been designed to deliver the nations largest national security missions. It combines three CCBs with the longest 5-m payload fairing. The Atlas V 400 and 500 series include a Common Centaur upper stage that can be configured with either a single or dual engine, depending on mission needs. While the HLV requirements are integral to the Atlas V common system design, only the 400 and 500 series are initially offered for flight. The HLV implementation will be completed upon order for a mission requiring this configuration.

The Atlas V system provides increased reliability over its predecessors. The increased reliability is achieved through a simplified design that incorporates fault avoidance, fault tolerance, and reduction of single-point failures (SPF). The involvement of Production Engineering and Test Engineering in all phases of the design process has lead to producibility and testability improvements that have yielded streamlined and repeatable manufacturing processes. The ultimate result is a reduction in nonconformances and defects, which translates into more reliable processes and hardware.

The robustness of the Atlas V system is enhanced by the use of common system elements assembled into a family of vehicles that satisfy a wide range of mission requirements while providing substantial performance margins. In addition to common elements, the Atlas V system features improved structural capability allowing it to withstand worst-case day-of-launch winds. The result is increased launch availability.

At the launch site, standard launch vehicle elements, facilities, and equipment allow standardized launch site procedures and common processes. Because of increased emphasis on factory testing and delivery of flight-worthy elements to the launch site, vehicle processing time is greatly reduced. Limiting launch site testing to system-level tests virtually eliminates the need for intrusions into subsystems. It also reduces the probability of inadvertent damage and increases launch availability.

Performing simultaneous processing of the booster and upper stage and encapsulation of a preprocessed payload, all in separate facilities, reduces critical path activities on the launch pad.

Finally, the robustness and flexibility of the Atlas V enable use of a class analysis approach to mission integration. Class analyses are performed during the non-recurring phase of the program to encompass the known variations in individual vehicles and missions. As a result, mission integration times and costs are reduced, and mission success is maintained by eliminating recurring characterization of launch vehicles and minimizing recurring analyses.

Centaur Upper Stage—Development began on the Centaur high-energy (high specific impulse) upper stage in 1958 to launch NASA spacecraft on lunar and planetary missions. NASA's ambitious planetary exploration goals required development of improved avionics capable of guiding the Surveyor soft-lander space probes on lunar missions. Throughout the operational history of Centaur, systematic upgrades to the avionics have provided outstanding orbital insertion accuracy with favorable cost and weight. Centaur's evolution to the current Atlas IIAS dual-engine Centaur (DEC) and Atlas IIIA single-engine Centaur (SEC), and the Atlas IIIB and V Common Centaur DEC and SEC configurations is shown in Figure A.1-3.

The first successful flight of the Centaur upper stage on an Atlas launch vehicle in November 1963 was the world's first in-flight ignition of a hydrogen-powered vehicle. Three years later, Centaur performed the first successful space restart of LH_2 engines in October 1966. With this flight, the R&D phase was completed, and Centaur became fully operational. Multiple engine starts after long coast periods in space enable Centaur to provide exceptionally accurate targeting. The Centaur upper stage has flown more than 100 missions on Atlas and Titan boosters, ranging from communications satellites to

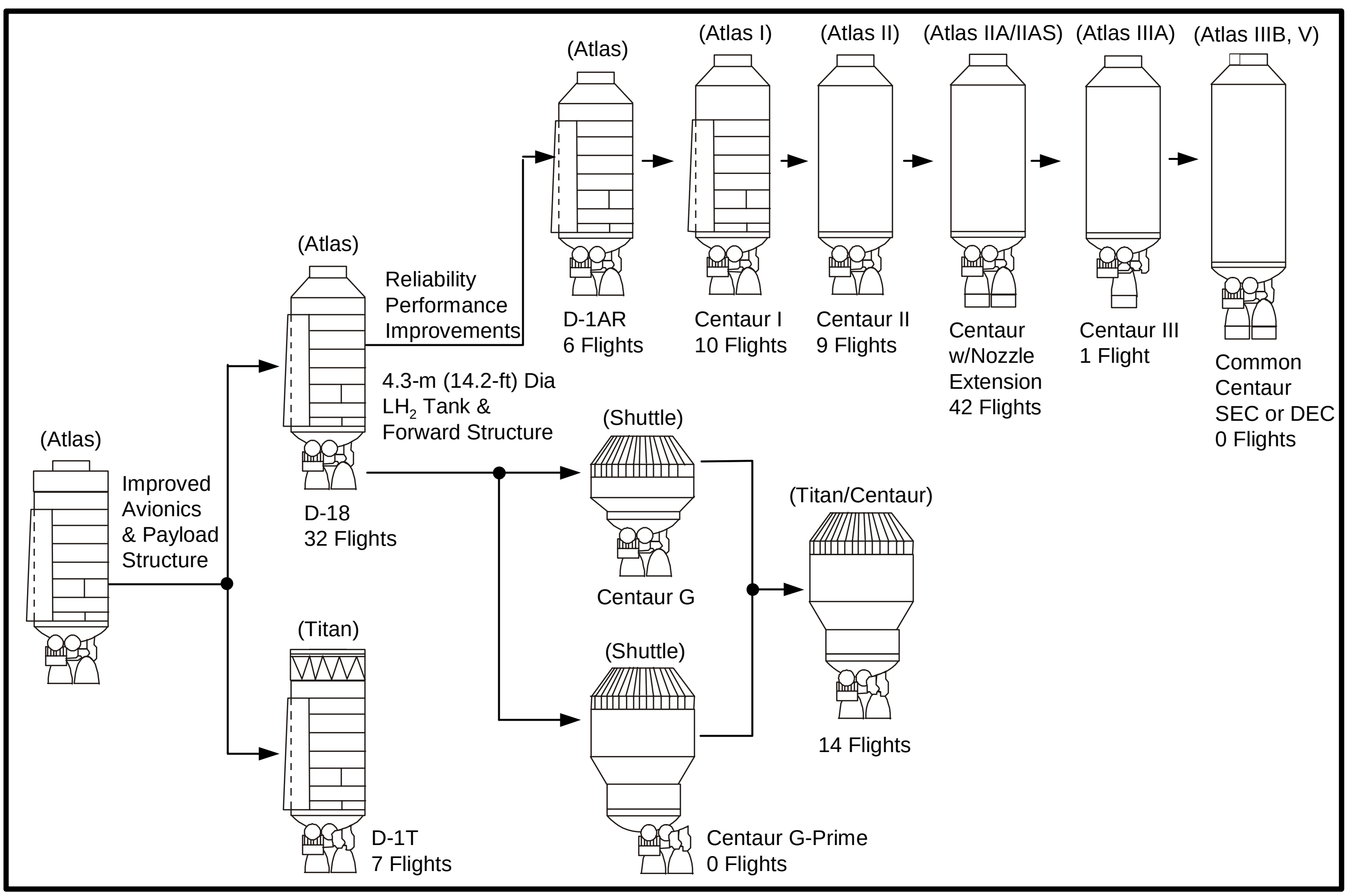


Figure A.1-3 Centaur evolution reflects changes to accommodate other boost vehicles and performance and reliability improvements.

earth orbit to the historic Mariner, Pioneer, Viking, and Voyager planetary exploration missions. Throughout this history, the design has been refined and enhanced, enabling Centaur to remain the most efficient, accurate upper stage in the world.

Atlas Centaur Launch System—More than 115 spacecraft have been integrated and launched on the Atlas Centaur space launch system during the past 35 years. Figure A.1-4, at the end of this appendix, illustrates the wide range of Atlas Centaur missions and launch services users. The delivery of lunar and planetary missions to precise orbits and the accommodation of diverse Earth-orbiting satellite platforms is noteworthy. Table A.1-1, at the end of this appendix, chronologically lists each Atlas Centaur flight with details of mission type and status. Fifty-five consecutive successful Atlas launches and 100% launch success for the Atlas II and III family is a reliability record unmatched in the industry.

A.2 VEHICLE DESIGN

The Atlas and Centaur stages for Atlas IIAS, III, and V are manufactured using common, qualified components and production procedures. The next sections describe the features of the Atlas family of launch systems from the Atlas IIAS to the Atlas V with an emphasis on evolutionary improvements. Atlas booster system characteristics are described first, followed by the Centaur upper stage system characteristics.

A.2.1 Atlas IIAS Major Characteristics

The Atlas IIAS, derived from the distinguished line of Atlas ICBMs and space launch vehicles, is a 1½-stage, liquid propellant (RP-1 fuel and LO_2 oxidizer) vehicle having a constant 3.05-m (10-ft) diameter and a total length of 24.9 m (81.7 ft). The full stage is the sustainer, which is powered from liftoff to propellant depletion. The half stage is the booster package, which is jettisoned during the Atlas powered phase. The Atlas IIAS is similar to the Atlas IIA, with the addition of strap-on solid rocket boosters to augment thrust during the boost phase. Improvements from earlier configurations include a hydrazine Atlas roll control module (ARCM), uprated Rocketdyne MA-5A engines, and stretched booster propellant tanks.

The Atlas IIAS primary propulsion is provided by the Rocketdyne MA-5A engine system, which includes one jettisonable, two-chamber booster engine and one single-chamber sustainer engine. Both engines are ignited before liftoff and develop a total rated thrust of 2,180 kN (490,000 lb_f). Nominal propulsion system operation is confirmed before committing the vehicle to flight. The IIAS is the final Atlas configuration to use the MA-5A engine. Four Thiokol Castor IVA solid rocket boosters (SRB), each generating 433.7 kN (97,500 lb_f), burn two at a time to augment thrust for the first ~110 seconds of flight. Two of the SRBs are ignited on the ground, and two are ignited in flight.

The Atlas stage avionics are integrated with the Centaur avionics system for guidance, flight control, instrumentation, and sequencing. An external equipment pod houses such Atlas systems as flight termination, data acquisition, pneumatics, and instrumentation.

A.2.1.1 Structure—The Atlas IIAS structure consists of three primary elements: the sustainer section, the thrust section, and the interstage adapter (ISA).

Sustainer Section—The pressure-stabilized fuel and oxidizer tanks are thin-wall, fully monocoque, welded, stainless-steel, and are separated by a common ellipsoidal bulkhead. Structural integrity of the tanks is maintained in flight by a fault-tolerant pressurization system and on the ground by internal tank pressure or by the application of external mechanical stretch. The sustainer section supports the sustainer engine, equipment pods, and booster jettison tracks.

The Atlas IIAS sustainer is functionally similar to the Atlas II sustainer, but includes an increase in tank skin gauges, the addition of the SRB load ring and forward cableway, and a revision to the booster staging system to accommodate the SRBs (Fig. A.2.1.1-1).

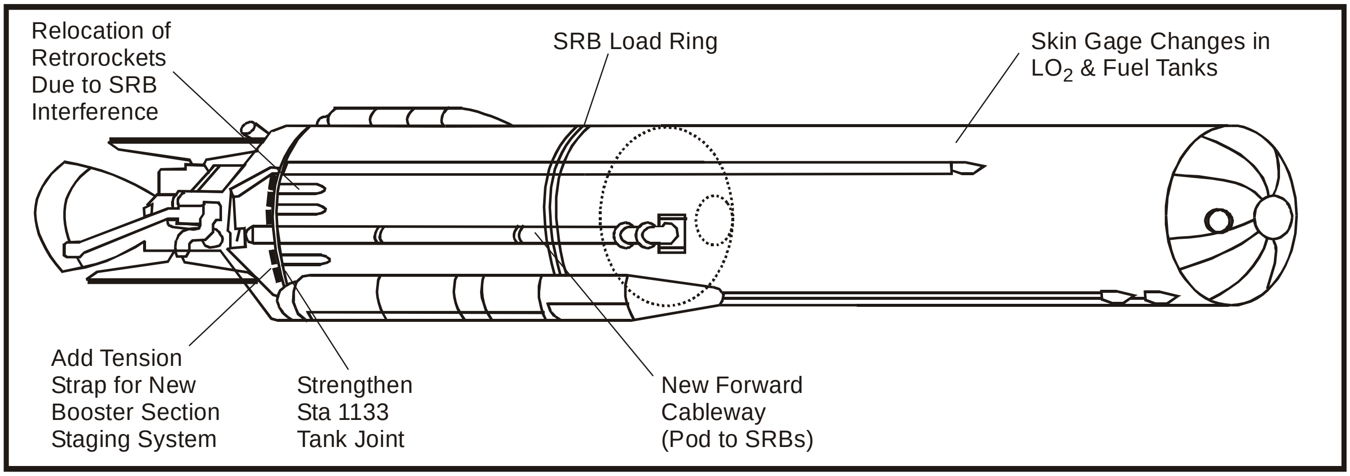


Figure A.2.1.1-1 Atlas IIAS Tank Design Updates

Thrust Section—The thrust section provides the structural support and an aerodynamic enclosure for the booster engine (Fig. A.2.1.1-2). The booster package is jettisoned about 165 seconds after launch. For the Atlas IIAS, integrally stiffened machined panels replace the II/IIA skin/stringer design to accommodate the SRBs (Fig. A.2.1.1-3). This thrust section design accommodates the increased base region heat loads and is sealed to keep out fuel-rich combustibles. Cork heat shields and engine boots limit the exposure of internal hardware to the base region heat loads. The thrust section is attached to the sustainer section with 10 pneumatic latches. The Atlas IIAS booster staging system components are designed to react Castor IVA SRB loads, and the Atlas IIAS booster staging system design is functionally redundant.

Figure A.2.1.1-2 The robust Atlas booster section is flight proven.

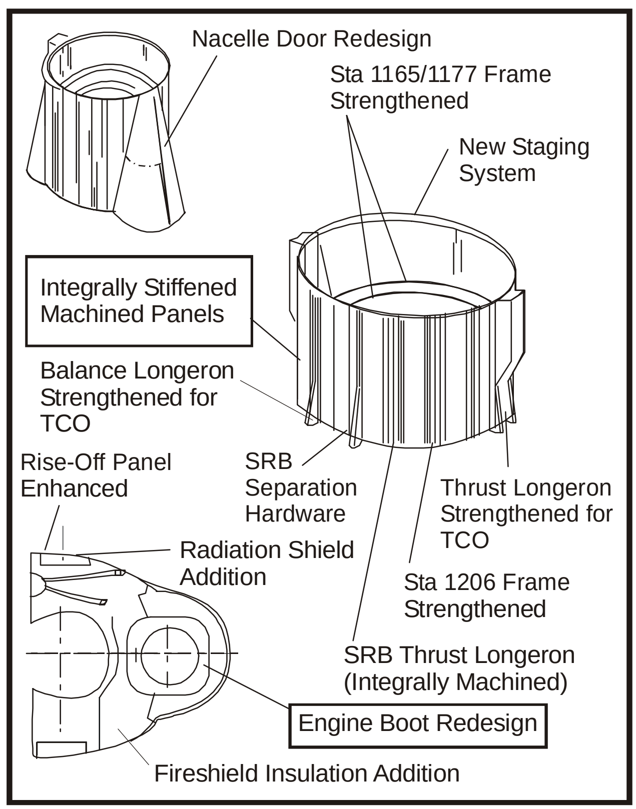


Figure A.2.1.1-3 The Atlas IIAS Booster Thrust Structure

Interstage Adapter—The ISA provides a physical connection between the Atlas and the Centaur stages (Fig. A.2.1.1-4). The 3.96-m (13-ft) long cylindrical adapter is of aluminum skin/stringer construction. The Atlas IIAS common adapter bolts to the Centaur aft ring and the Atlas forward ring. The ISA houses the hydrazine-based Atlas roll-control module (ARCM) and contains the Atlas/Centaur separation system.

Figure A.2.1.1-4 The interstage adapter provides the attachment between Atlas and Centaur stages.

Atlas/Centaur Separation System—A fault-tolerant, pyrotechnically actuated flexible linear-shaped charge (LSC) separation system is used to sever the ISA about 1.3-cm (0.5-in.) aft of the ISA/Centaur interface.

A.2.1.2 Pneumatics—Tank pressures are controlled to maintain structural integrity and provide a positive suction pressure to the engines.

Computer-Controlled Atlas Pressurization System (CCAPS)—The CCAPS is a closed-loop tank pressure control system using flight software logic and is active before liftoff through booster engine cutoff (BECO). It controls ullage pressure in the launch vehicle RP-1 and LO_2 tanks by adding helium as required. Atlas operates under continually decaying residual pressure throughout the sustainer phase of flight. Redundant pressurization solenoid and pyrovalves control tank pressures within the desired limits. Before liftoff, the ground system pressure control unit and independent ground system relief valves control tank pressurization and venting.

Helium Supply System—Ten storage bottles supply helium for the pneumatics system and for tank pressure control on the Atlas IIAS. This system is jettisoned with the booster section.

Engine Control System—This system consists of a single titanium bottle that supplies helium pressurant to the booster engine control package. A dedicated high-pressure helium bottle supplies the pneumatically operated booster section staging latches.

LO_2 Propellant Tank Vent System—An LO_2 boiloff valve controls the LO_2 propellant saturation pressure during cryogenic tanking. Just before launch, the valve is locked closed and remains locked throughout flight (because venting is not required).

A.2.1.3 Propulsion—The Atlas IIAS propulsion system consists of five flight-proven subsystems: the MA-5A engine system; SRBs; SRB attach, disconnect, and jettison (ADJ) system; ARCM; and ground start system.

MA-5A Engine System—Main propulsion is provided by the Rocketdyne MA-5A engine system. It is a calibrated, fixed-thrust, pump-fed system that uses a propellant combination of LO_2 (oxidizer) and RP-1 (fuel) for both the booster and sustainer engines (Fig. A.2.1.3-1). The MA-5A engine system uses two RS-27 engines packaged in a jettisonable booster engine system and one sustainer engine. The booster engine system is jettisoned during the Atlas boost phase after the prescribed optimum axial acceleration is achieved. After booster engine cutoff (BECO), separation latches open, propellant lines seal and disconnect, electrical connections disconnect, and the booster engine system slides away along a pair of guide rails. The single sustainer engine continues the boost phase until propellant depletion. All engines undergo hot-firing tests during individual engine acceptance tests.

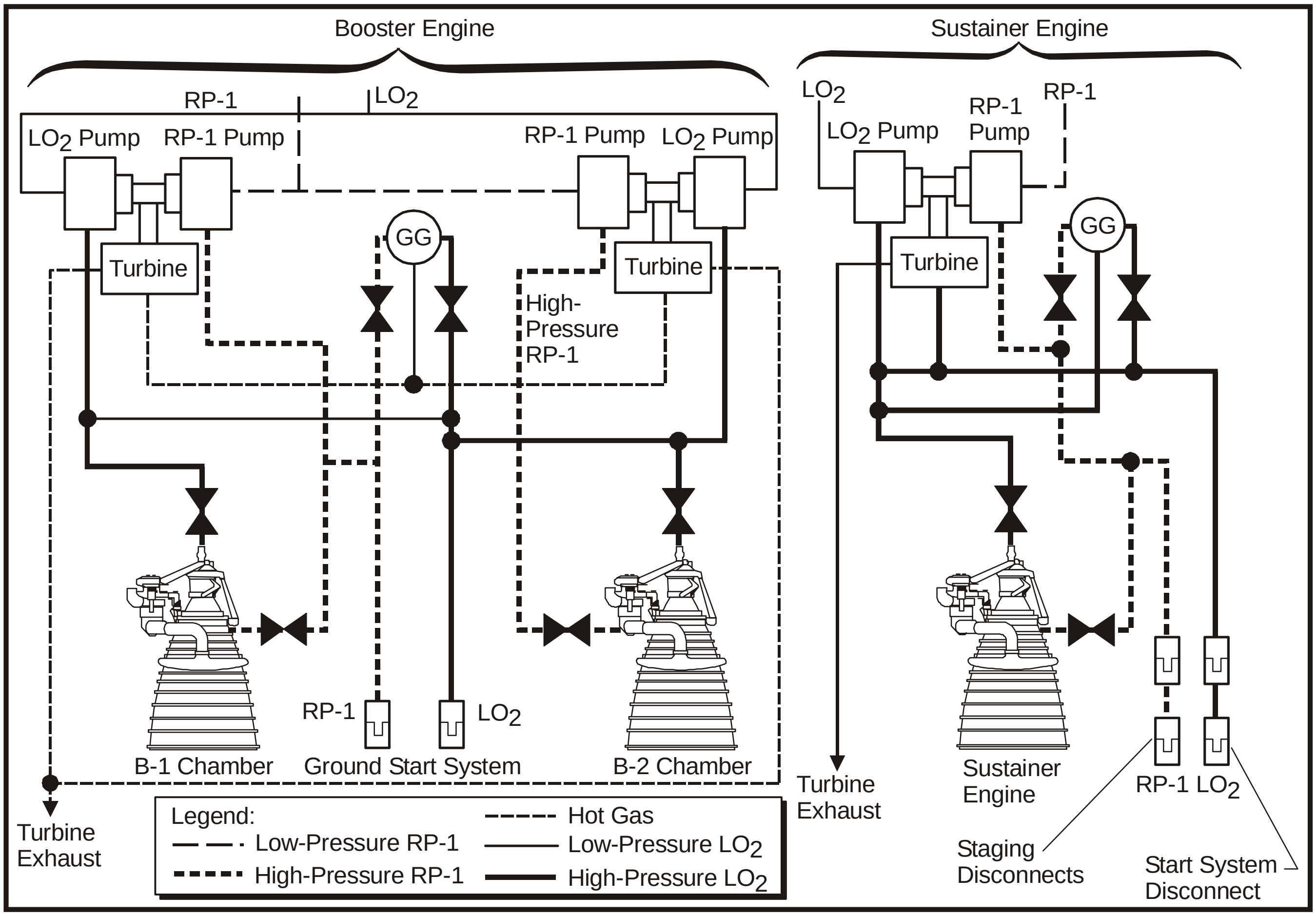


Figure A.2.1.3-1 The reliable flight-proven MA-5A engine system has demonstrated 100% Mission Success.

Solid Rocket Booster—The Atlas IIAS uses four Thiokol Castor IVA SRBs. Two are ignited at launch and two are ignited in flight after burnout of the ground-lit pair.

Attach, Disconnect, and Jettison System (ADJ)—The SRB ADJ system reacts SRB flight loads and safely jettisons the SRBs with significant outboard rotation.

ARCM—The MA-5A engine system provides roll control during the initial boost phase. The ARCM, which replaced the vernier engines used on Atlas I, provides roll control during the Atlas IIAS sustainer solo phase. The ACRM includes four hydrazine thrusters (222.4 N [50 lb_f] of initial thrust per unit) with two thrusters aligned in each roll direction, a blowdown hydrazine storage sphere, control valves, and associated plumbing.

Ground Start System—The ground start system supports ignition of the booster and sustainer engines. This system consists of pressure rated bottles for RP-1 and LO_2 with controls necessary to fill the bottles, pressurize them, feed RP-1 and LO_2 to the Atlas gas generators, and vent the system after the MA-5A engine start and before launch. The ground start system provides propellants needed to start the gas generators that drive the engine propellant pumps until their output flow and pressure is sufficient to feed the gas generators and main engines (booster and sustainer).

A.2.1.4 Hydraulic Thrust Vector Control (TVC)—The Atlas MA-5A hydraulic system is composed of two independent systems, one for the booster and one for the sustainer. Hydraulic pumps, driven from the engine turbopump accessory drive pads, provide hydraulic pressure and fluid flow to the servo actuator assemblies for gimbal control of the engine thrust chambers. Pitch, yaw, and roll control of the vehicle during the booster phase of flight and pitch and yaw control during the sustainer phase of flight are provided. The sustainer system also provides fluid power to the MA-5A control package for propellant flow control. Hydraulic capability loss is prevented by using RP-1 fuel, which can be directed into the hydraulic system.

A.2.1.5 Avionics—The avionics system consists of four flight-proven subsystems: flight control, telemetry, flight termination, and electrical power. All avionics packages are line-replaceable in the field.

Flight Control Hardware—Atlas IIAS flight control hardware consists of the Atlas servo inverter unit (SIU) and two rate gyro units (RGU). It is integrated with Centaur avionics. All aspects of guidance, flight control, and vehicle sequencing are managed by the FCS software within the Centaur inertial navigation unit (INU) (Fig. A.2.1.5-1). Attitude control is accomplished by a closed-loop hydraulic servo positioning system for the booster engine. The RCU provides alternating current power (synchronized by a Centaur reference signal) to the RGUs. Two RGUs, one forward and one aft, provide pitch

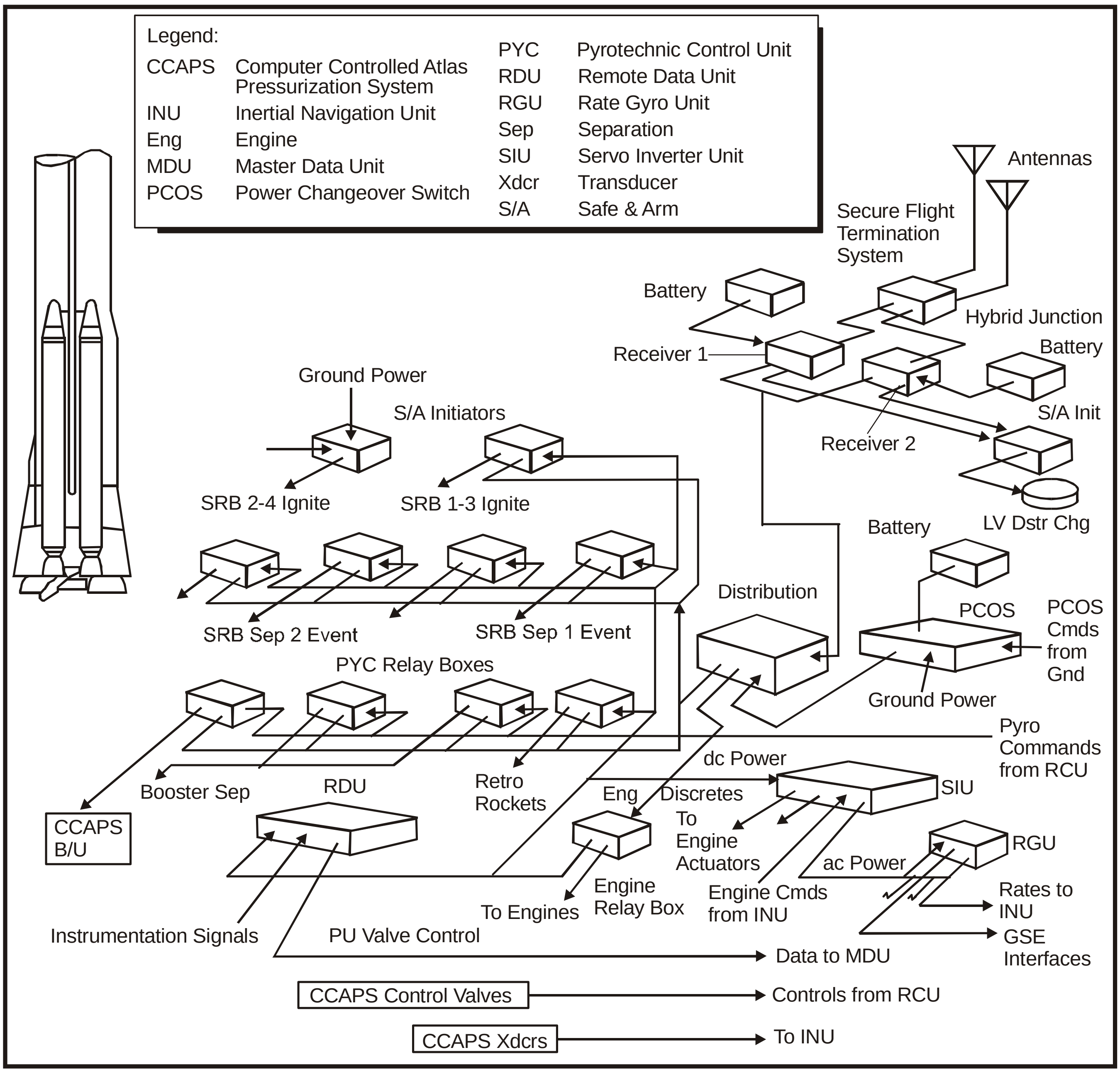


Figure A.2.1.5-1 Atlas IIAS Avionics for the Atlas Stage and yaw rate signals to the Centaur INU for vehicle stabilization. The RGUs provide true body rates to the INU. The inertial measurement subsystem provides vehicle roll rates and attitude for roll control. The flight control system provides the capability to ignite two SRBs during flight. This is accomplished with two initiation units and a pyrotechnic battery. The flight control subsystem (FCS) has the capability to provide two SRB jettison events. This is accomplished with four pyrotechnic control units (two for each separation event).

Telemetry Subsystem—The Atlas portion of the data acquisition system (DAS) consists of a remote data unit (RDU), vehicle transducers, and instrumentation harnessing. The RDU provides transducer excitation and signal conditioning for all Atlas measurements. They are digitally encoded and transmitted to the master data unit (MDU) located on the Centaur vehicle.

Secure Flight Termination System (FTS)—The Atlas stage FTS provides the capability for range safety termination of Atlas flight if there is a flight malfunction. The Atlas FTS is completely independent from controls of other vehicle systems and from the Centaur upper-stage FTS.

For Atlas IIAS, the FTS also includes a commanded SRB destruct capability. Figure A.2.1.5-2 illustrates the secure FTS for Atlas IIAS. Each SRB is equipped with an inadvertent separation destruct system (ISDS) to terminate thrust capability in the unlikely event of an unplanned SRB jettison. With unplanned jettison, lanyard pull switches initiate a destruct signal that fires a conical-shaped charge located on the forward bulkhead of the SRB. Thrust capability is negated within 1 second after premature separation from Atlas. The system is single-fault tolerant to destruct and single-fault tolerant

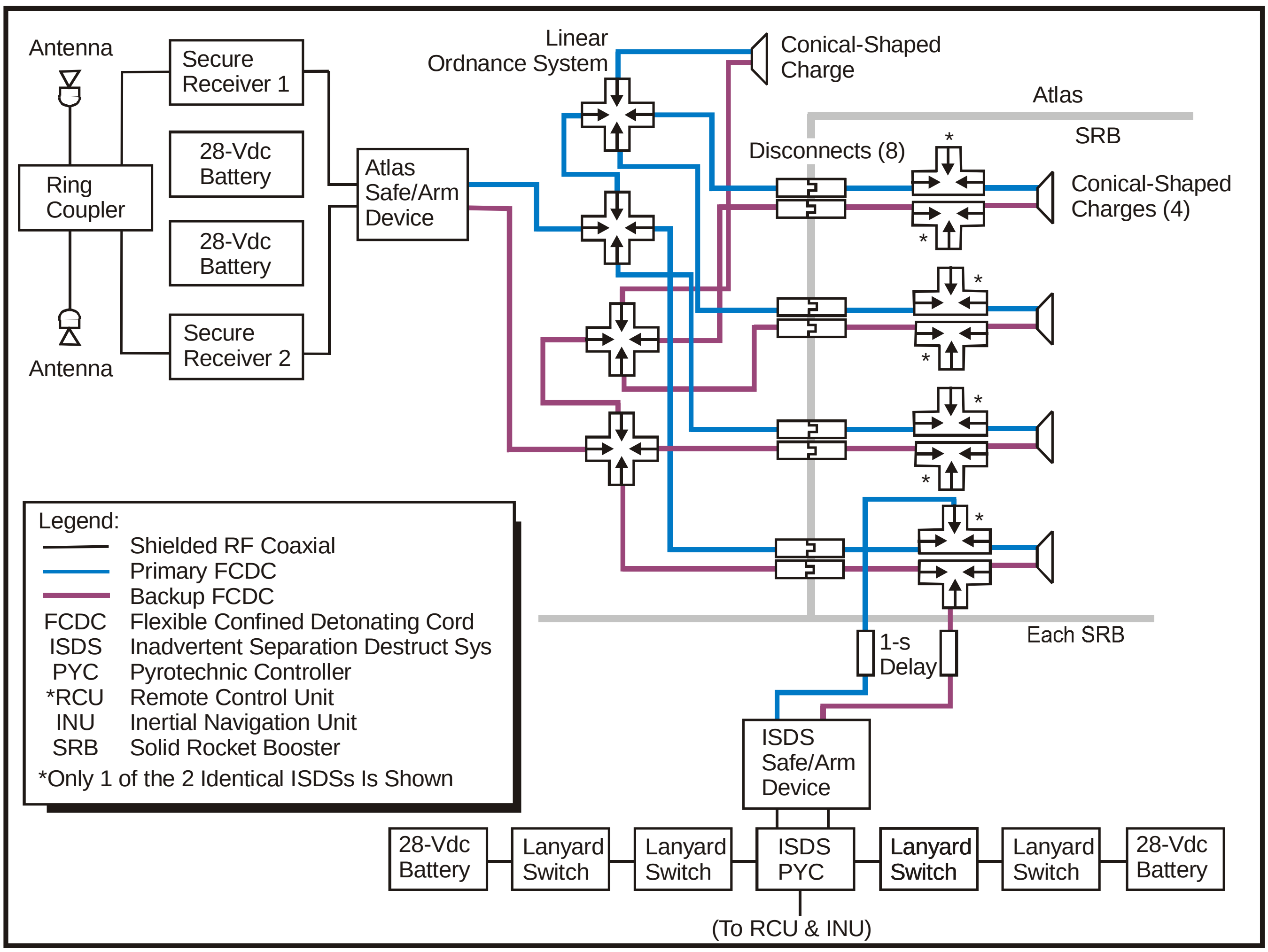


Figure A.2.1.5-2 The Atlas IIA/IIAS FTS is autonomous, secure, and approved for use at both ranges.

against inadvertent destruct. With normal Atlas IIAS flight, the system is safed before SRB jettison. Single-fault tolerance to safe and inadvertent safe is also included.

Electrical Power Subsystem—The electrical power system consists of a 28-Vdc main vehicle battery, and associated electrical harnesses. The Atlas pod contains electrical distribution (through the D-box), a power changeover switch (PCOS), and a single-point grounding system. The PCOS transfers from ground to internal power before liftoff. The main vehicle battery supplies power to three separate buses.

The pyrotechnic electrical power system for each SRB is separate from the main vehicle power system. The SRB separation pyros primary and backup controllers are powered from separate pyrotechnic batteries, making them completely redundant.

Propellant Utilization (PU)—The Atlas IIAS PU system measures and controls the fuel and oxidizer mixture ratio to the sustainer engine. The system is software-based and consists of differential pressure transducers, associated sense lines, flight software control logic, Atlas SIU-based servo loop circuitry, and a sustainer engine flow control, or PU valve. With the PU valve, sustainer engine propellant mixture ratio is controlled to uniformly deplete all usable tank propellants at sustainer engine cutoff (SECO). The PU system is used on all Atlas IIAS booster stages.

Atlas Propellant-Level Indicating System (PLIS)—The Atlas PLIS is used during tank filing operations to indicate the level of oxidizer in the propellant tanks. The system consists of hot-wire sensors near the top of the oxygen tank and associated hardware for ground control and display. The sensors are used for two primary reasons: (1) to indicate when a minimum propellant level has been reached to ensure sufficient loading to meet mission requirements, and (2) to indicate when a maximum level has been reached to ensure that sufficient ullage volume is present in the tanks for safe tank venting. Atlas fuel tank propellant levels are indicated using a sight gauge temporarily installed on the vehicle during fuel tanking operations.

Atlas Propellant Depletion System—The Atlas IIAS (MA-5A) propellant depletion system is used to sense the onset of propellant outage and to initiate SECO. The system consists of two types of sensors: (1) redundant pressure switches in the sustainer engine that close when LO_2 manifold pressure drops, indicating LO_2 depletion, and (2) redundant optical sensors in the sustainer thrust cone that sense the absence of fuel as the fuel level passes below the optical sensors. The output of these sensors is sent to the Centaur INU. When either fuel or oxidizer depletion is indicated, the INU immediately commands SECO and begins the post-SECO sequence. The primary purpose of the pressure switches is to indicate to the flight computer as quickly as possible that the engine is running out of oxidizer so the INU can command subsequent events, thereby maximizing vehicle performance. The primary purpose of the optical fuel depletion sensors is to prevent the engine from operating on only oxidizer.

A.2.2 Atlas III Major Characteristics

The Atlas III operates as a single stage booster with no SRBs. This major simplification was achieved by eliminating the jettisonable booster package, stretching the stage length by 4.4 m (14.5 ft), and upgrading the propulsion system to a single, throttleable Pratt & Whitney/NPO Energomash RD-180 engine. The result is higher performance, fewer parts, higher reliability, and lower cost.

A.2.2.1 Structure

Sustainer Section—The pressure-stabilized Atlas III tank is functionally similar to the Atlas IIAS tank, but includes a 3.05-m (10-ft) LO_2 tank stretch, uprated skin gauges, and an elliptical aft fuel bulkhead. The Atlas III uses 12 retrorockets of the Atlas IIAS design to accommodate the higher booster staging mass.

Thrust Section—The Atlas III thrust structure (Figure A.2.2.1-1) has a similar integrally machined panel design as the Atlas IIAS, but with all provisions for booster package jettison and SRB attachment eliminated, including the separation latches, jettison tracks, and the SRB attach fittings. The thrust structure is upgraded to support the new engine truss attachment interface and to enclose the RD-180 engine within a cylindrical aluminum engine fairing with two external bottle fairings that contain six of 13 helium bottles used for tank pressurization.

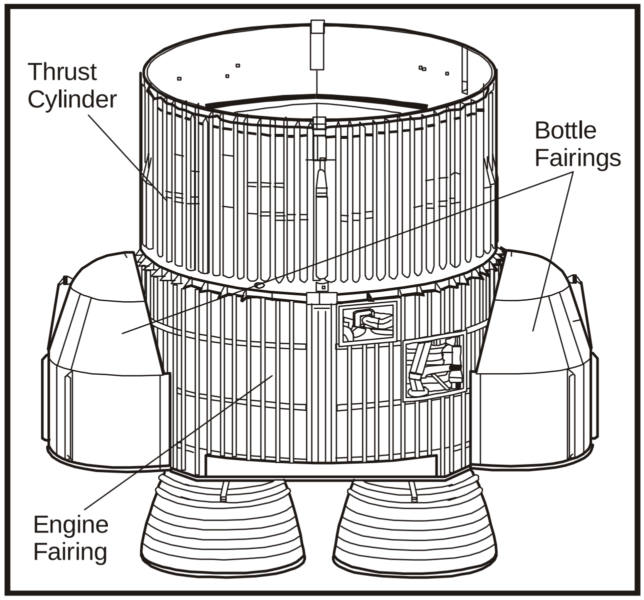


Figure A.2.2.1-1 IIIA Thrust Structure

Interstage Adapter—The Atlas III ISA is 457-mm (18-in.) longer to accommodate the longer single-engine Centaur installation, but retains the same Atlas and Centaur structural interfaces. This ISA uses aluminum-lithium alloy skin and stringers for reduced weight compared to the aluminum IIAS design. The ARCM installation in the ISA was eliminated (Sect. A.2.2.4).

A.2.2.2 Pneumatics

Helium Supply System—The Atlas III system carries 13 bottles of the same design as Atlas IIAS through the entire Atlas flight. The bottle arrangement has been configured with seven bottles located in a ring in the base of the thrust section and three bottles in each of the two bottle fairings.

Engine Control System—The RD-180 engine on the Atlas III carries an engine controls bottle similar to the Atlas IIAS design integral with the engine.

A.2.2.3 Propulsion

The Atlas III uses a single, high performance, throttleable RD-180 engine system and a simplified ground start system.

RD-180 Engine System—The RD-180 engine, provided by RD AMROSS, a joint venture of Pratt & Whitney and NPO Energomash for use on the Atlas III, is a two-chamber design fed by a common turbopump assembly. The engine is a total propulsive unit with an integral start system and hydraulics for control valve actuation and thrust vector gimbaling, pneumatics for valve actuation, and thrust frame to distribute loads, all self-contained as part of the engine. Nominal thrust at sea level is 3826 kN (860,200 lb_f). The RD-180 operates on a staged combustion cycle using the same LO_2 oxidizer and RP-1 fuel as the MA-5A and is capable of continuous throttle between 47% and 100% of nominal thrust, allowing for substantial control over launch vehicle and payload environments. Programmed thrust profiles can throttle the engine back to minimize vehicle loads during the peak transonic load and high dynamic pressure periods of flight while otherwise maximizing performance at higher sustained acceleration. The two-chamber RD-180 (Fig. A.2.2.3-1) is a derivative of the four-chamber RD-170/171 engines used on the Russian Zenit (more than 25 flights) and Energia (two flights) boosters. Initial test firings of the RD-180 occurred in July 1997 and the engine was flight proven on the May 2000 inaugural mission of Atlas III.

SRB and ADJ—Since no SRBs are used, the ADJ system is eliminated.

Ground Start System—The RD-180 engine carries an integral start pressurization bottle and hypergol ampoules. The ground system provides hydraulic pressure to operate the start valve.

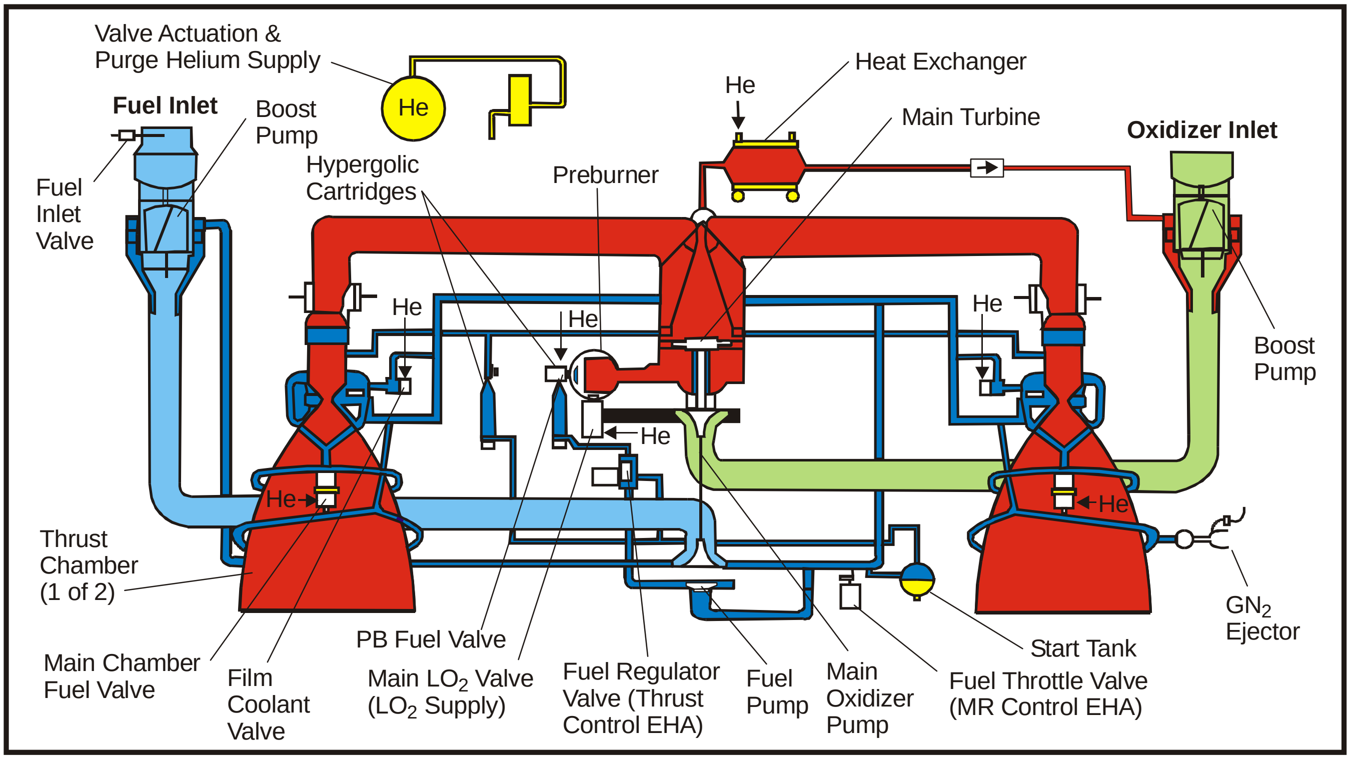


Figure A.2.2.3-1 RD-180 Engine Schematic

A.2.2.4 Hydraulic Thrust Vector Control

An integral TVC system on the dual chamber RD-180 is powered by high-pressure RP-1 tapped from the main fuel pump. This eliminates the need for a dedicated hydraulic pump. Two servoactuators on each thrust chamber provide pitch, yaw, and roll control throughout booster flight. This eliminates the separate ARCM (Sect. A.2.1.3) that is required for roll control of earlier Atlas configurations during sustainer solo flight, where the single-chamber sustainer engine is inherently incapable of integral roll control.

A.2.2.5 Avionics

Flight Control Hardware—Atlas III flight control hardware consists of the newly added Atlas remote control unit (ARCU) and two RGUs. The ARCU is similar in concept to the Centaur remote control unit (CRCU) and is under INU control through the 1553 data bus. It provides sequencing, switching, and power distribution functions for the Atlas stage. It is integrated with Centaur avionics. One major change on the Centaur upper stage is in the TVC for the single RL10 engine. One ECU (controlled over the 1553 data bus) and two EMAs replace the DEC hydraulic actuators and provide redundancy for engine TVC. All aspects of guidance, flight control, and vehicle sequencing are managed by the FCS software within the Centaur INU (Fig. A.2.2.5-1). Atlas attitude control is accomplished by a hydraulic servo positioning system for the booster engine. The CRCU provides alternating current power (synchronized by a Centaur reference signal) to the RGUs. Two RGUs, one forward and one aft, provide pitch and yaw rate signals to the Centaur INU for vehicle stabilization. The RGUs provide true body rates to the INU. The inertial measurement subsystem provides vehicle roll rates and attitude for vehicle control.

Telemetry Subsystem—The Atlas III telemetry subsystem is similar to the Atlas IIAS subsystem. Only minor changes have been incorporated into the MDU and RDU designs.

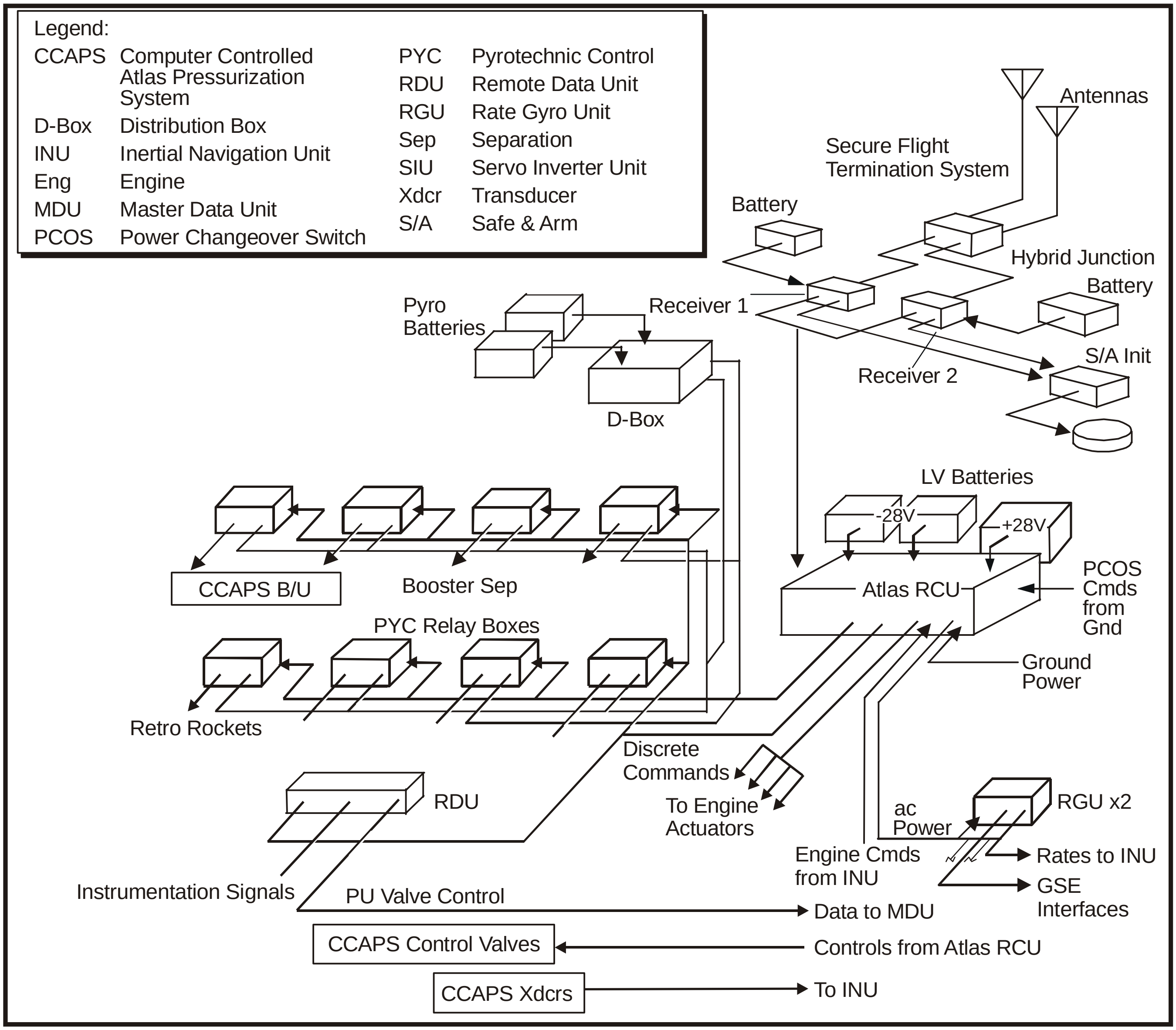


Figure A.2.2.5-1 The Atlas RCU combines several functions on the Atlas IIIA and IIIB avionics suite.

Secure Flight Termination—Atlas III uses the Atlas IIAS system without the SRB ISDS or SRB conical-shaped charge components. Engine shutdown is handled by the Centaur SFTS and the INU for Atlas IIIA/IIIB (Fig. A.2.2.5-2).

Electrical Power System—The electrical power system consists of a 28 Vdc main vehicle battery, two smaller batteries used for the-28 Vdc Atlas RCU (for RD180 control), and associated electrical harnesses. The Atlas RCU contains electrical distribution, a power changeover switch (PCOS), and a single-point grounding system. The PCOS transfers from ground to internal power before liftoff. The main vehicle battery supplies power to three separate buses. The other two batteries supply -28 Vdc power redundantly to the one -28 Vdc bus.

The pyrotechnic electrical power system is separate from the main vehicle power system. Two pyrotechnic batteries supply power to the pyrocontrollers through the D-box.

Propellant Utilization—The Atlas III uses a similar PU sensing system to Atlas IIAS, but with mixture ratio adjustments sent to the RD-180 engine from the INU through the Atlas RCU.

Atlas Propellant Depletion System—The Atlas III uses redundant RP and LO_2 depletion sensors to trigger a commanded BECO with depletion of either fuel or LO_2. Fuel depletion sensors are the same redundant optical sensors used on Atlas IIAS. Pressure sensors are used to detect LO_2 depletion on the Atlas III.

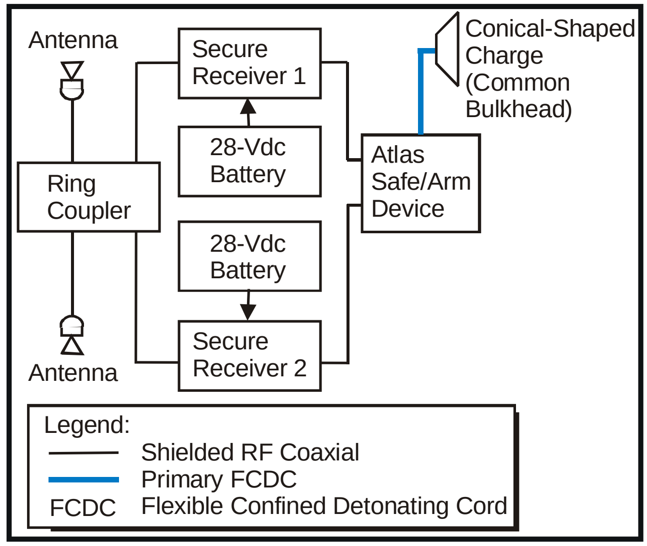


Figure A.2.2.5-2 The Atlas IIIA and IIIB flight termination system (FTS) uses a simplified version of the Atlas IIAS FTS.

A.2.3 Atlas V Major Characteristics

Overview—The Atlas V launch vehicle system is based on the newly developed, structurally stable 3.8-m (12.5-ft) diameter Common Core Booster™ (CCB) powered by a single RD-180 engine. To meet the evolving needs of payload users, the Atlas V 400 and 500 series can tailor performance by incorporating SRBs. The mission design capabilities for Atlas V are significantly enhanced relative to current Atlas and Titan vehicles. Atlas V more than doubles the capability for GTO customers, from 4,037 kg (8,900 lb) to GTO for an Atlas III to greater than 8,618 kg (19,000 lb) to GTO for the Atlas V 500 series. The capabilities available for heritage vehicles have been maintained and enhanced in the Atlas V design. One example is optimized in-flight retargeting based on actual booster performance. Another is use of multiple, Centaur upper stage burns with extended coast times of over 5 hours for GSO three-burn missions.

Commonality—The cornerstone of the Atlas V system design is the use of common system elements. The same or very similar hardware elements are used across the Atlas V 400 series and 500 series. The CCB used for the 400 and 500 series is identical, up to and including the solid rocket motor (SRM) attach fittings and harnessing to fly the SRMs as are the avionics components and the Centaur upper stage. The three different lengths of 5-m payload fairing used for 500 series vehicles are all common, with the exception of a kittable payload section to provide the differing lengths. The same is true for the two available lengths (LPF and EPF) of the 4-m fairing used for Atlas 400 series vehicles and for the stretched EPF (XEPF) under development (Sect. 4.1.1.1).

Reliability—There are two fundamental ways in which reliability is infused into the Atlas V system design: fault avoidance and fault tolerance. The essence of fault avoidance is creating a simpler design where fewer opportunities for failure exist, or through greater margins—moving away from the edge of the system's capability. An example of fault avoidance is reduction in the number of engines. Fault tolerance, on the other hand, is accomplished through redundancy, or by implementing processes like prelaunch checkout of the RD-180 engine. This capability allows verification that the engine has properly started and is up to 60% power before the commitment to launch. If a problem is detected, the engine can be shut down and the problem corrected.

Extreme care is taken to reduce the probability of failure through qualification and acceptance testing and process controls for systems such as engines that are susceptible to single point failures (SPF) and where redundancy is cost prohibitive. The booster and upper stage liquid engines are qualified with substantial margin. Margins in thrust level, run duration, and inlet conditions are demonstrated. Then, each flight engine will be acceptance tested to full flight duration.

One measure of the success of the Atlas V reliability design process is the reduced number of SPF susceptabilities inherent in the system. Once structural SPFs are subtracted, active SPFs have been reduced approximately 70% to 30 for the Atlas V compared to the Atlas IIAS.

The Atlas V family of launch vehicles can be launched from CCAFS Launch Complex 41 (LC-41) to support various mission needs. The Atlas V 400 and 500 series are under development for initial launch capability in 2002 from the Eastern Range. The Atlas V 400 series is shown in expanded view in Figure A.2.3-1. The Atlas V 500 series is shown in Figure A.2.3-2. The components that comprise the Atlas V system are described in the following paragraphs.

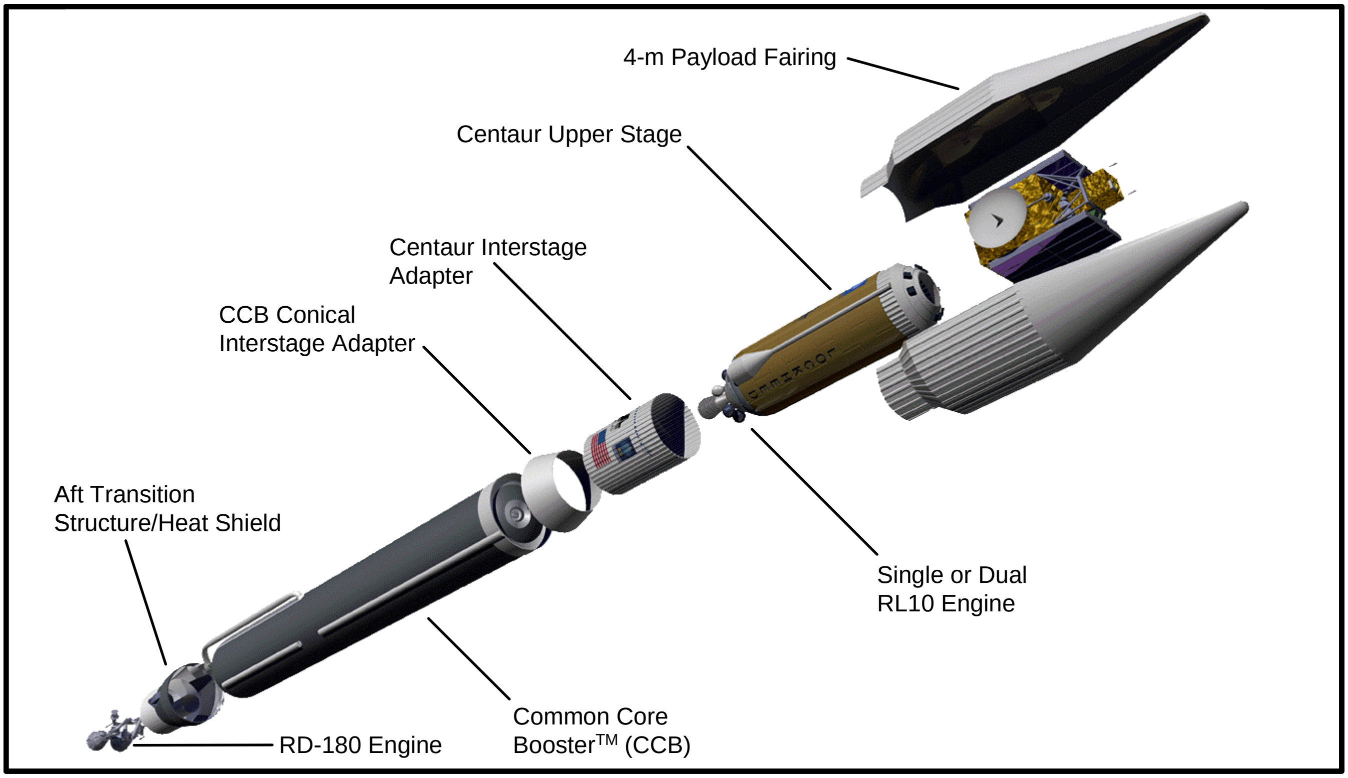


Figure A.2.3-1 Atlas V 400 Series Launch Vehicle

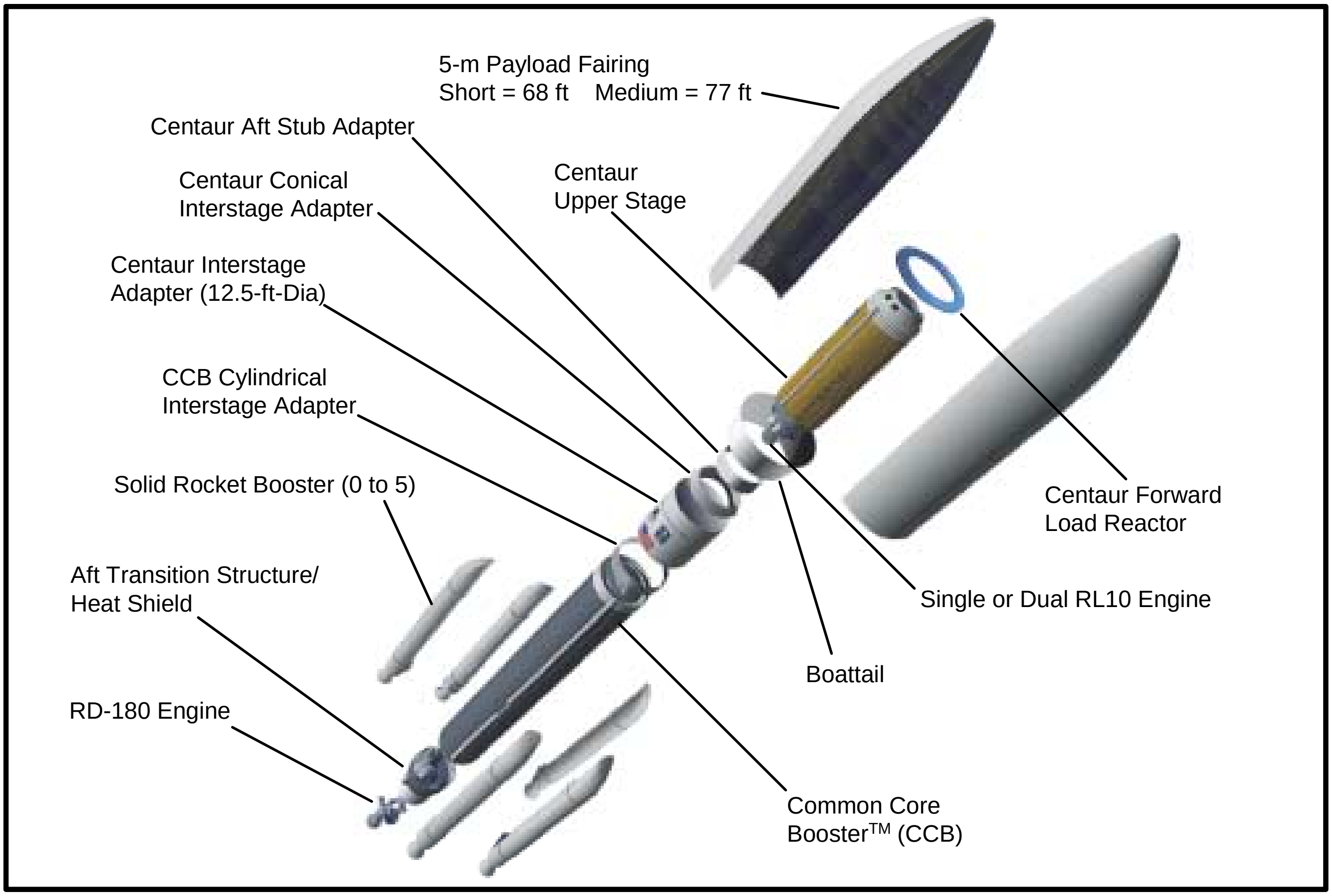


Figure A.2.3-2 Atlas V 500 Series Launch Vehicle

A.2.3.1 Common Core Booster™

The CCB provides the main propulsive stage for the Atlas V family. It is powered by a single RD-180 engine fueled by LO_2/RP propellants. The CCB is fully common and interchangeable between the Atlas V 400 and 500 series, and can accommodate up to five strap-on SRBs.

The CCB, 3.8-m (12.5-ft) in diameter and 33-m (109-ft) long, is constructed of structurally stable aluminum isogrid tanks. These tanks are comprised of only 16 parts, while Atlas IIAS uses over 100 parts, another example of simplified design.

Avionics components used during CCB flight are located in a pod structure, which runs along the side of the tank.

An expanded view of the CCB is shown in Figure A.2.3.1-1, and a description of CCB components is included in the following paragraphs.

CCB LO_2 Tank—The structurally stable, welded aluminum CCB LO_2 tank is composed of a forward and aft barrel section joined by a splice ring, a one-piece spun forward dome with manhole cover, and a one-piece spun aft dome with a sump. Each barrel section is made of four isogrid aluminum panels welded together. The splice ring is an aluminum machined roll ring forging. A forward adapter skirt joins the LO_2 tank to the CCB interstage adapter (conical for the 400 series and cylindrical for the 500 series). The adapter is aluminum 2014 isogrid panels with aluminum ring frames. The LO_2 intertank skirt attaches to aft end of the tank and joins the LO_2 tank to the RP tank. The skirt is aluminum isogrid with aluminum ring frames.

CCB RP Tank—The structurally stable, welded aluminum CCB RP tank is composed of a single barrel, a one-piece spun forward dome with a manhole cover, and a one-piece spun aft dome with a sump. The barrel section is made of four isogrid aluminum panels welded together. The RP intertank

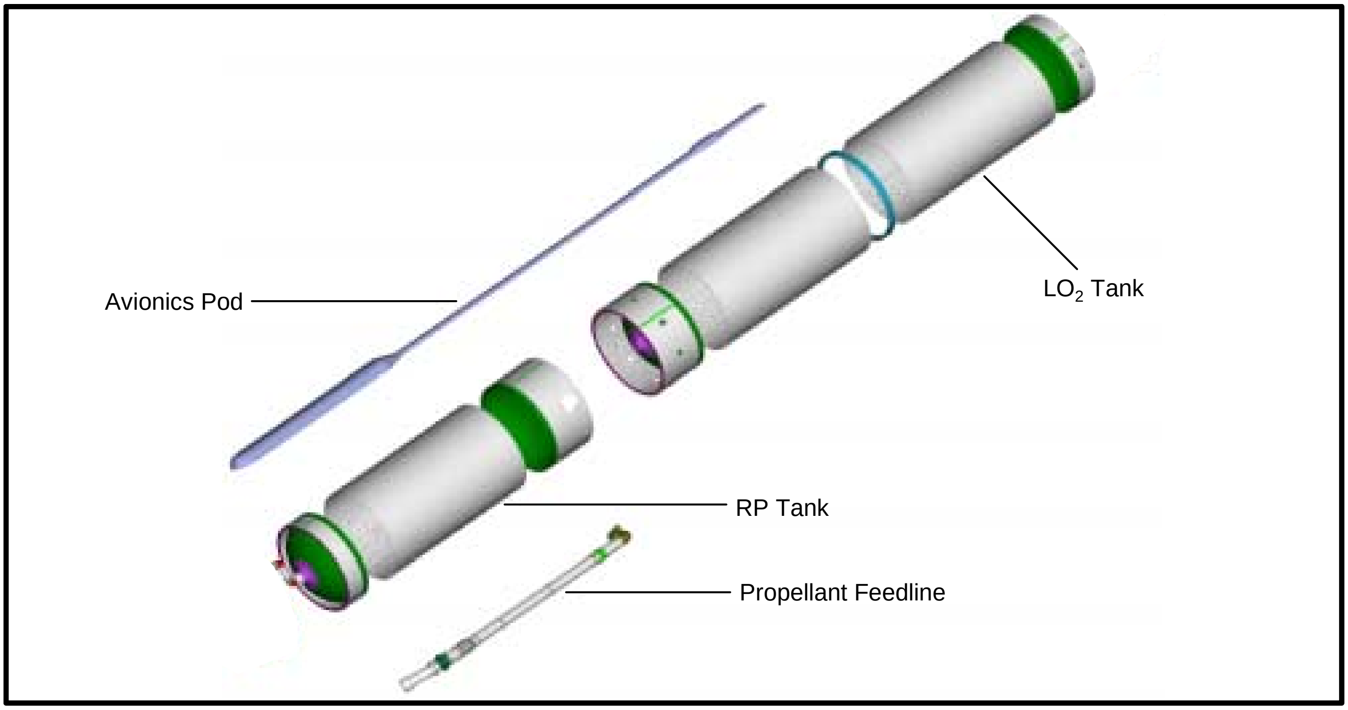


Figure A.2.3.1-1 Common Core Booster™ Components

skirt attaches to the forward end of the tank and joins the RP tank to the LO_2 tank. The adapter is aluminum isogrid with aluminum ring frames. The RP adapter skirt joins the aft end of the tank to the aft transition structure. The skirt is aluminum 2014 isogrid panels with aluminum ring frames.

Propellant Feedlines—The CCB propellant feedlines are aluminum tube assemblies that deliver LO_2 and RP propellants from the tanks to the RD-180 engine. The LO_2 feedline runs from the bottom of the LO_2 tank, along the outside of the RP tank to the aft transition structure (ATS) and the engine. The RP feedline runs from the bottom of the RP tank to the aft transition structure and the engine.

Avionics Pod—The electronics that control CCB flight are housed in the avionics pod. The main equipment pod is located along the outside of the RP tank and provides mounting and protection for the booster remote control unit (BRCU), batteries, ordnance remote control units (ORCA), instrumentation system, and flight termination system. The upper part of the avionics pod provides a raceway for the electrical harnessing and mounting and protection for the rate gyro unit.

A.2.3.2 Aft Transition Structure and Heat Shield—The RD-180 engine is attached to the CCB through the ATS and heat shield. The ATS provides the structural load path for the engine thrust loads and houses the helium bottles used to the pressurize the propellant tanks. The heat shield provides thermal protection to the CCB during engine firing and provides an environmental closure for the engine compartment. The ATS is constructed of aluminum 7075 ring frames and fittings with aluminum 2040 skins. The ATS is built from integrally machined aluminum components, harnesses, tubing, and helium bottles. The heat shield is a composite structure with graphite epoxy facesheets over aluminum honeycomb, and aluminum ring frames.

A.2.3.3 RD-180 Engine—The Atlas V uses the same RD-180 engine flown on Atlas III. The RD-180 is certified for Atlas V 400 and 500 configurations with more than 24,200 seconds in 138 tests on 32 engine builds. The Atlas V uses fewer propulsion systems and staging events than heritage Atlas or Titan vehicles. An Atlas IIAS can place 3719 kg (8200 lb) into GTO but requires nine propulsion subsystems and seven staging events. The Atlas V 401 configuration delivers 4944 kg (10,900 lb) to GTO with only two propulsion subsystems and three staging events. The Atlas V pressurization system uses only five helium bottles, kept at ambient temperature while the Atlas II pressurization system uses 13 cryogenic helium bottles.

A.2.3.4 Solid Rocket Booster—The SRB is 1.5-m (5-ft) in diameter, 20.4-m (67-ft) long, and weighs approximately 46,000 kg (51 tons). Each SRB is identical, interchangeable, and is designed for reliability. The SRBs are all ground-lit (no air ignition as used on Atlas IIAS), have a fixed nozzle that is canted at 3° (no TVC needed), and are monolithic in design (no segment joints). The SRBs use simplified, proven elements adapted from operational U.S. Government systems, including a graphite-epoxy case, carbon phenolic nozzle, and high-performance Class 1.3 HTPB propellant. The SRBs are manufactured by Aerojet and shipped by truck to the launch site in a ready-to-fly configuration where they are installed with no site processing on the launch vehicle in the Vertical Integration Facility (VIF).

A.2.3.5 CCB Conical Interstage Adapter—The CCB conical interstage adapter provides the diameter change from the 3.8-m (12.5-ft) diameter CCB to the 3.05-m (10-ft) diameter Centaur. The adapter is a graphite-epoxy composite structure with aluminum rings. The joint between the CCB conical interstage adapter and the Centaur interstage adapter is assembled at the launch site in the VIF. The CCB conical interstage adapter is used on the Atlas V 400 series.

A.2.3.6 CCB Cylindrical Interstage Adapter (ISA)—The CCB cylindrical ISA provides the structural attachment between the CCB and the Centaur interstage adapter. After final assembly, the CCB ISA provides the second of two vehicle attach points via stiff link to the mobile launch platform (MLP). It is constructed from an aluminum rolled-ring forging. The CCB cylindrical ISA is used on the Atlas V 500 series.

A.2.3.7 Centaur Forward Load Reactor (CFLR)—The CFLR is attached to the Centaur forward adapter and provides load sharing between the Centaur structure and the 5-m PLF. The CFLR is installed between the Centaur and the 5-m PLF base module in the VIF before spacecraft mate. The CFLR also accommodates a work access platform inside the PLF while the launch vehicle is in the VIF. The CFLR is used on the Atlas V 500 series.

A.2.3.8 Centaur Conical Interstage Adapter—The Centaur conical interstage adapter provides the transition from the 3.05-m (10-ft) diameter Centaur aft stub adapter to the 3.8-m (12.5-ft) diameter Centaur interstage adapter. It is an aluminum skin stringer structure. The Centaur conical interstage adapter is bolted to the Centaur aft stub adapter, and this subassembly is then installed inside the C-ISA. The Centaur conical interstage adapter is used on the Atlas V 500 series.

A.2.3.9 Centaur Interstage Adapter (C-ISA)—The 3.8-m (12.5-ft) diameter Centaur interstage adapter provides the attachment between the CCB, the Centaur, and the 5-m PLF boattail. The Centaur engine nozzles are extended inside the adapter. The adapter is a composite structure with graphic-epoxy facesheets over an aluminum honeycomb core. The 3.8-m (12.5-ft) diameter Centaur interstage adapter is used on the Atlas V 500 series.

A.2.3.10 Centaur Interstage Adapter (Short)—The Centaur interstage adapter (C-ISA short) attaches the Centaur stage to the CCB conical interstage adapter. The construction is aluminum-lithium skin stringer and frame. The Centaur separates from the top of the adapter following completion of the CCB engine burn. The Centaur interstage adapter is used on the Atlas V 400 series.

A.2.3.11 Centaur Aft Stub Adapter—The Centaur aft stub adapter is an aluminum structure that provides the structural attachment to the Centaur stage. The aft end of the adapter is bolted to the Centaur conical interstage adapter. The Centaur aft stub adapter is used on the Atlas V 500 series.

A.2.3.12 5-Meter Diameter Payload Fairing Boattail Assembly—The 5-m PLF boattail assembly provides the transition structure between the 3.8-m (12.5-ft) diameter Centaur interstage adapter and the 5-m diameter PLF modules. It is a composite structure of graphite-epoxy facesheets over an aluminum honeycomb core. The 5-m PLF boattail assembly is used on the Atlas V 500 series.

A.2.3.13 Centaur Forward Adapter (CFA)—For Atlas IIIB and V, the stub adapter and the equipment module have been combined into a single, more producible structure called the CFA.

A.2.3.14 Avionics—The following avionics improvements have been incorporated between Atlas III and Atlas V:

The Centaur avionics are maintained (INU/CRCU/ECU/MDU/RDU) from Atlas III except PYCs have been replaced by ORCAs for all pyrotechnic events. The ORCA is a 30 channel, sequenceable output, three inhibit pyrocontroller which is under INU control through the 1553 data bus. Two ORCAs and two dedicated pyrobatteries are provided in a block redundant design. Also, the MVB has been upgraded for capacity.

On the booster, the ARCU has been replaced by the BRCU and the D-Box eliminated along with all wiring inputs from the booster to the INU. Redundant main batteries have been incorporated in the design, while the -28-Vdc batteries are no longer required due to electrical circuit redesign in the BRCU. INU inputs are now routed to the BRCU and sent to the INU over the 1553 data bus. Identical to Centaur, the PYCs have been replaced by ORCAs for all pyrotechnic events. Two ORCAs and two dedicated pyrobatteries are provided in a block redundant design. The Atlas SFTS has been replaced by the Automatic Destruct System (ADS), a system which senses inadvertent stage separation and autonomously commands destruct (the existing Centaur capability is maintained).

A significant example of added redundancy is the new fault-tolerant inertial navigation unit (FTINU) that is under development by Honeywell for service on Atlas V in 2003. This component provides enhanced mission accuracy and redundancy through a fully fault-tolerant pentad of ring laser gyros. The FTINU also houses and executes the flight software to guide and control the vehicle through its mission.

The basic functions of the Atlas V avionics system (compared to Atlas III) are unchanged. Many subsystems are identical in design from Atlas III (e.g., PU, EMA TVC, INU, etc). Evolution of the Atlas vehicle avionics system is depicted in Figure A.2.3.14-1.

A.2.4 Centaur Upper Stage Major Characteristics

Centaur IIA/IIAS—The Centaur vehicle used for Atlas IIA and IIAS is 3.05 m (10 ft) in diameter and 10.0-m (33-ft) long. It uses LH_2 and LO_2 propellants. The propulsion system uses two regeneratively cooled and turbopump-fed RL10A-4 or RL10A-4-1 engines, manufactured by Pratt & Whitney. Centaur avionics packages mounted on the forward equipment module control and monitor all vehicle functions. Variation between configurations are only in the propulsion system options and the avionics harnessing that connects the Centaur flight computer to the Atlas stage.

Centaur III—The Centaur IIIA SEC tank design is the same as the other Centaur vehicles and uses only one RL10A-4-1 engine. The overall length of the SEC is 10.5 m (34.3 ft). The additional length is due to the relocation of the single RL10A-4-1 engine onto an engine support beam spanning the centerline of the vehicle. The Centaur IIIB SEC/DEC has an extended tank that is 11.74-m (38.52-ft long). Both the SEC and DEC use RL10A-4-2 engines. The SEC also incorporates electromechanical actuators to perform thrust vector control, in place of the hydraulic actuation system. Avionics harnessing has been updated to accommodate these vehicle changes.

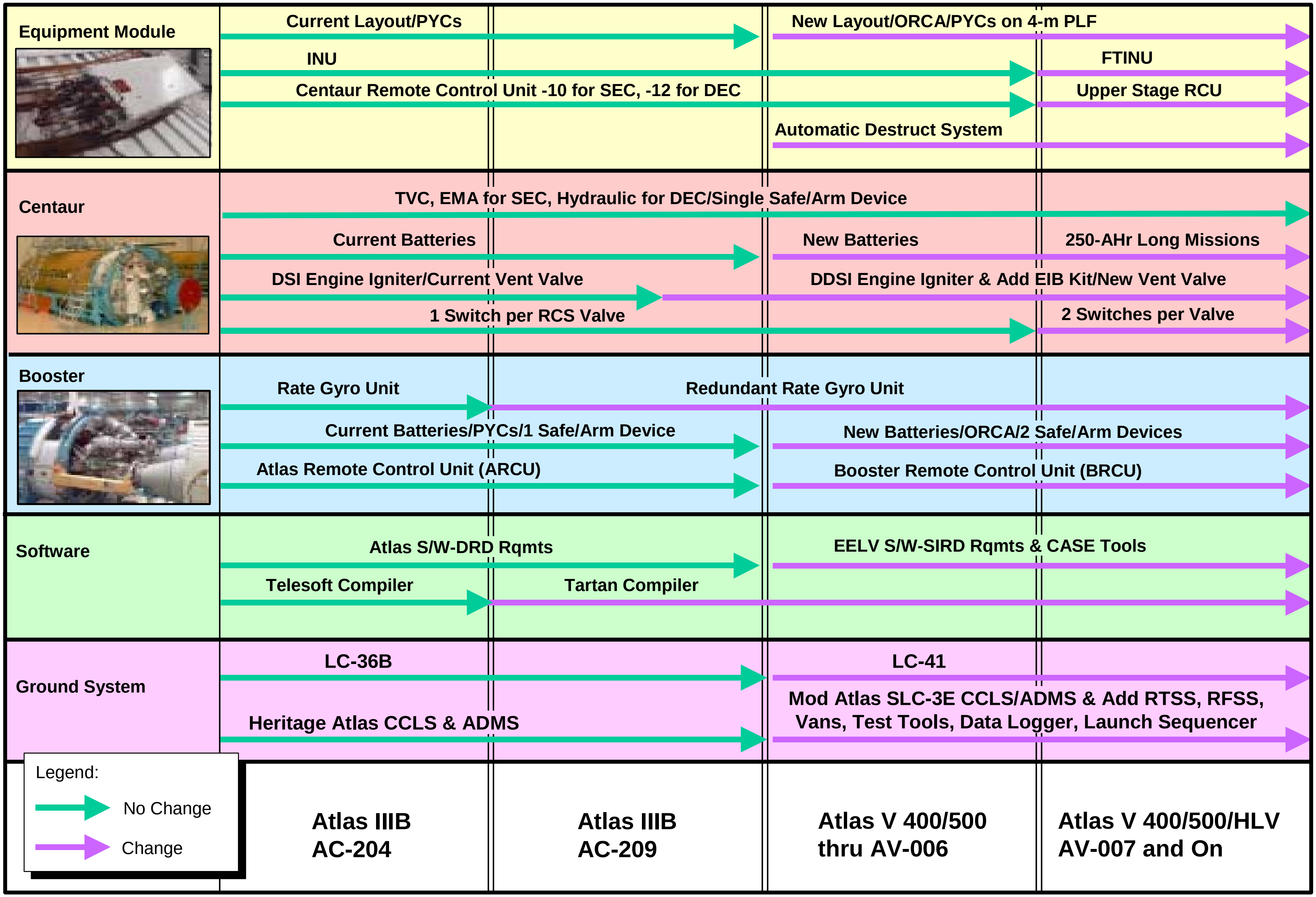


Figure A.2.3.14-1 Electrical System Evolution

Common Centaur—All Atlas V 400 and 500 configurations use a stretched version of the flight-proven Centaur upper stage used for Atlas III (CIII), which can be configured as a SEC or a DEC depending on mission needs. The common Centaur is flown on Atlas IIIB and Atlas V and has a 1.7-m (5.5-ft) stretch in tank length for added performance compared to the Atlas IIAS Centaur. The Common Centaur uses the RL10A-4-2 engine, and the nozzles are locked into the extended position before flight.

A.2.4.1 Structure—The Centaur structural system consists of three major structural elements: the propellant tank, Centaur forward adapter (CFA), and tank insulation.

Propellant Tank—The propellant tank structure provides primary structural integrity for the Centaur vehicle and support for all upper-stage airborne systems and components (Fig. A.2.4.1-1). A double-wall, vacuum-insulated intermediate bulkhead separates the propellants. The tanks are constructed of thin-wall fully monocoque corrosion resistant steel. Tank stabilization is maintained by internal pressurization.

Figure A.2.4.1-1 The monocoque Centaur tank is structurally efficient.

In dual-engine (IIAS, IIIB, and V) Centaur configurations, engines are mounted directly to the propellant tank aft bulkhead. The single-engine (IIIA, IIIB, and V) Centaur incorporates an engine support beam that attaches to the aft bulkhead in the existing engine mount locations, but provides centerline mounting of the single engine 452-mm (17.5-in.) aft of the dual-engine location. Redesigned thrust vector control actuator supports have also been incorporated.

Centaur Forward Adapter (CFA)—The CFA combines the functions of the stub adapter and equipment module used on earlier Atlas configurations into a single, more producible structure that provides mounting for avionics packages, electrical harnesses, and the forward umbilical panel (Fig. A.2.4.1-2 and A.2.4.1-3). The spacecraft adapter bolts to the forward ring of the CFA. The load carrying capability of the CFA has been enhanced so it can react the loads of a 5,900- to 9,072-kg (13,000- to 20,000-lb) payload. Localized structure improvements and use of aluminum-lithium stringers provide a cost and mass-efficient structural enhancement without requiring requalification of the avionics hardware mounted on the CFA.

Figure A.2.4.1-2 The Centaur forward adapter provides a robust, stable interface for launch vehicle avionics and the attached payload adapter and spacecraft.

Figure A.2.4.1-3 The Centaur forward adapter allows line replacement of all avionics in the field.

Tank Insulation—Centaur tank insulation minimizes ice formation and LO_2 and LH_2 boiloff on the ground and during atmospheric ascent. It consists of closed-cell polyvinyl chloride (PVC) foam panels adhesively bonded to the exterior surface of the LH_2 and LO_2 tanks (Fig. A.2.4.1-4). The fixed foam replaces jettisonable insulation panels used for Atlas I.

Figure A.2.4.1-4 Fixed foam bonded directly to the Centaur tank is a simple and efficient insulator.

A.2.4.2 Pneumatics—The pneumatic system controls tank pressure, provides reaction control and engine control bottle pressure, and provides purges. The pneumatic system consists of the computer-controlled vent and pressurization system (CCVAPS), helium supply system, and purge supply system.

CCVAPS—The CCVAPS maintains the ullage tank pressures to ensure structural integrity. It prevents tank under pressure or overpressure, maintains the structural integrity of the intermediate bulkhead, and ensures positive suction pressure to the Centaur engines. The CCVAPS is a software-based fault-tolerant, closed-loop tank pressure control system. Three redundant ullage transducers in each tank measure pressure for the INU. The INU commands either pressurization or venting to achieve the desired tank pressure profile. Pressurization control is affected by locking the LO_2 and LH_2 vent valves and cycling the fault-tolerant pressurization solenoid valves. Vent control is active during boost phase and Centaur coast phases by cycling the LH_2 solenoid vent valve or unlocking the LH_2 or LO_2 self-regulating vent valves.

Helium Supply System—The baseline configuration for short-duration coast missions (<25 minutes) requires two 66.0-cm (26-in.) composite helium storage spheres charged to 27,580 kPa (4,000 psi). The spheres are graphite-overwrapped and reinforced with a 301 CRES metallic liner that provides leak-before-burst capability. A pressure regulator provides 3,450-kPa (500-psig) helium for engine and reaction control system (RCS) controls. The two helium storage bottles provide ample helium for launch of direct ascent and first descending node geosynchronous transfer trajectory missions. This complement of bottles is used on all Atlas geosynchronous transfer orbit (GTO) class missions. For longer coast duration missions, provisions exist for addition of two more helium bottles as required.

Purge Supply System—This dedicated airborne system provides helium purge gas to critical components to prevent air and moisture injection and freezing through the Atlas boost phase.

A.2.4.3 Propulsion—The Centaur propulsion system uses LH_2 and LO_2 propellants. Primary propulsion is provided by two Pratt & Whitney RL10A-4 or RL10A-4-1 engines. The single-engine Centaur uses the single RL10A-4-1 engine with an extendible nozzle.

Main Engines—RL10 engines are gimbaled, turbopump-fed, and regeneratively cooled and consist of a fixed primary nozzle and an optional secondary extendible nozzle (Fig. A.2.4.3-1). The 51-cm (20-in.) long columbium extendible nozzle provides enhanced engine performance through an increase in nozzle expansion ratio. Atlas V uses the extended nozzle with the extension locked in place before flight. Engine prestart and start functions are supported by supplying pressure-fed propellants to the engine pumps (Fig. A.2.4.3-2 for the IIAS/IIIB DEC, Fig. A.2.4.3-3 for the IIIA/IIIB SEC). The engines provide pitch, yaw, and roll control during powered phases of flight. The RL10A-4 engine develops a thrust of 91.19 kN (20,500 lb_f) at a specific impulse of 442.5 seconds. The RL10A-4-1 engine variant develops a thrust of 97.86 kN (22,000 lb_f) at a specific impulse of 444.0 seconds. Extendible nozzles increase engine thrust by 1.4 kN (300 lb_f) per engine and specific impulse by 6.5 seconds. Both the RL10A-4 and RL10A-4-1 engines are fully qualified and flight-proven.

Figure A.2.4.3-1 The RL10A-4 engine family can use an extendible nozzle for enhanced performance.

Reaction Control System—The hydrazine-based RCS provides pitch, yaw, and roll control for Centaur during coast phases of flight and provides propellant settling during the coast up through main engine start (MES). The single-engine Centaur also uses the RCS during powered phases of flight to perform roll control. The RCS is located on the Centaur aft bulkhead providing fault-tolerant control and avoiding contamination of the spacecraft when used.

Hydraulic TVC System—The dual-engine Centaur hydraulic system consists of one hydraulic power unit with two hydraulic servo actuators for each engine. The servo actuators provide engine gimbaling for steering and vehicle control during the burn phases.

The single-engine Centaur does not have a hydraulic TVC system. The hydraulic power unit and servo actuators have been removed and replaced by an electronic control unit (ECU) and two electromechanical actuators (EMA) for thrust vector control of its engine.

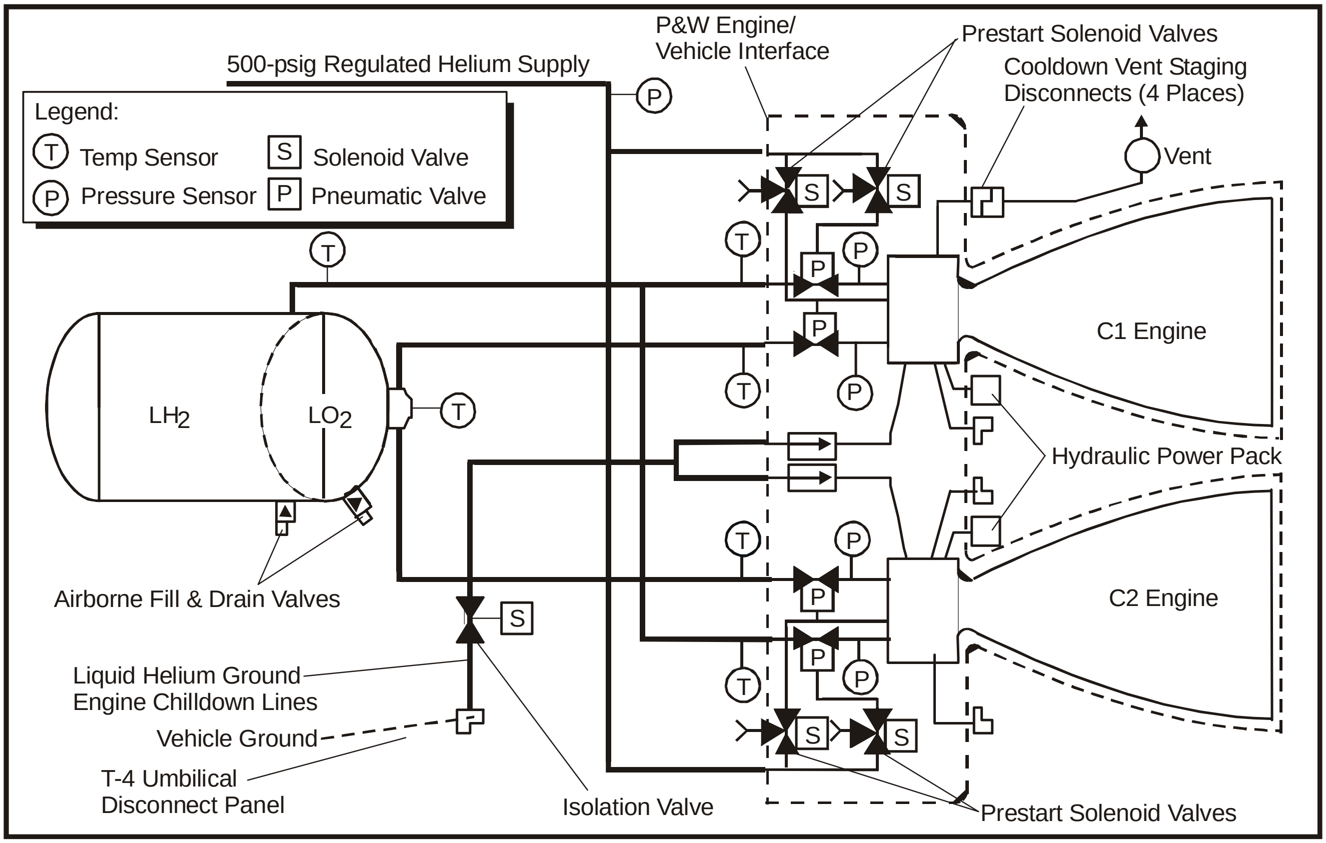


Figure A.2.4.3-2 The Atlas IIAS/IIIB (DEC) Centaur propulsion system consists of two liquid hydrogen/oxygen main engines.

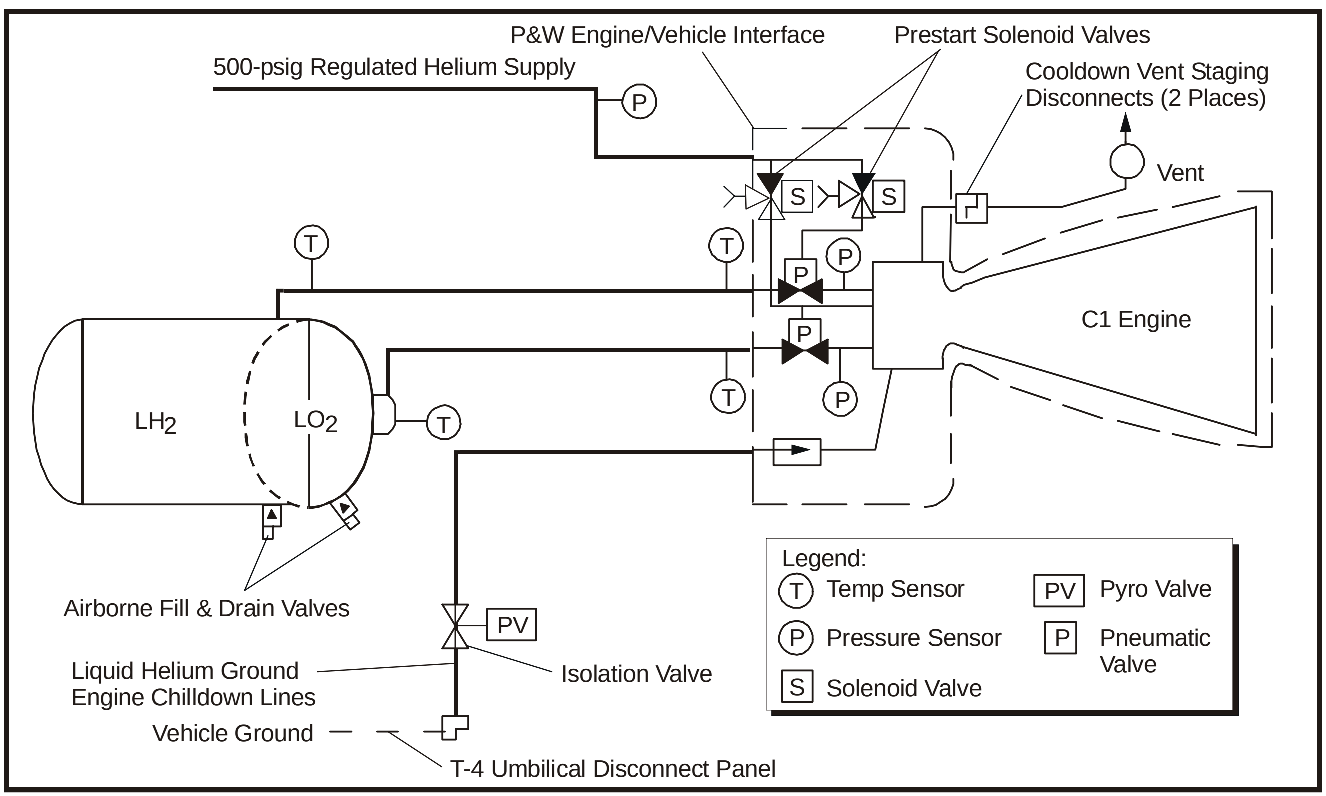


Figure A.2.4.3-3 The Atlas IIIA/IIIB Single-Engine Centaur (SEC) propulsion system consists of a single liquid hydrogen/oxygen main engine.

A.2.4.4 Avionics—Centaur avionics are flight-proven subsystems consisting of guidance, navigation and flight control, telemetry and tracking, secure flight termination capability, and electrical power distribution (Fig. A.2.4.4-1 for IIAS, Fig. A.2.4.4-2 for IIIA/IIIB). These systems control and monitor all vehicle functions during prelaunch and flight and are line-replaceable in the field.

Flight Control Subsystem—The Centaur uses the most accurate guidance system available in the expendable launch vehicle industry. It is based on a standard 1750A processor and ring laser gyros that are mounted inside the INU on the Centaur equipment module. The FCS performs all vehicle-required computations, including guidance, navigation, attitude control, sequencing and separation fire commands, propellant utilization control, and tank pressurization control. Input to the FCS includes incremental velocities and time, quaternion information, Atlas rate data, Atlas and Centaur tank pressures, and Atlas and Centaur PU data. Output from the FCS includes command of 128 solid-state switches in the Centaur remote control unit (CRCU) and 64 solid-state switches in the Atlas RCU (ARCU). Based on commands from the INU, the RCUs control all vehicle-sequencing events: RCS, engines, pressure control, TVC and propellant depletion.

The FCS performs all attitude control, guidance, and navigation computations for both the Atlas and Centaur phases of flight. FCS-commanded open-loop pitch and yaw steering occurs during the early Atlas phase based on winds-aloft measurements taken shortly before launch. Closed-loop guidance steering is then initiated to provide guidance steering based on the mission trajectory requirements.

The ring laser gyro system provides high-accuracy rate measurements for processing by the flight computer. The strapped-down inertial measurement subsystem (IMS) provides the FCS incremental

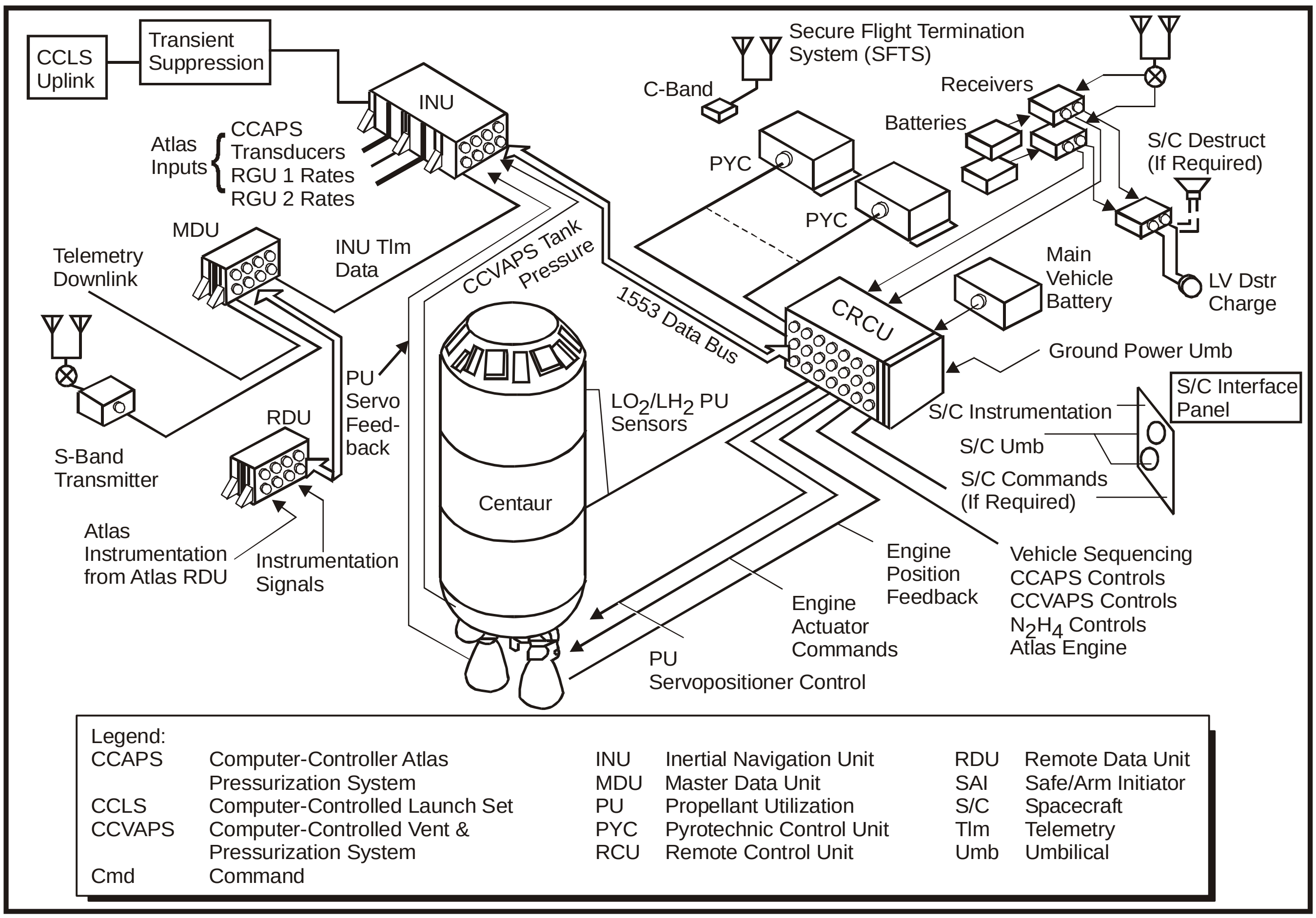
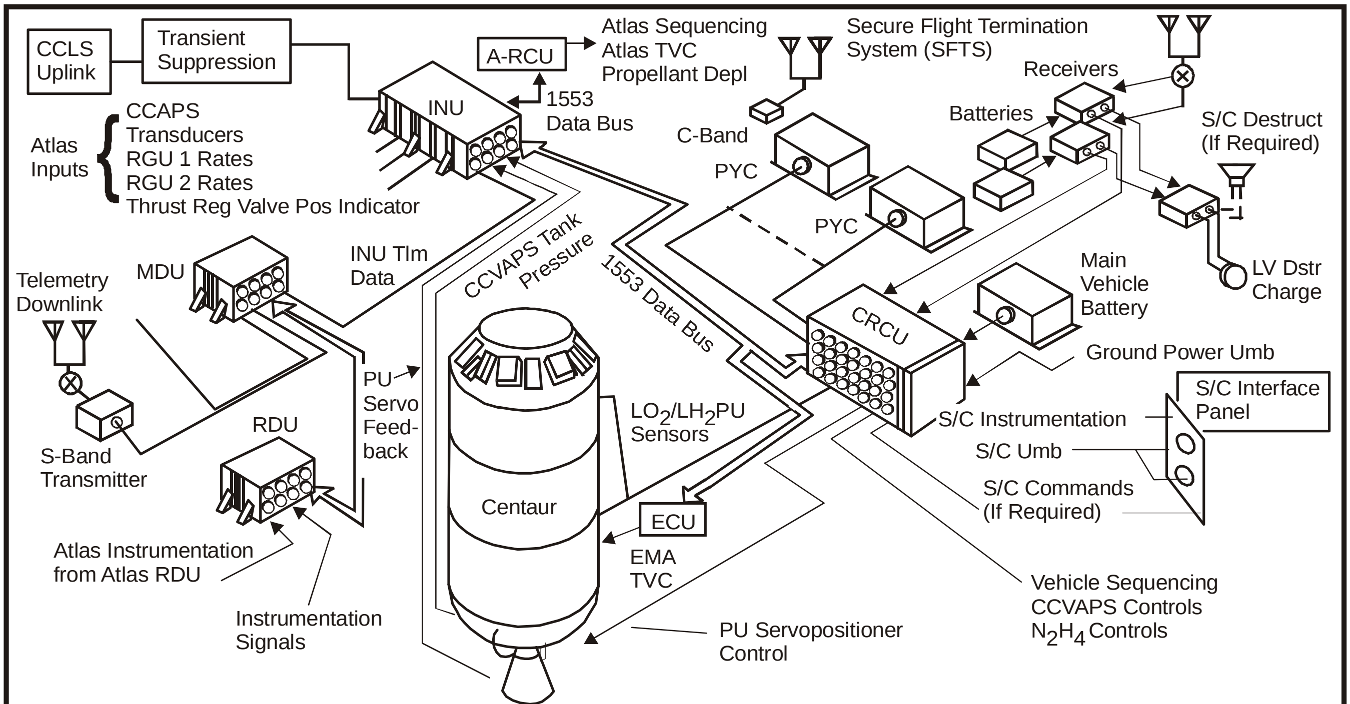


Figure A.2.4.4-1 The Centaur avionics system uses flight-proven hardware to deliver unmatched flight capability (Atlas II/III DEC shown).



Legend:

A-RCU	Atlas RCU	C-RCU	Centaur RCU	RCU	Remote Control Unit
CCAPS	Computer-Controller Atlas Pressurization System	ECU	Electronic Control Unit	RDU	Remote Data Unit
		EMA	Electromechanical Actuator	SAI	Safe/arm Initiator
CCLS	Computer-Controlled Launch Set	INU	Inertial Navigation Unit	S/C	Spacecraft
CCVAPS	Computer-Controlled Vent & Pressurization System	MDU	Master Data Unit	Tlm	Telemetry
		PU	Propellant Utilization	TVC	Thrust Vector Control
Cmd	Command	PYC	Pyrotechnic Control Unit	Umb	Umbilical

Figure A.2.4.4-2 The single-engine Centaur avionics system retains flight-proven hardware.

timing pulses from a precision timing reference, incremental velocity pulses from the accelerometers, and body-to-inertial attitude quaternion data.

Telemetry and Tracking—The Atlas telemetry DAS (Fig. A.2.4.4-3) monitors several hundred vehicle parameters in addition to guidance and navigation (G&N) data before launch and throughout all phases of flight. Critical vehicle parameters are also monitored via hardwired landline instrumentation. Measurements include acceleration, vibration, temperatures, pressures, displacement, currents and voltages, engine pump speeds, and discretes. The programmable data acquisition telemetry system consists of an MDU and two RDUs, one on Atlas and one on the aft end of Centaur. The MDU, located on the Centaur equipment module, provides transducer excitation, signal conditioning, and encoding for all Centaur front-end measurements in addition to receiving and formatting data from the INU and two RDUs. The INU provides data to the MDU over the 1553 data bus. The MDU provides two pulse-code modulation (PCM) outputs; one is connected to the radio frequency (RF) transmitter, the other is used to provide a hardline umbilical link.

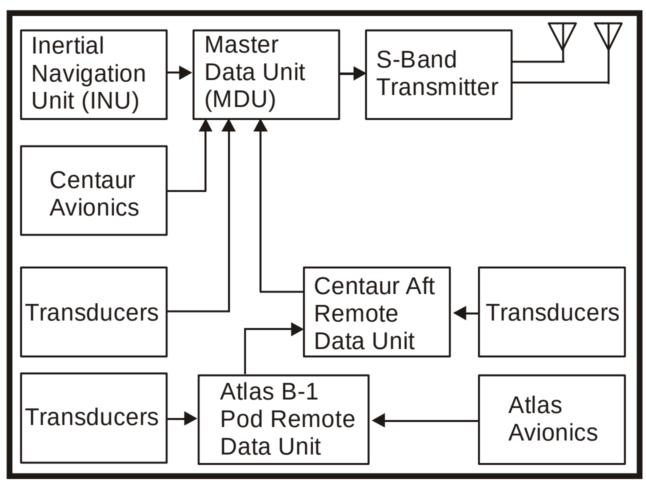


Figure A.2.4.4-3 The telemetry data acquisition system monitors the state and performance of all Atlas systems.

The C-band tracking system aids in determining the real-time position of the launch vehicle for launch site Range Safety tracking requirements. The airborne transponder returns an amplified RF signal when it detects a tracking radar interrogation. The baseline system consists of a noncoherent transponder, a power divider, and two antennas. The Centaur C-band tracking system meets requirements of Eastern and Western Ranges.

FTS—The Centaur FTS provides the capability for termination of Centaur flight in case of an in-flight malfunction. The Centaur FTS is independent of controls of other vehicle systems and from the Atlas FTS. This system has redundancy and is approved for launches at Cape Canaveral Air Force Station and Vandenberg Air Force Base. The secure receiver responds only to high-alphabet code word messages. These messages command engine shutdown, vehicle destruct, and system disable. A payload destruct system can be added, if required, to disable the propulsive capability of the spacecraft. A conical-shaped destruct charge in the payload adapter can be directed at the payload propulsion system. Figure A.2.4.4-4 illustrates the Centaur FTS.

Electrical Power System—The electrical power system consists of a 28 Vdc main vehicle battery, two pyrotechnic batteries (as needed on a mission-peculiar basis), the RCU, a single-point grounding system, and associated electrical harnesses. The RCU provides 128 channels of solid-state switching under INU software control via a MIL-STD-1553B data bus. Arm-safing of critical sequence functions is included. The RCU also provides power changeover capabilities from ground power to internal main vehicle battery power to meet power distribution requirements. Electrical harnesses provide interconnect wiring for avionics equipment with other Centaur systems, interconnections between Atlas and Centaur, power distribution, command, and telemetry signal paths.

Propellant Utilization—The Centaur PU system ensures that an optimum mixture of LH_2 and LO_2 residuals remain in the Centaur propellant tanks at the end of the mission. The PU system continuously measures the mass of the remaining propellants using one externally mounted differential pressure transducer in each of the LH_2 and LO_2 tanks that measure propellant head pressure. INU software converts this head pressure to mass and calculates the sensed mass imbalance between the two tanks. The INU then commands a drive motor on the RL10 engine oxygen flow control valve to correct the sensed error by changing the engine mixture ratio. Drive motor electronic controls are in the RCU.

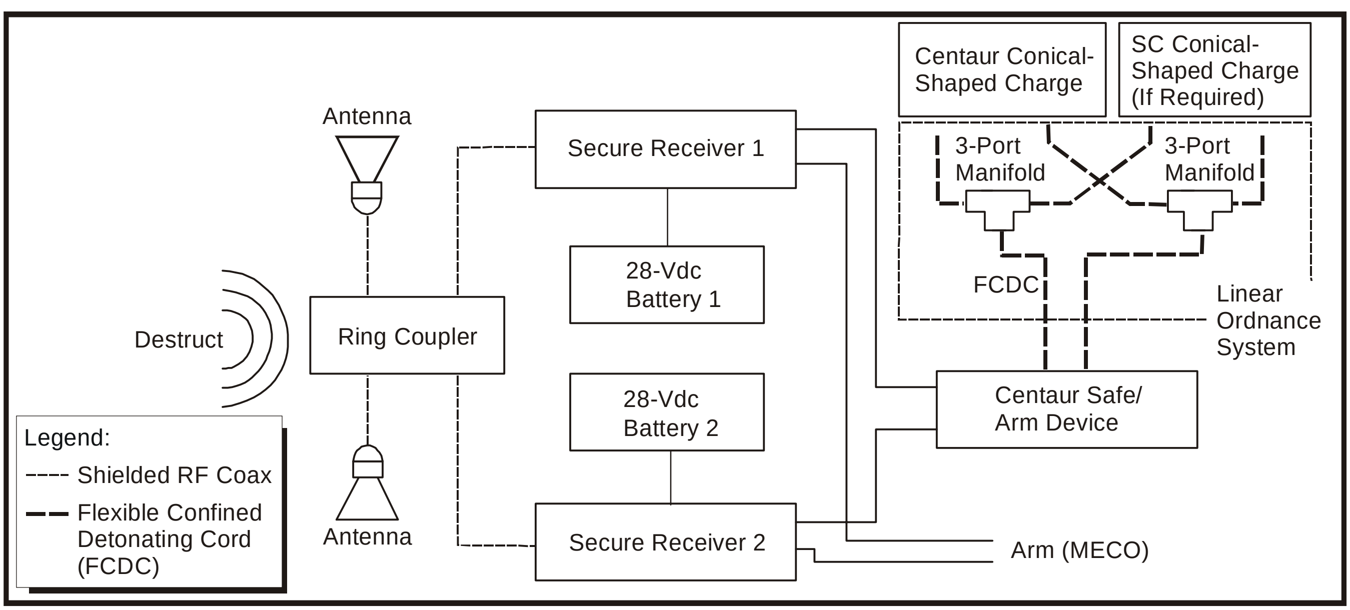


Figure A.2.4.4-4 The Centaur FTS responds to launch site-generated high-alphabet code words.

Propellant Level Indication System—The Centaur PLIS is used during tanking operations to indicate levels of fuel and oxidizer in the propellant tanks. The system consists of hot wire sensors near the top of the propellant tanks and associated hardware for ground control and display. The sensors prevent underloading and overloading of propellants.

A.2.4.5 Centaur Extended Mission Kit (EMK)—As Centaur's primary mission role in recent years has been the launch of geosynchronous communications satellites, power, thermal control, and fluids management hardware on Centaur has been mass-optimized to enable maximum performance for these types of missions. To meet requirements of unique payloads, an EMK has been designed and manufactured to provide Centaur extensions in power and fluid management capabilities required to extend mission duration by up to 2 hours. The EMK adds helium bottles and radiation shielding on the LO_2 tank side wall (Fig. A.2.4.5-1). Centaur's avionics and electrical components are covered with special thermal paints, tapes, and additional radiation shielding to maintain their operating temperatures (Fig. A.2.4.5-2). Several minor modifications to the Centaur aft bulkhead were incorporated across the fleet to allow "kit-able" installation of the EMK. The first flight of the EMK occurred on December 2, 1995, with launch of the NASA/Solar and Heliospheric Observatory (SOHO) on Atlas IIAS.

An EMK for Atlas V extended missions to GSO, typically accommodates 3-burns and over 5 hours of coast time. This EMK includes 250 AH batteries for replacement of two 150 AH batteries, LH_2 tank sidewall radiation shielding, and one added helium bottle installation.

A.2.5 Payload Fairing (PLF)

The PLF protects all components forward of the Centaur from liftoff through atmospheric ascent. A variety of PLF options are available (Sect. 4.1.1 and Appendix D). For example, the Atlas V 400 series combines the CCB with either the large payload fairing (LPF) or the extended-length large payload fairing (EPF). The Atlas V 500 series combines the CCB with a larger and newly available 5-m class diameter payload fairing. The 5-m payload fairing can be selected in either short or medium lengths to accommodate spacecraft volume growth. The Atlas V 500 PLF is a derivative of the composite PLF used

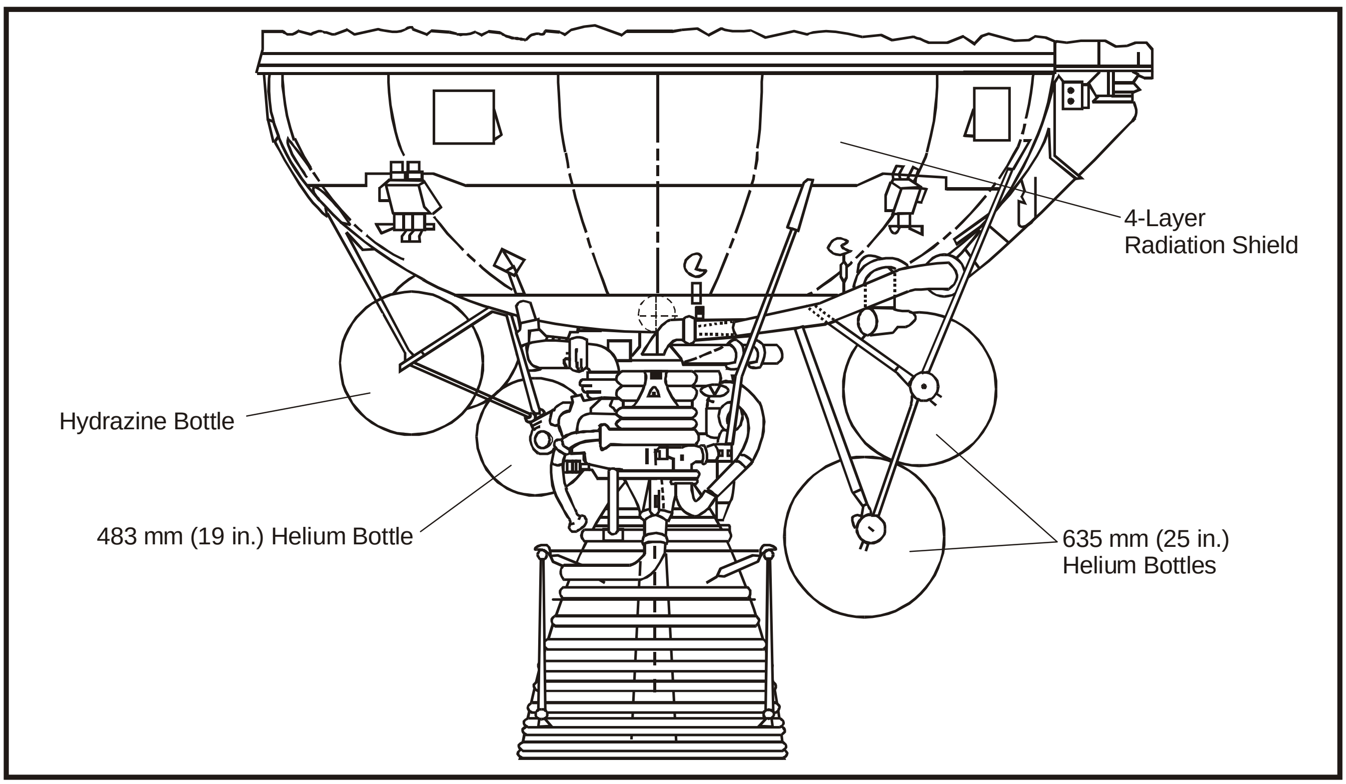


Figure A.2.4.5-1 Centaur Aft Bulkhead for Extended Coasts on the Ariane 5 vehicle. Two lengths of the 5-m PLF are available: the 5-m short, which is 20.7-m (68-ft) long or the 5-m medium, which is 23.4-m (77-ft) long.

A.2.6 Spacecraft and Launch Vehicle Interface Accommodations

Lockheed Martin offers a variety of payload adapters to meet launch vehicle and spacecraft interface requirements (Reference Section 4.1.2 and Appendix E).

A.3 ATLAS AND CENTAUR PRODUCTION AND INTEGRATION

Lockheed Martin provides a combination of experienced production leadership and a production organization dedicated to continuous process improvement supported by state-of-the-art computerized management systems. Our site management concept encourages personnel at each location to concentrate on attaining their respective goals to achieve overall program milestones.

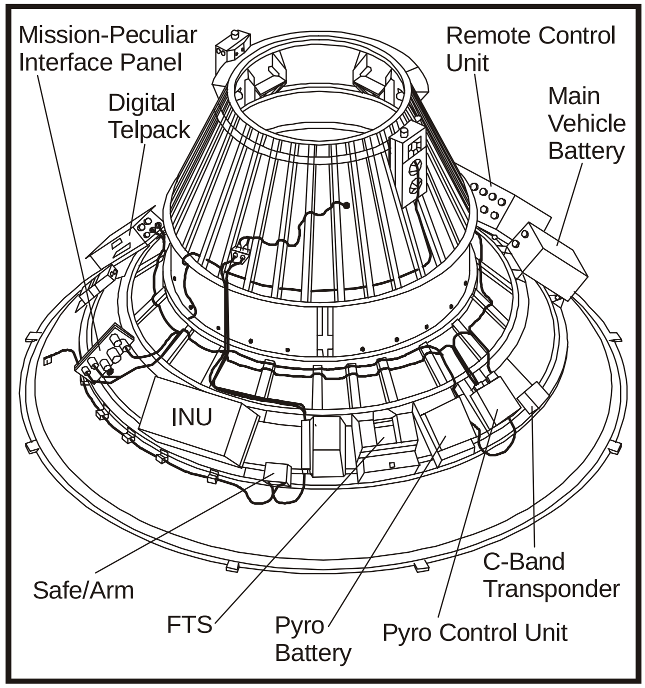


Figure A.2.4.5-2 Typical Centaur Forward Adapter for Extended Coasts

Figures A.3-1 and A.3-2 are top-level roadmaps for Atlas IIAS/III and Atlas V manufacturing. The figures represent the major components of the Atlas and also show the major reviews performed during vehicle manufacturing that provide the checks and balances to ensure Mission Success.

The Atlas V launch vehicle uses common processes and hardware with kits to support mission-unique requirements. Proven manufacturing processes are used to minimize the variability of quality, schedule, and cost. Each factory is focused on continuous improvement through assessment of current process results, identification of root cause, and implementation of improvements.

To accomplish this, manufacturing processes are continuously selected and analyzed for improvement opportunities leading to cycle-time reduction, cell manufacturing, design improvements, producibility improvements, and process improvements. This is typically done using a Kaizen event format. Examples of these improvements include welded engine gimbal mounts, pressure-assisted cryogenic seals, elliptical forward bulkhead, common tank structure for Atlas III and Atlas V Centaurs, near-real-time digital x-ray, cellular manufacturing centers, reduced parts count, and reduced tooling.

Lessons learned from each manufactured vehicle are analyzed and improvements are scheduled for block change implementation on future vehicles.

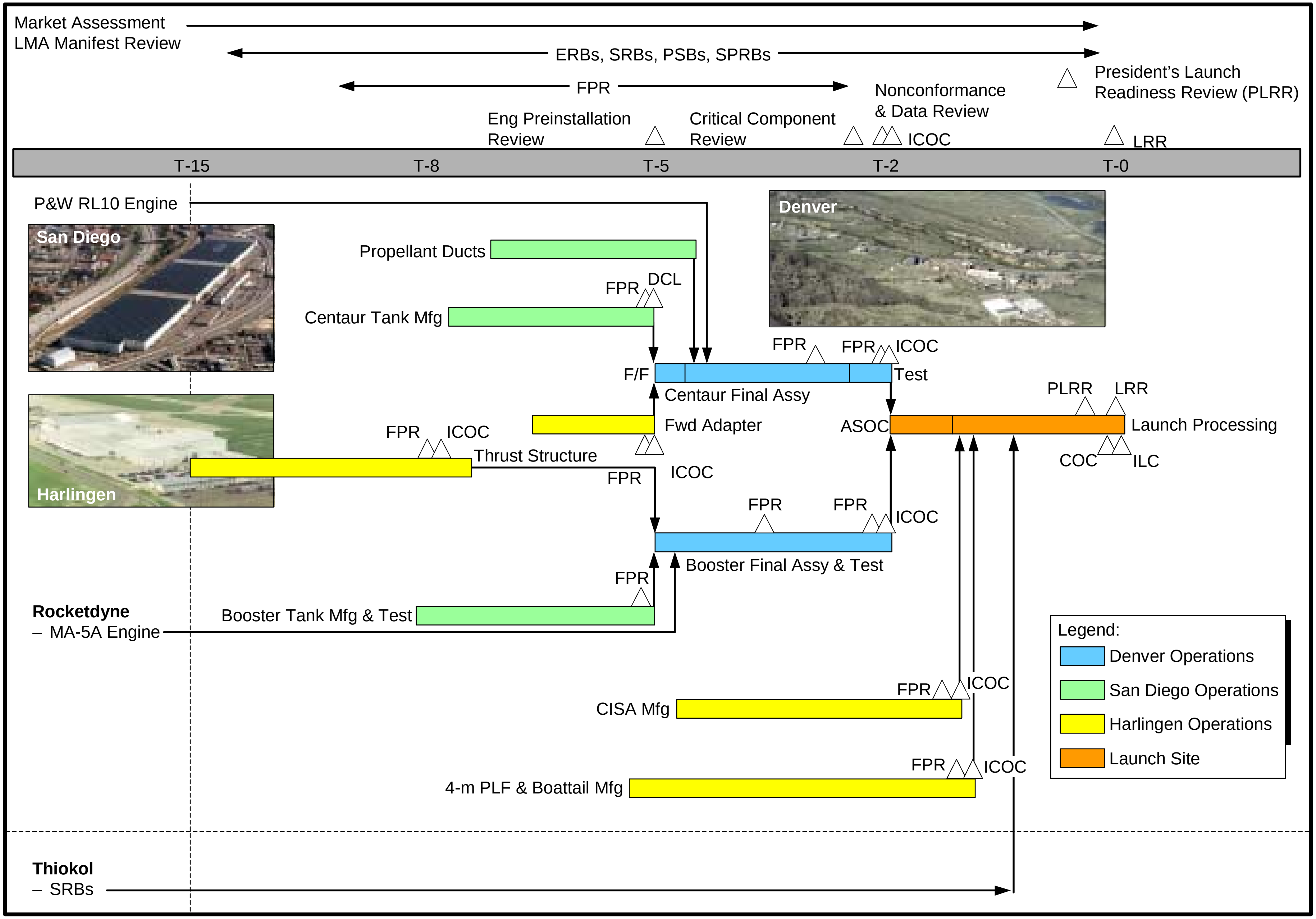


Figure A.3-1 Atlas IIAS and III Manufacturing Road Map

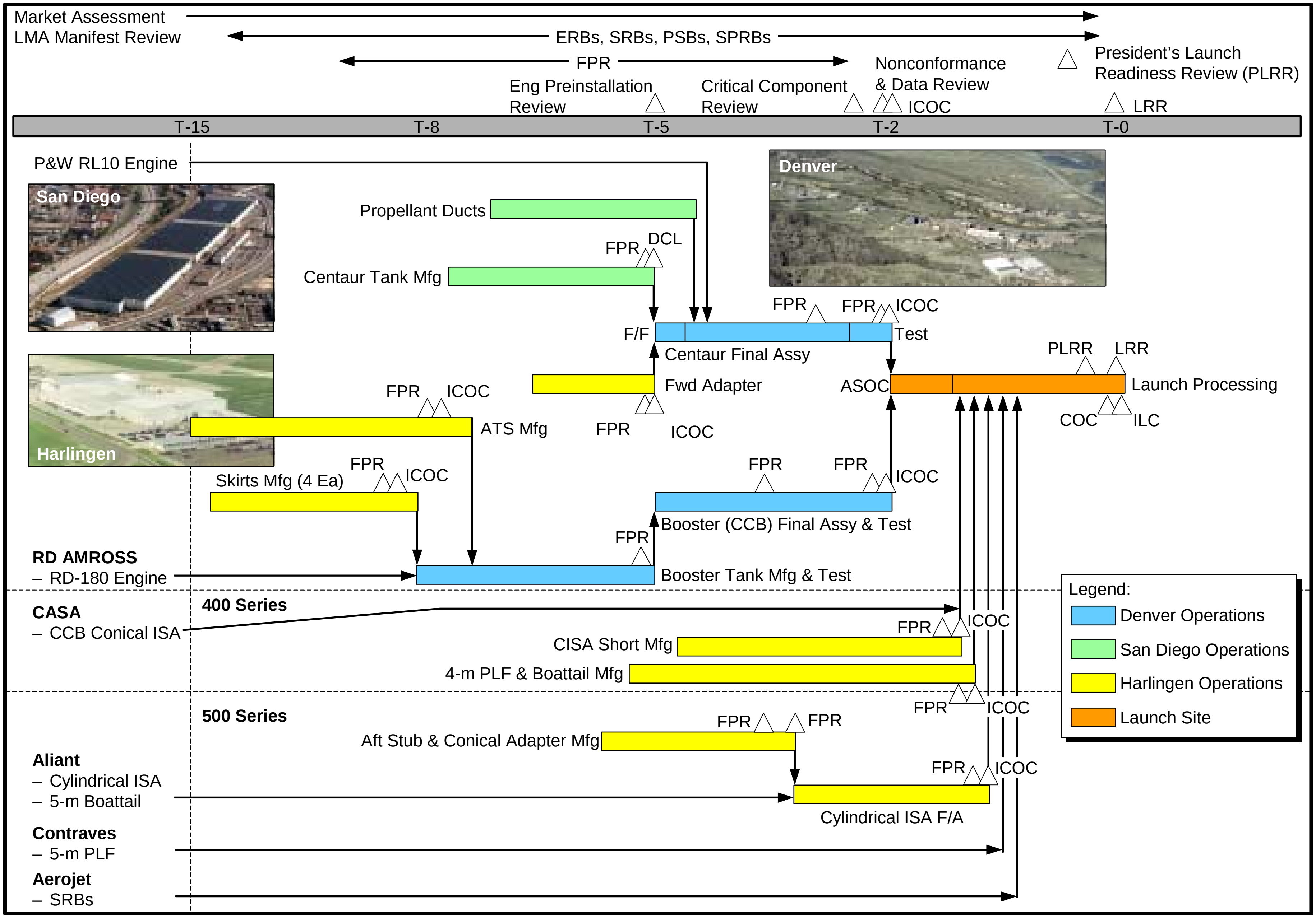


Figure A.3-2 Atlas V Manufacturing Road Map

A.3.1 Organization

Manufacturing of the Atlas V launch vehicle uses an extended enterprise to focus on the key competencies of each supplier. This approach enables each supplier to focus on their area of expertise resulting in providing quality products at a competitive price. Internally within Lockheed Martin, the extended enterprise includes our San Diego, Harlingen, Denver and launch site facilities. As shown in Figure A.3.1-1 major assemblies are produced at our San Diego, Denver factory, and Harlingen facilities that are provided to our Denver final assembly facility for integration, test, and delivery to the launch site. Each is structured as an integral part of the manufacturing flow, focused on specific key competencies (Fig. A.3.1-2).

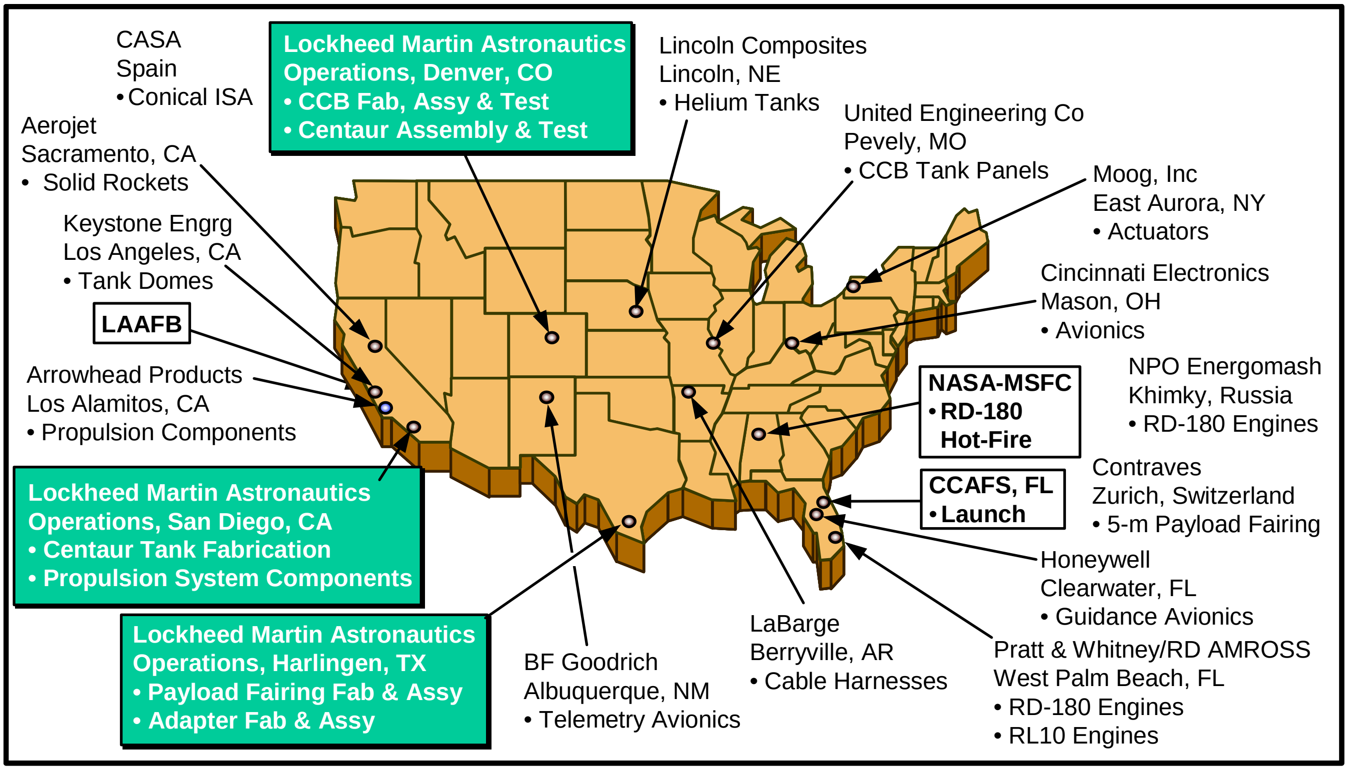


Figure A.3.1-1 Extended Enterprise

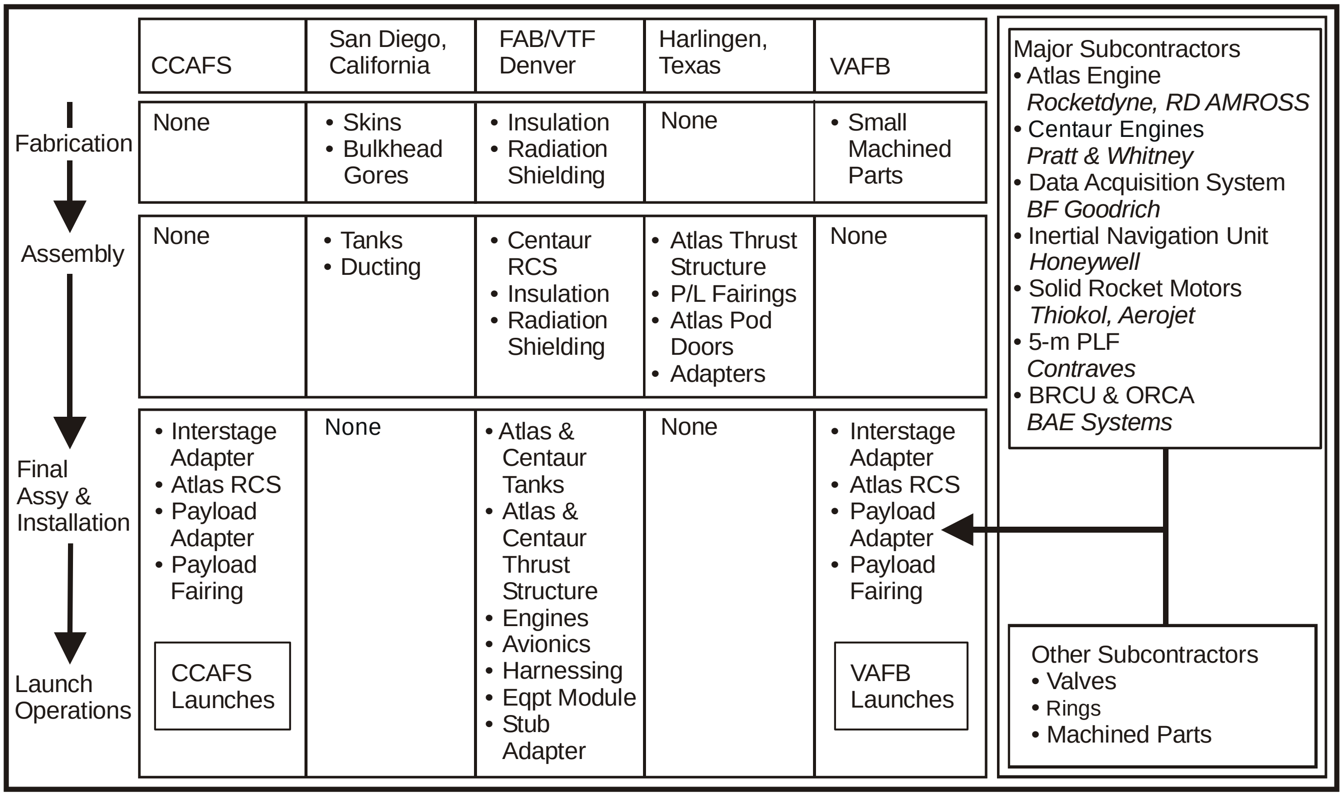


Figure A.3.1-2 Our production approach is mature and proven.

Overall accountability for Atlas production quality, schedule, and cost lies with LMAO Production Operations (Fig. A.3.1-3). Within this organization are all production sites and core support functions of material planning, manufacturing engineering, acquisition management, environmental management, and business management.

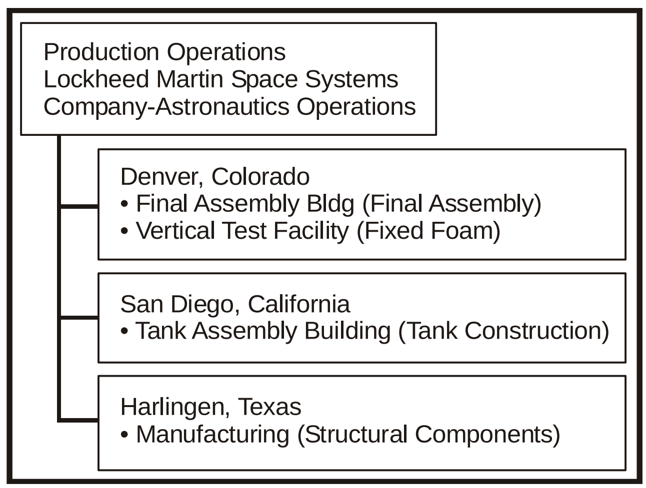


Figure A.3.1-3 Atlas Production Operations Organization

Denver Factory, Denver, Colorado—Denver operations specializes in three areas: (1) aluminum welding of structurally stable tank structures, (2) launch vehicle component installation/final assembly, and (3) system acceptance testing (Fig. 3.1-4). The Atlas V booster (CCB) tank structures are welded in the Denver second floor factory automated weld center, using automated processing and digital x-ray techniques. Final assembly for the CCB and Centaur is performed in the Denver final assembly building (FAB).

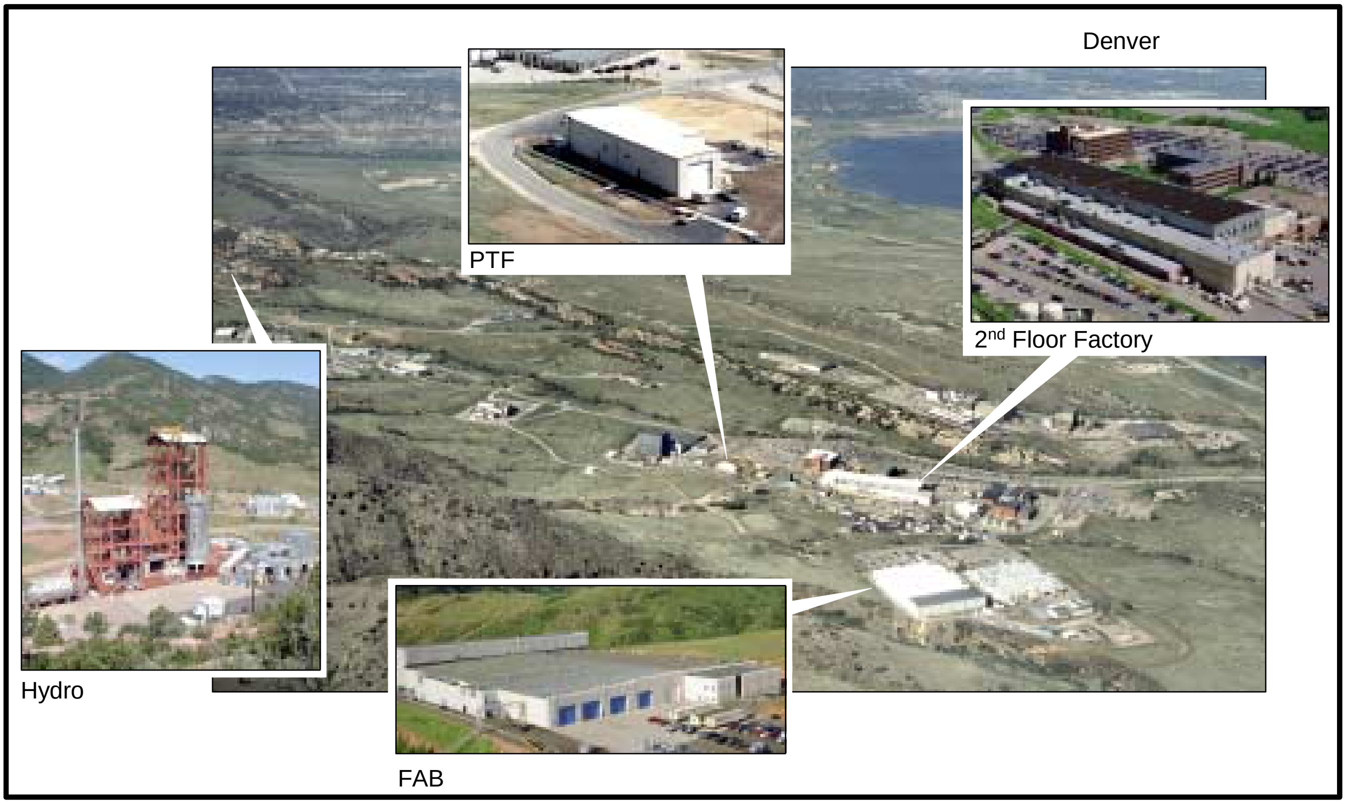


Figure A.3.1-4 Denver Operations

The FAB is a final assembly and storage building encompassing 18,580 m^2 (200,000 ft^2), of which 3,715 m^2 (40,000 ft^2) operate as a Class 100,000 clean room for Centaur final assembly and test. This facility ensures a clean environment for upper-stage production. In addition to a modernized Atlas first-stage final assembly area, the FAB houses subassembly manufacturing, completed vehicle and work-in-process storage, support areas such as tool and print cribs, production test laboratories, and office spaces. Completed in 1995, the FAB is a comprehensive, state-of-the-art launch vehicle manufacturing, integration, and test facility. A modernized fixed-foam application cell within the existing Vertical Test Facility (VTF) (Fig. A.3.1-5) is used to apply fixed foam insulation to Centaur upper stages. Testing is performed in three specialized locations. After weld operations are completed on the CCB tank structures, the tanks are hydrostatically tested and cleaned to LO_2 clean standards. High-pressure system testing of the CCB and Centaur systems are performed in the Pressure Test Facility. System-level acceptance testing is per-formed in the FAB. Following final acceptance, the completed CCB and Centaur are delivered to the launch site by air transport (C5 or AN124 aircraft) (Fig. A.3.1-6, Fig. A.3.1-7, Fig. A.3.1-8, and Fig. A.3.1-9).

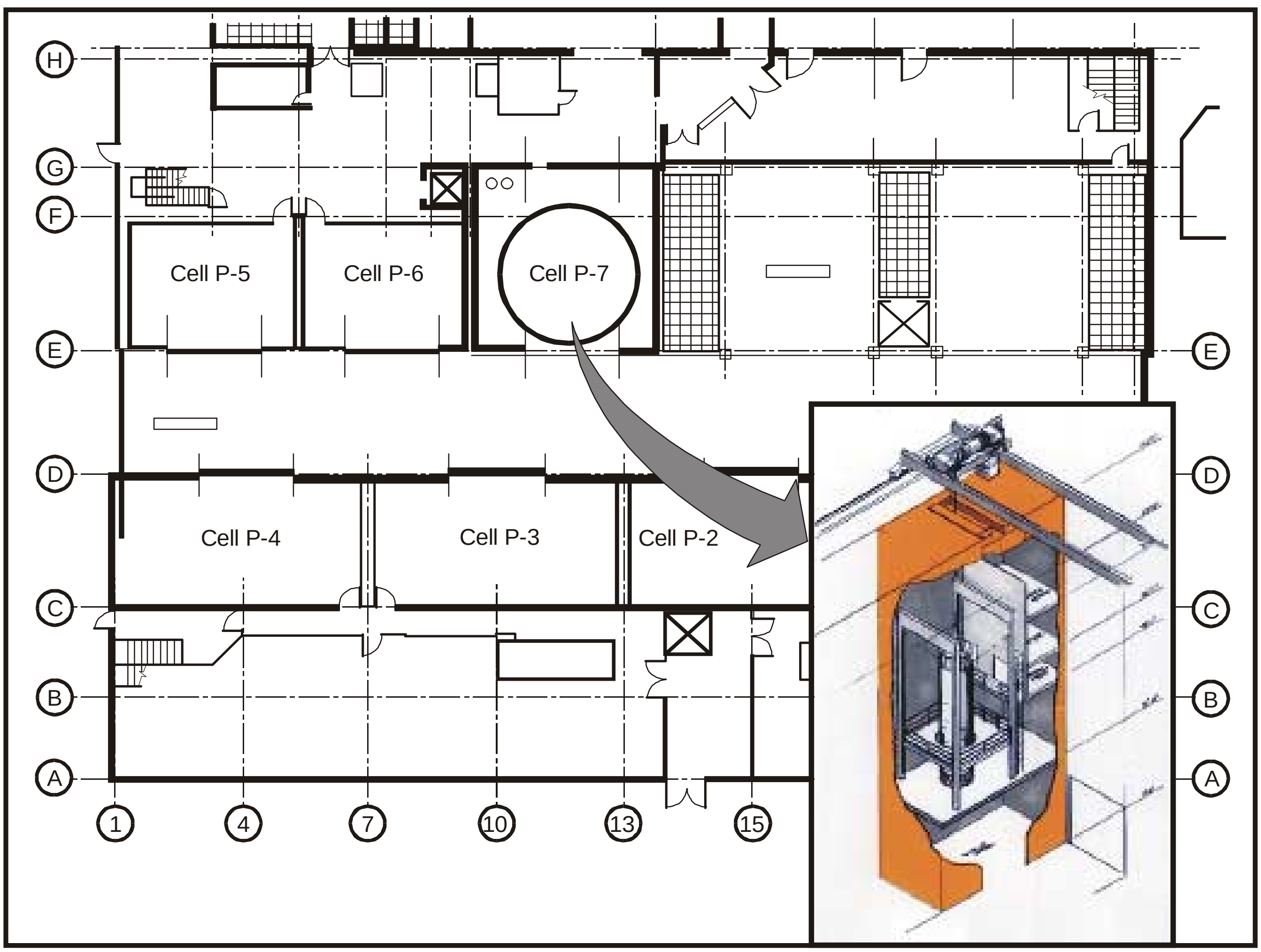


Figure A.3.1-5 A modern fixed-foam facility supports Centaur assembly in Denver.

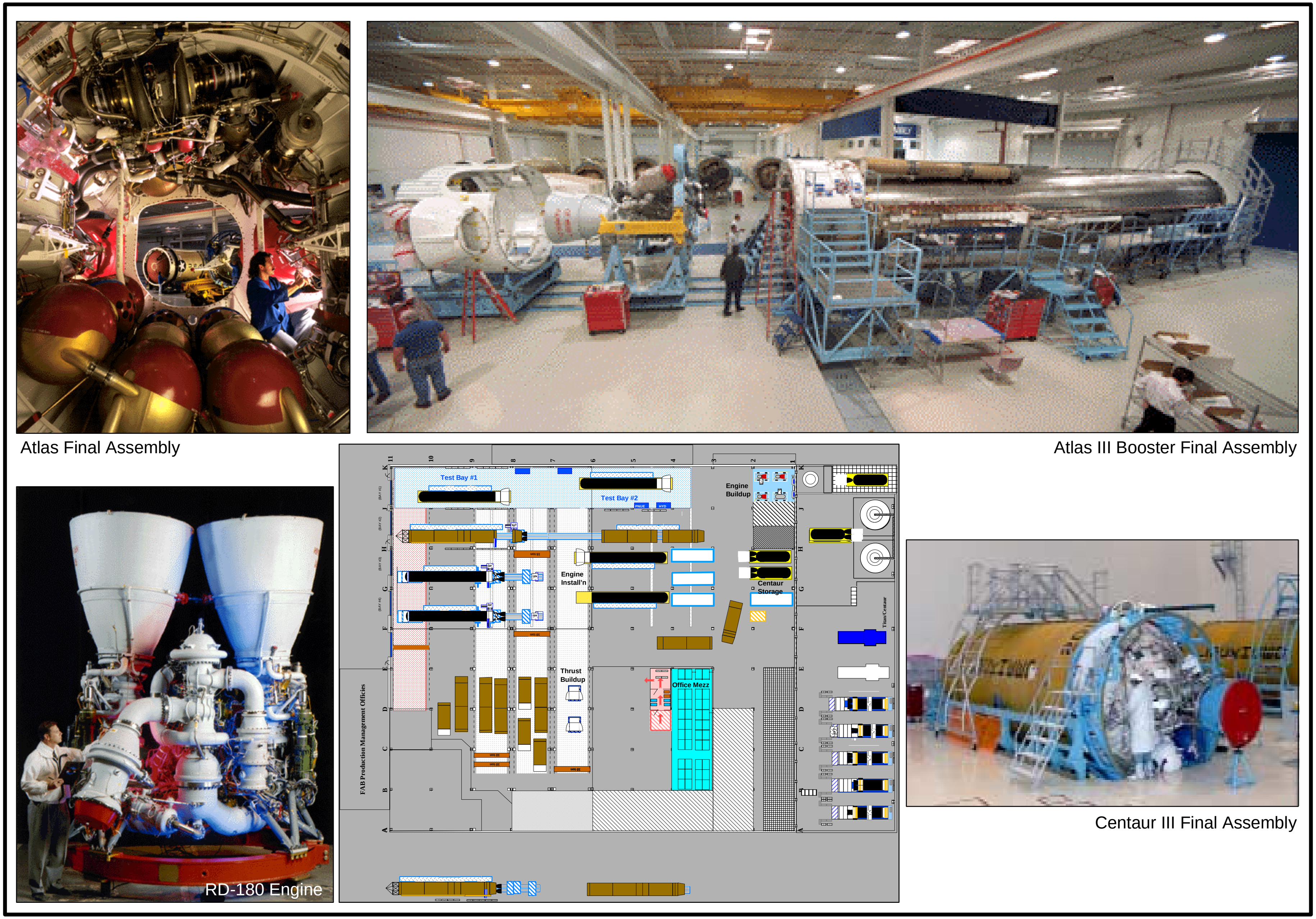


Figure A.3.1-6 Denver Production Flow (Final Assembly)

Welding (Booster)

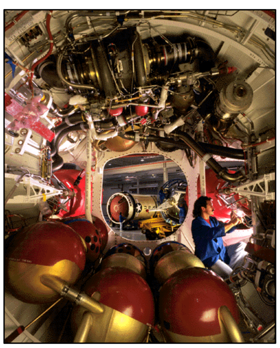
- Variable-Polarity Gas-Tungsten Arc Welding (VPGTAW) Process
- 30 years of Titan Heritage (2,112 Titan Flight Tanks Welded)
- Same Material (2014 Al) As Titan
- Continuous Improvement in First-Pass Yield
  - — 97.7% First-Pass Yield Titan IV
  - — 99.95% First-Pass Yield on Validation LO_2 & RP Tanks
- Improved Tooling

Figure A.3.1-7 CCB Weld Process Summary

Integration & Test

- Hardware Integration Proven on Atlas III
- Centaur Forward Adapter & Engine Mating Proven on Atlas III
- System-Level Test Verified on Atlas/ Centaur III

Figure A.3.1-8 FAB Key Process, Integration, and Test



Hardware Installation/Final Assembly

- 134 + Centaurs Produced/Launched
- Centaur IIIB Assembly Proven on Centaur II/III
- CCB Assembly Proven on Atlas III/Titan IV
- CCB & Centaur Process Reliability Improvements Demonstrated on Atlas II/III
- Atlas II/IIII & Titan Lessons Learned Incorporated into Atlas V Product & Process Design

Figure A.3.1-9 FAB Key Process—Hardware Installation and Final Assembly

San Diego Facility, San Diego, California—San Diego Operations (SDO) is a world-class welding facility that specializes in thin wall, stainless steel resistance, and fusion welding (Fig. A.3.1-10 and A.3.1-11). SDO has been recognized by the Edison Welding Institute as being World Class and has been featured on the cover of the American Welding Society's Welding Journal. SDO also specializes in cryogenic testing and pressure/leak testing. SDO provides the following components for the Atlas V program: Centaur tank structure, Centaur reaction control system, Centaur propulsion ducts, and and propulsion ducts for the Atlas V Booster (CCB) (Fig. A.3.1-12).

Figure A.3.1-10 San Diego Operations Tank Assembly Building

Welding (Centaur)

- Resistance Seam/Spot Welding
- Recognized by EWI As "World Class"
- Proven on 134 Centaurs
- Continuous Improvement in First-Pass Yield
  — 99.84% in 1993
  — 99.98% in 1999

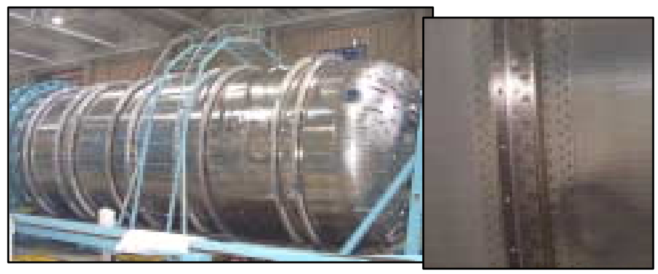


Figure A.3.1-11 Centaur Weld Summary

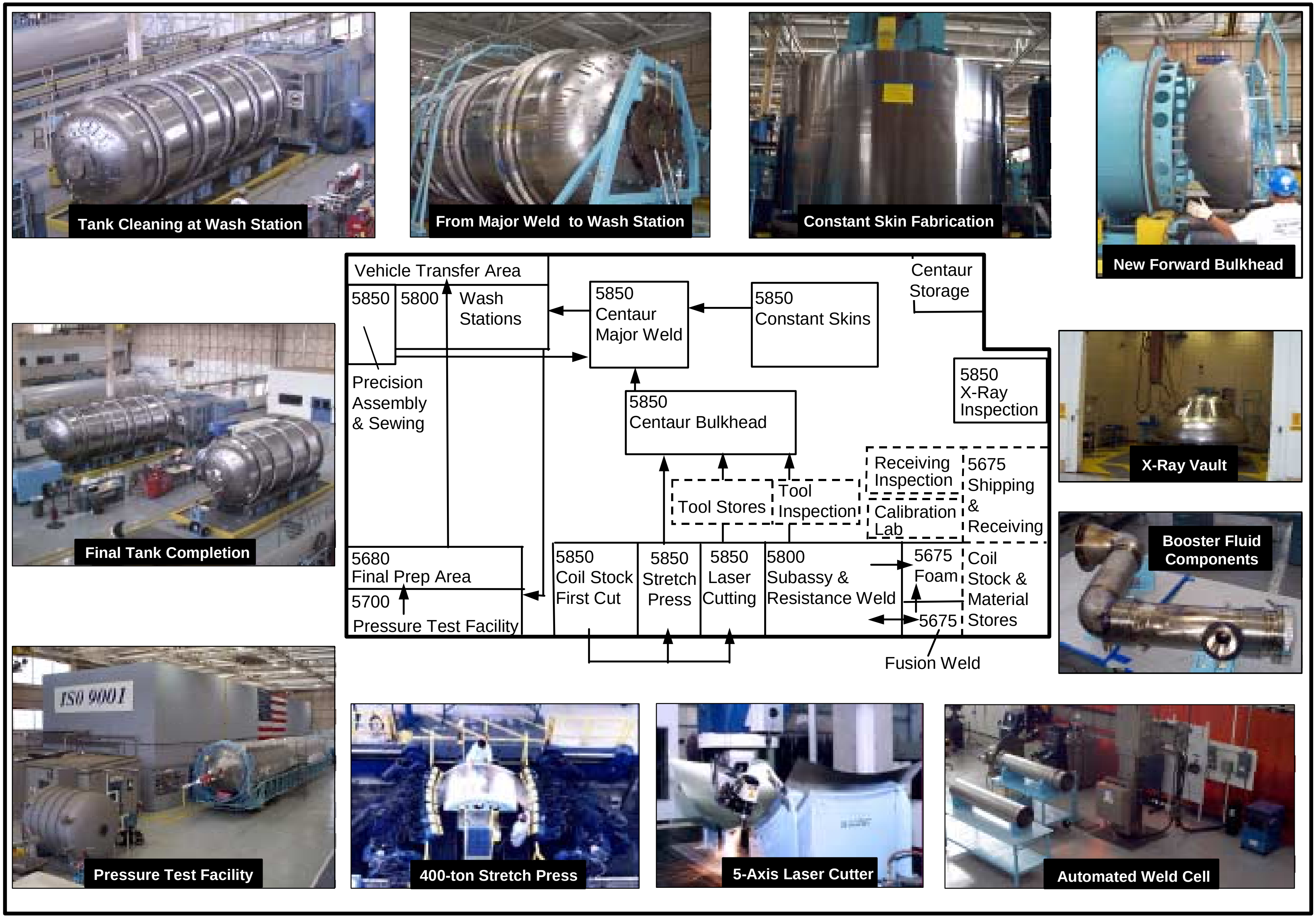


Figure A.3.1-12 San Diego Production Flow

Harlingen Facility, Harlingen, Texas—The Harlingen facility has a diverse array of technical expertise and equipment that permits complete in house fabrication and assembly of major structures (Fig. A.3.1-13). The facility has implemented total quality management, is certified to ANSI/ISO/ASQC 9002, and practices continuous improvement to achieve a high level of quality that results in significant reductions in rework and rejection.

Figure A.3.1-13 Harlingen Operations

The facility performs subassembly, intermediate, and final assembly tasks on major structures for Atlas, Titan, and Athena launch vehicles. Harlingen operations provides the following components for the Atlas V program: Centaur forward adapter, C ISA short (400 series), 4-meter payload fairing and boattail (400 series), Aft stub and conical adapters (500 series), final assembly of the cylindrical ISA, and CCB aft transition structure (ATS), CCB skirts (Fig. A.3.1-14).

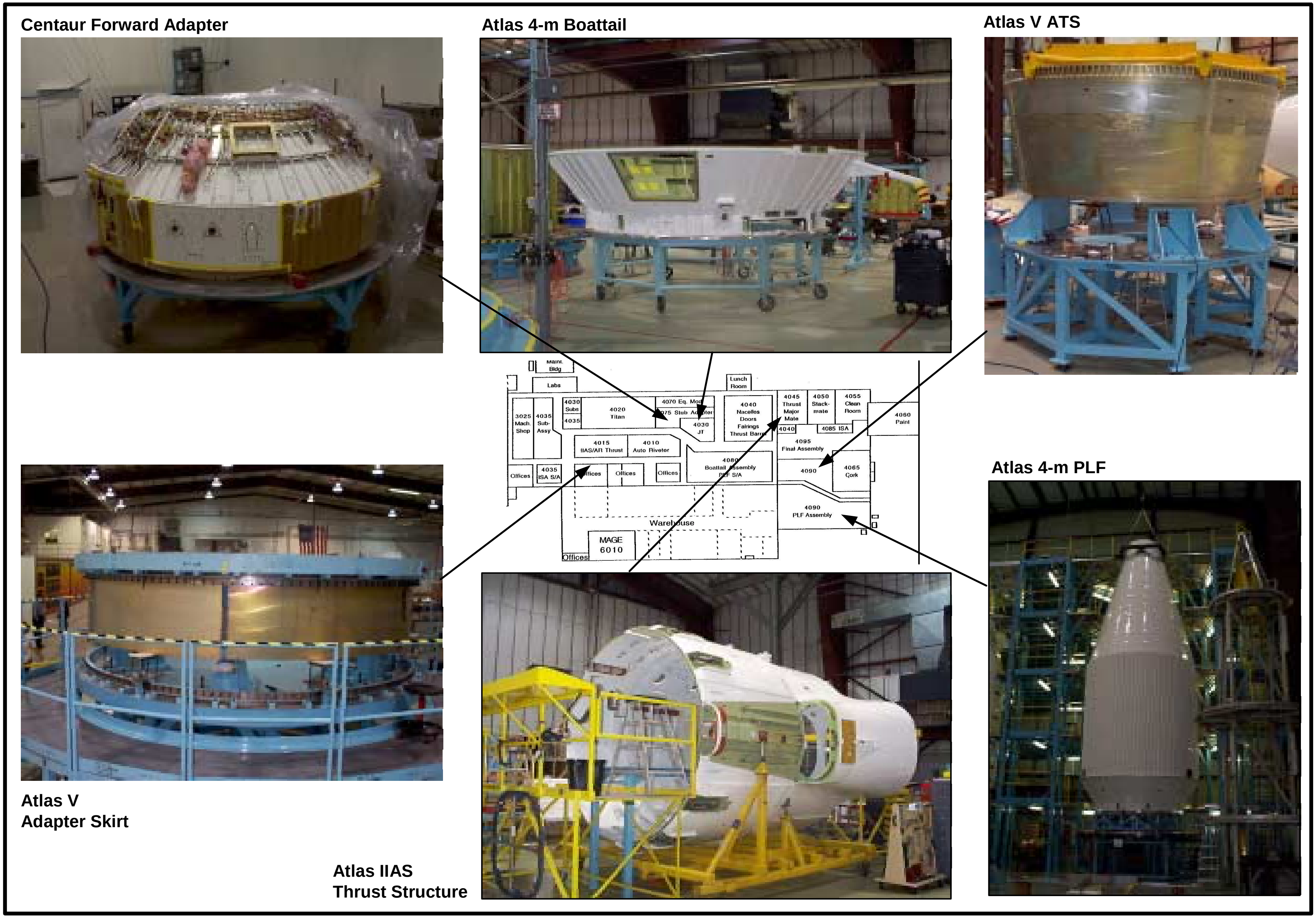


Figure A.3.1-14 Harlingen Production Layout

The key processes for this site are hole drilling and fastener installation (Fig. A.3.1-15). Harlingen ships its products to San Diego, California; Denver, Colorado; and launch sites for final assembly with specific launch vehicles.

Cape Canaveral Air Force Station (CCAFS), Florida—CCAFS is the final destination of the Atlas IIAS, III, and V hardware for East Coast launches. CCAFS concentrates on the final vehicle integration and system checks before launching the vehicles.

Vandenberg Air Force Base (VAFB), California—VAFB is the final destination for the Atlas IIAS hardware for West Coast launches. VAFB concentrates on final vehicle integration and system checks before launching vehicles.

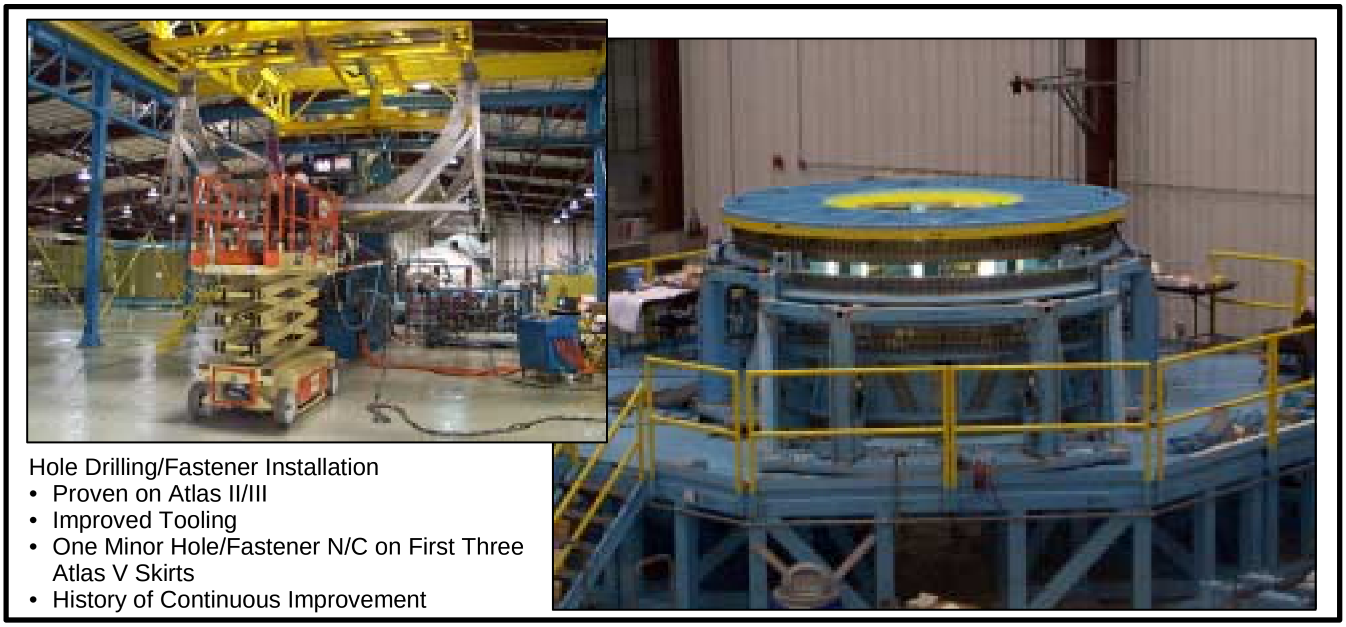


Figure A.3.1-15 Harlingen Key Process

A.4 VEHICLE RELIABILITY PREDICTIONS

Many vehicle and process enhancements that have increased performance and operability over the history of the Atlas program have also increased reliability. The analysis method used by the Atlas program to calculate vehicle reliability is the reliability growth model developed by J. T. Duane. This method uses demonstrated flight data to predict the future performance of the Atlas launch system (Fig. A.4-1). The conservative Reliability Growth Model is based on observations of the reduction in failure rate as the cumulative number of missions increases. When mean missions between flight failures are plotted on a log-log scale against the cumulative missions flown, data points fall approximately in a straight line. The steepness of the slope of this line give a measure of the growth rate of improvements in reliability over time. In our case, the high value achieved is associated with performance upgrades of our operational launch vehicles. This growth rate is the result of our vigorous failure analysis and corrective action system, and our focus on controlling processes. We expect a high probability of mission success occurrence by using only proven hardware of high inherent design reliability and processes that are controlled and stable.

Today's demonstrated reliability record of 0.9721 is based on flight history through Atlas IIIA (Fig. A.4-1). This record is based on timely, successful launches and satellite system support. For the Atlas program, it represents pride in playing an integral part in successful space program for more than 35 years.

For Atlas V, as noted earlier, the design philosophy for reliability is controlled by fault avoidance and fault tolerance. Fault avoidance focuses on component selection, reduction in number of critical subsystems, minimizing single point failures, and applying lessons learned. Examples of fault avoidance are:

1) Reduction in the number of critical subsystems,
2) Reduction in the number of staging events,
3) Reduction in overall parts count,
4) Booster main engine health check before launch,
5) A structurally stable booster,
6) No SRM thrust vector control,
7) No SRM joints,

The Atlas V program has designed out significant failure modes that are present in other launch vehicles that are at the edge of their performance margins.

Fault tolerance focuses on designing the vehicle to achieve mission success despite the possible existence of faults. Though there are no formal requirements beyond the primary reliability requirement for the overall launch vehicle, fault tolerance is incorporated through redundancy in both the hardware and software as well as through the general design process. Examples of Atlas V fault tolerance are:

1) Dual direct spark ignition for the RL10 engines,
2) New fault tolerant inertial navigation unit (FTINU),
3) Redundant avionics interfaces for control of engine functions,
4) Redundant booster control systems,
5) Dual redundant power input in the Booster Remote Control Unit (BRCU),
6) New redundant rate gyro unit,
7) Error correcting code.

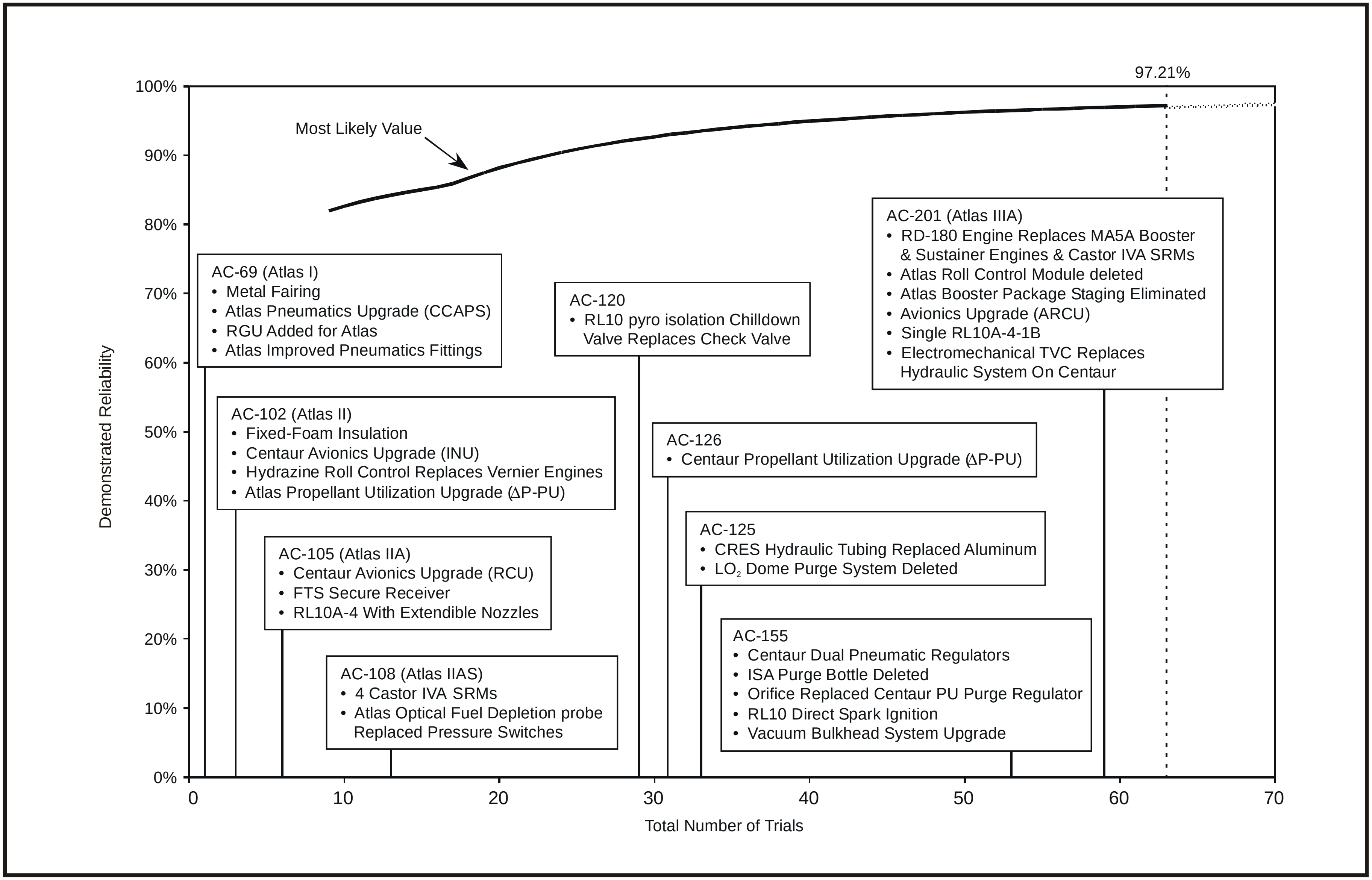


Figure A.4-1 Atlas/Centaur Reliability Using Duane Reliability Growth Method—Through AC-201

With the system design based on common hardware elements (and requirements that are derived and verified by qualification and acceptance testing), the final design encompasses all requirements across all vehicle configurations and missions. The robustness that is achieved from this design approach enables the system to fly through some failures using the available excess performance, stability margin, and structural margin.

The reliability predictions for Atlas V based on this design approach are summarized in Table A.4-1. These predictions were performed for all flight and ground subsystems required for launch and mission success during the time from launch commit through the completion of CCAM. For the Atlas V program, reliability predictions are defined at two levels: Design Reliability and Mission Reliability. Design reliability predictions account for potential mission failure modes that have their genesis in the design of system hardware. The failure rates used to calculate design reliability are derived from MIL-HDBK-217 (Reliability Predictions for Electronic Equipment), Reliability Analysis Center Non-electronic Parts Reliability Database (NPRD-95), and subcontractor data. Mission reliability on the other hand, includes design reliability as well as failure modes introduced by manufacturing processes, assembly, test, and system integration. Mission reliability is calculated by performing a Bayesian Update using historical flight data from existing Atlas, Titan, and Centaur vehicles. Because flight data contain all potential failure modes, both design and process related, the mission reliability is the better estimate of the true reliability of the flight hardware. By comparison, the calculated value for Atlas IIAS design reliability is 0.9873.

In summary, the evolutionary foundation of the Atlas V program optimizes system reliability and overall vehicle robustness to continue the strong Atlas tradition of mission success.

Table A.4-1 Typical Atlas V Reliability Predictions—GTO Mission Profile

Atlas V Vehicle Configuration	Design Reliability	Mission Reliability
AV-401	0.9954	0.9865
AV-531	0.9930	0.9826
AV-551	0.9916	0.9804

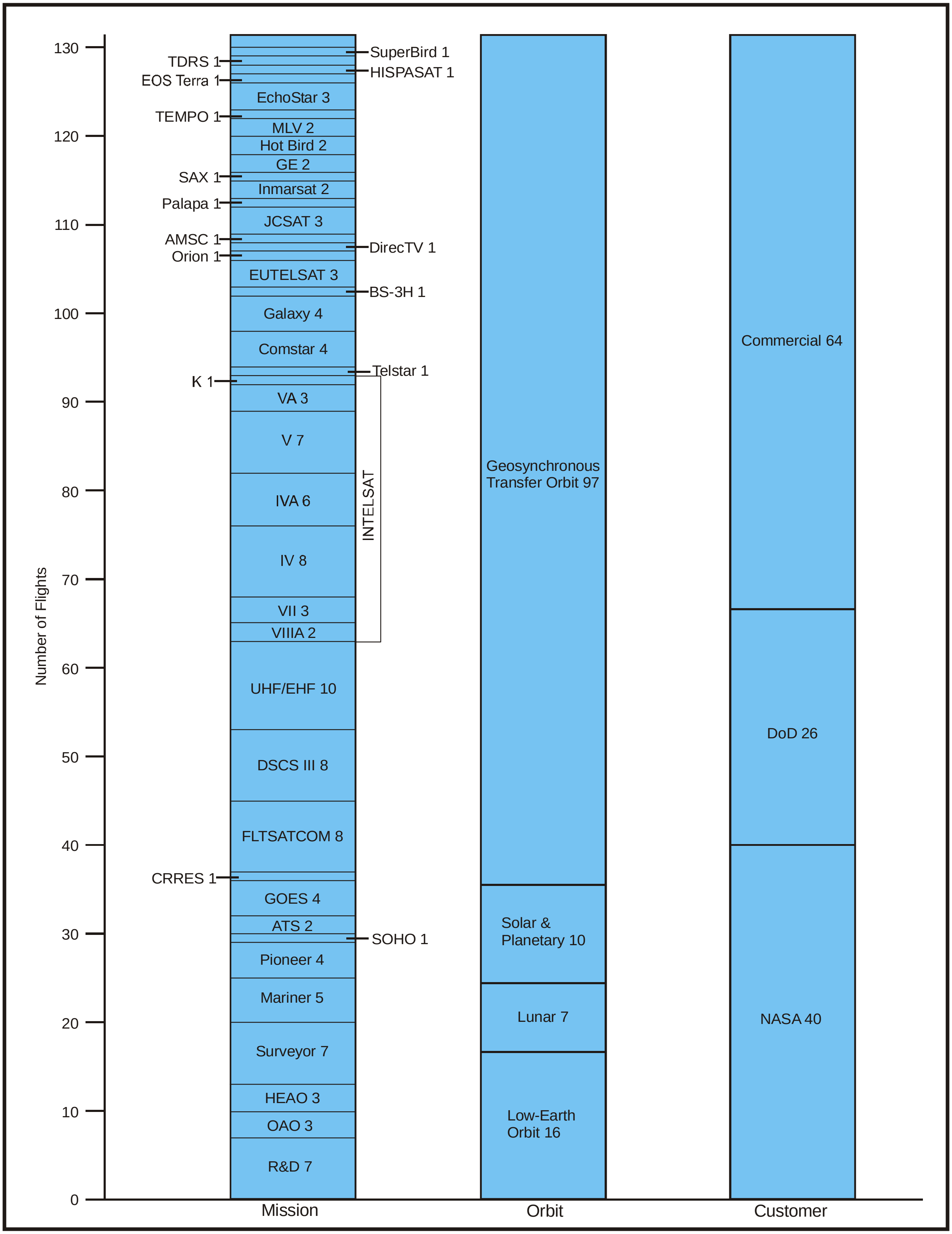


Figure A.1-4 Diverse payloads demonstrate Atlas versatility and adaptability.

Table A.1-1 Atlas/Centaur Launch History

Date	Mission	Vehicle	LV Type	Mission Type	LV Results
1962					
May 8	R&D	AC-1	LV-3C/A	R&D	Failure
1963					
November 27	R&D	AC-2	LV-3C/B	R&D	Success
1964					
June 30	R&D	AC-3	LV-3C/C	R&D	Failure
December 11	R&D	AC-4	LV-3C/C	R&D	Success
1965					
March 2	R&D	AC-5	LV-3C/C	R&D	Failure
August 11	R&D	AC-6	LV-3C/D	R&D	Success
1966					
April 7	R&D	AC-8	LV-3C/D	R&D	Failure
May 30	Surveyor	AC-10	LV-3C/D	Lunar Intercept	Success
September 20	Surveyor	AC-7	LV-3C/D	Lunar Intercept	Success
October 26	Mariner Mars	AC-9	LV-3C/D	Interplanetary	Success
1967					
April 17	Surveyor	AC-12	LV-3C/D	Lunar Intercept	Success
July 14	Surveyor	AC-11	LV-3C/D	Lunar Intercept	Success
September 8	Surveyor	AC-13	SLV-3C/D	Lunar Intercept	Success
November 7	Surveyor	AC-14	SLV-3C/D	Lunar Intercept	Success
1968					
January 7	Surveyor	AC-15	SLV-3C/D	Lunar Intercept	Success
August 10	ATS-D	AC-17	SLV-3C/D	GTO	Failure
December 7	OAO-A	AC-16	SLV-3C/D	LEO	Success
1969					
February 24	Mariner Mars	AC-20	SLV-3C/D	Interplanetary	Success
March 27	Mariner Mars	AC-19	SLV-3C/D	Interplanetary	Success
August 12	ATS-E	AC-18	SLV-3C/D	GTO	Success
1970					
November 30	OAO-B	AC-21	SLV-3C/D	LEO	Failure
1971					
January 25	INTELSAT IV	AC-25	SLV-3C/D	GTO	Success
May 8	Mariner Mars	AC-24	SLV-3C/D	Interplanetary	Failure
May 30	Mariner Mars	AC-23	SLV-3C/D	Interplanetary	Success
December 19	INTELSAT IV	AC-26	SLV-3C/D	GTO	Success
1972					
January 22	INTELSAT IV	AC-28	SLV-3C/D	GTO	Success
March 2	Pioneer F	AC-27	SLV-3C/D	Interplanetary	Success
June 13	INTELSAT IV	AC-29	SLV-3C/D	GTO	Success
August 21	OAO-C	AC-22	SLV-3C/D	LEO	Success
1973					
April 5	Pioneer G	AC-30	SLV-3D/D-1A	Interplanetary	Success
August 23	INTELSAT IV	AC-31	SLV-3D/D-1A	GTO	Success
November 3	Mariner Venus/Mercury	AC-34	SLV-3D/D-1A	Interplanetary	Success
1974					
November 21	INTELSAT IV	AC-32	SLV-3D/D-1A	GTO	Success

Table A.1-1 Atlas/Centaur Launch History (cont)

Date	Mission	Vehicle	LV Type	Mission Type	LV Results
1975					
February 20	INTELSAT IV	AC-33	SLV-3D/D-1A	GTO	Failure
May 22	INTELSAT IV	AC-35	SLV-3D/D-1A	GTO	Success
September25	INTELSAT IVA	AC-36	SLV-3D/D-1AR	GTO	Success
1976					
January 29	INTELSAT IVA	AC-37	SLV-3D/D-1AR	GTO	Success
May 13	COMSTAR	AC-38	SLV-3D/D-1AR	GTO	Success
July 22	COMSTAR	AC-40	SLV-3D/D-1AR	GTO	Success
1977					
May 26	INTELSAT IVA	AC-39	SLV-3D/D-1AR	GTO	Success
August 12	HEAO-A	AC-45	SLV-3D/D-1AR	LEO	Success
September 29	INTELSAT IVA	AC-43	SLV-3D/D-1AR	GTO	Failure
1978					
January 6	INTELSAT IVA	AC-46	SLV-3D/D-1AR	GTO	Success
February 9	FLTSATCOM	AC-44	SLV-3D/D-1AR	GTO	Success
March 31	INTELSAT IVA	AC-48	SLV-3D/D-1AR	GTO	Success
May 20	Pioneer Venus	AC-50	SLV-3D/D-1AR	Interplanetary	Success
June 29	COMSTAR	AC-41	SLV-3D/D-1AR	GTO	Success
August 8	Pioneer Venus	AC-51	SLV-3D/D-1AR	Interplanetary	Success
November 13	HEAO-B	AC-52	SLV-3D/D-1AR	LEO	Success
1979					
May 4	FLTSATCOM	AC-47	SLV-3D/D-1AR	GTO	Success
September 20	HEAO-C	AC-53	SLV-3D/D-1AR	LEO	Success
1980					
January 17	FLTSATCOM	AC-49	SLV-3D/D-1AR	GTO	Success
October 30	FLTSATCOM	AC-57	SLV-3D/D-1AR	GTO	Success
December 6	INTELSAT V	AC-54	SLV-3D/D-1AR	GTO	Success
1981					
February 21	COMSTAR	AC-42	SLV-3D/D-1AR	GTO	Success
May 23	INTELSAT V	AC-56	SLV-3D/D-1AR	GTO	Success
August 5	FLTSATCOM	AC-59	SLV-3D/D-1AR	GTO	Success
December 15	INTELSAT V	AC-55	SLV-3D/D-1AR	GTO	Success
1982					
March 4	INTELSAT V	AC-58	SLV-3D/D-1AR	GTO	Success
September 28	INTELSAT V	AC-60	SLV-3D/D-1AR	GTO	Success
1983					
May 19	INTELSAT V	AC-61	SLV-3D/D-1AR	GTO	Success
1984					
June 9	INTELSAT VA	AC-62	G/D-1AR	GTO	Failure
1985					
March 22	INTELSAT VA	AC-63	G/D-1AR	GTO	Success
June 29	INTELSAT VA	AC-64	G/D-1AR	GTO	Success
September 28	INTELSAT VA	AC-65	G/D-1AR	GTO	Success
1986					
December 4	FLTSATCOM	AC-66	G/D-1AR	GTO	Success
1987					
March 26	FLTSATCOM	AC-67	G/D-1AR	GTO	No Trial

Table A.1-1 Atlas/Centaur Launch History (cont)

Date	Mission	Vehicle	LV Type	Mission Type	LV Results
1989					
September 25	FLTSATCOM	AC-68	G/D-1AR	GTO	Success
1990					
July 25	CRRES	AC-69	Atlas I	Elliptical	Success
1991					
April 18	BS-3H	AC-70	Atlas I	GTO	Failure
December 7	EUTELSAT II	AC-102	Atlas II	Supersynchronous	Success
1992					
February 10	DSCS III B14	AC-101	Atlas II	GTO	Success
March 13	Galaxy V	AC-72	Atlas I	GTO	Success
June 10	INTELSAT-K	AC-105	Atlas IIA	GTO	Success
July 2	DSCS III B12	AC-103	Atlas II	GTO	Success
August 22	Galaxy I-R	AC-71	Atlas I	GTO	Failure
1993					
March 25	UHF F/O F1	AC-74	Atlas I	Subsynchronous	Anomaly
July 19	DSCS III B9	AC-104	Atlas II	GTO	Success
September 3	UHF F/O F2	AC-75	Atlas I	Subsynchronous	Success
November 28	DSCS III B10	AC-106	Atlas II	GTO	Success
December 15	Telstar 401	AC-108	Atlas IIAS	GTO	Success
1994					
April 13	GOES I	AC-73	Atlas I	Supersynchronous	Success
June 24	UHF F/O F3	AC-76	Atlas I	Subsynchronous	Success
August 3	DirecTV D2	AC-107	Atlas IIA	Supersynchronous	Success
October 6	INTELSAT 703	AC-111	Atlas IIAS	Supersynchronous	Success
November 29	Orion F1	AC-110	Atlas IIA	Supersynchronous	Success
1995					
January 10	INTELSAT 704	AC-113	Atlas IIAS	Supersynchronous	Success
January 28	EHF F4	AC-112	Atlas II	Subsynchronous	Success
March 23	INTELSAT 705	AC-115	Atlas IIAS	Supersynchronous	Success
April 7	AMSC-1	AC-114	Atlas IIA	Supersynchronous	Success
May 23	GOES-J	AC-77	Atlas I	Supersynchronous	Success
May 31	EHF F5	AC-116	Atlas II	Subsynchronous	Success
July 31	DSCS III B7	AC-118	Atlas IIA	GTO	Success
August 28	JCSAT-3	AC-117	Atlas IIAS	Supersynchronous	Success
October 22	EHF F6	AC-119	Atlas II	Subsynchronous	Success
December 2	SOHO	AC-121	Atlas IIAS	Libration Point	Success
December 14	Galaxy IIIR	AC-120	Atlas IIA	Subsynchronous	Success
1996					
January 31	Palapa	AC-126	Atlas IIAS	Supersynchronous	Success
April 3	Inmarsat 1	AC-122	Atlas IIA	GTO	Success
April 30	SAX	AC-78	Atlas I	LEO	Success
July 25	EHF F7	AC-125	Atlas II	Subsynchronous	Success
September 8	GE-1	AC-123	Atlas IIA	Supersynchronous	Success
November 21	Hotbird	AC-124	Atlas IIA	GTO	Success
December 17	Inmarsat 3	AC-129	Atlas IIA	GTO	Success

Table A.1-1 Atlas/Centaur Launch History (concl)

Date	Mission	Vehicle	LV Type	Mission Type	LV Results
1997					
February 16	JCSAT 4	AC-127	Atlas IIAS	Supersynchronous	Success
March 8	DBS/TEMPO	AC-128	Atlas IIA	Subsynchronous	Success
April 25	GOES K	AC-79	Atlas I	Supersynchronous	Success
July 27	Superbird C	AC-133	Atlas IIAS	Supersynchronous	Success
September 4	GE-3	AC-146	Atlas IIAS	Supersynchronous	Success
October 5	Echostar III	AC-135	Atlas IIAS	GTO	Success
October 24	DSCS III	AC-131	Atlas IIA	Subsynchronous	Success
December 8	Galaxy 8i	AC-149	Atlas IIAS	Supersynchronous	Success
1998					
January 29	Capricorn MLV-7	AC-109	Atlas IIA	GTO	Success
February 27	INTELSAT 806	AC-151	Atlas IIAS	GTO	Success
March 16	UHF-FO F8	AC-132	Atlas II	Subsynchronous	Success
June 18	INTELSAT 805	AC-153	Atlas IIAS	GTO	Success
October 9	HotBird 5	AC-134	Atlas IIA	GTO	Success
October 20	UHF-FO F9	AC-130	Atlas IIA	Subsynchronous	Success
1999					
February 15	JCSAT-6	AC-152	Atlas IIAS (1N)	Supersynchronous	Success
April 12	Eutelsat W3	AC-154	Atlas IIAS (1N)	Supersynchronous	Success
September 23	EchoStar V	AC-155	Atlas IIAS (1N)	Supersynchronous	Success
November 22	UHF F/O F10	AC-136	Atlas IIA (1)	PVA/ITO	Success
December 18	EOS Terra	AC-141	Atlas IIAS (1N)	LEO/Sunsynchronous	Success
2000					
January 20	MLV-8 DSCS III	AC-138	Atlas IIA (4)	GTO	Success
February 3	Hispasat-1C	AC-158	Atlas IIAS (1N)	GTO	Success
May 3	GOES-L	AC-137	Atlas IIA (1)	GTO	Success
May 24	Eutelsat W4	AC-201	Atlas IIIA	Supersynchronous	Success
June 30	TDRS-H	AC-139	Atlas IIA (1N)	Subsynchronous	Success
July 14	EchoStar VI	AC-161	Atlas IIAS (1N)	Supersynchronous	Success
October 19	MLV-9 DSCS B-11	AC-140	Atlas IIA (4)	GTO	Success
December 5	MLV-11	AC-157	Atlas IIAS (1N)	GTO	Success

APPENDIX B—MISSION SUCCESS AND PRODUCT ASSURANCE

B.1 PRODUCT DELIVERY SYSTEM

Lockheed Martin Astronautics Operations (LMAO) operates an ISO 9001 (International Organization for Standardization 9001:1994, Quality Systems, Model for quality assurance in design, development, production, installation, and servicing) registered quality management system. LMAO is internationally accredited through the British Standards Institute (BSI) under registration number FM 35743. The registration was conferred on December 13, 1996, and includes LMAO's Denver, Colorado; San Diego, California; Harlingen, Texas; Cape Canaveral Air Force Station, and Vandenberg Air Force Base facilities. Scope of the registration includes design, development, test, manufacture, and assembly of advanced technology systems for space and defense, including space systems, launch systems, and ground systems.

Adherence to the ISO 9001 quality standard is revalidated at 6-month intervals by BSI, an independent, third-party registrar. In addition, ISO 9001 compliance is monitored and certified onsite by the U.S. government's Defense Contract Management Agency (DCMA), which also maintains insight into LMAO processes.

ISO 9001 is executed through LMAO's internal policies, practices, and procedures, which are described in the *Product Delivery System Manual* (PDSM). ISO 9001 is a basic quality management system that provides a framework of 20 major elements. The PDSM addresses each element and provides an overview of the element and flowdown references to individual procedures. LMAO also includes mission success as a 21st element because of the deep focus on and commitment to mission success principles.

The PDSM elements and section numbers are:

4.1 Management Responsibility
4.2 Quality System
4.3 Contract Review
4.4 Design Control
4.5 Document and Data Control
4.6 Purchasing
4.7 Control of Customer-Supplied Product
4.8 Product Identification and Traceability
4.9 Process Control
4.10 Inspection and Testing
4.11 Control of Inspection, Measuring, and Test Equipment
4.12 Inspection and Test Status
4.13 Control of Nonconforming Product
4.14 Corrective and Preventive Action
4.15 Handling, Storage, Packaging, Preservation, and Delivery
4.16 Control of Quality Records
4.17 Internal Quality Audits
4.18 Training
4.19 Servicing
4.20 Statistical Techniques
4.21 Mission Success

LMAO is maintaining ISO 9001 registration to the latest released version of the standard.

B.1.1 Mission Success—Independent Oversight

The Mission Success organization provides an evaluation for program and LMAO management independent from Engineering and Product Assurance, to ensure all potential mission impacts at Lockheed Martin and/or suppliers are resolved prior to launch. This is accomplished through their participation in program activities, such as tabletops and reviews, and through the reporting of all functional failures to the Mission Success organization. Mission Success Engineers (MSE) coordinate technical evaluations with appropriate subject-matter experts to establish mission impact. Impact failures are presented to the Space Program Reliability Board (SPRB) to ensure complete analysis and effective corrective action is taken to mitigate mission impact. The SPRB is chaired by the program vice president, co-chaired by the Mission Success organization, and is made up of technical experts who ensure complete investigation and resolution of all hardware concerns potentially affecting mission success. Mission Success is tasked to ensure all applicable flight constraints are resolved before launch.

Alerts—The Government-Industry Data Exchange Program (GIDEP) review process includes evaluating GIDEP alerts for impacts to hardware. All known alerts will be reviewed for impacts before each flight.

Reviews—The Mission Success organization participates in engineering, factory, data, and readiness reviews to support program management in determining hardware flight worthiness and readiness. Figure B.1.1-1 is a flow of the acceptance process, including progressive reviews and acceptance throughout design, production, test, and launch site operations.

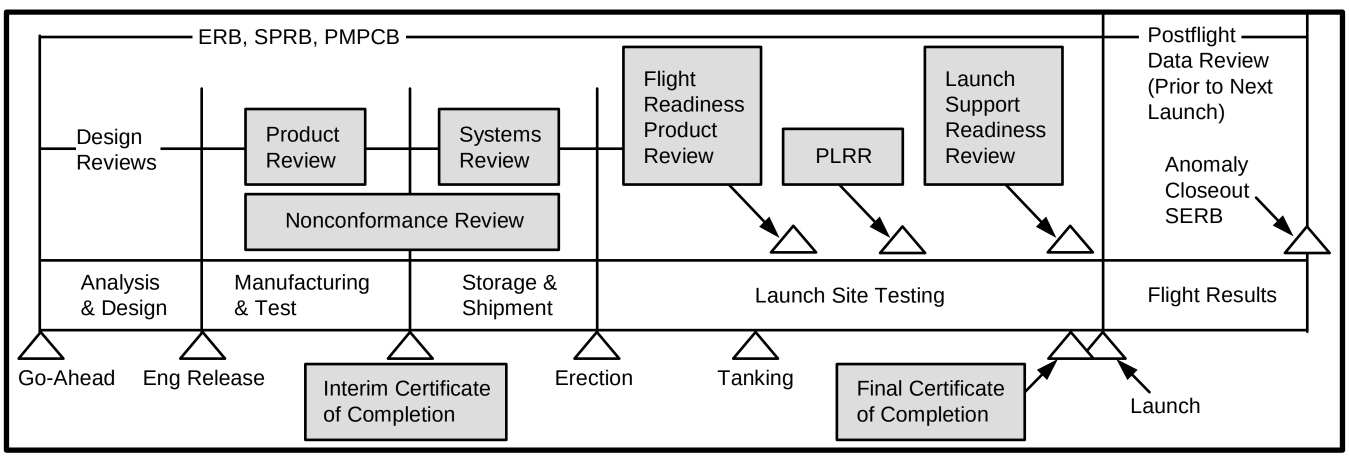


Figure B.1.1-1 Mission Success Acceptance Process

B.1.2 Product Assurance

Product Assurance (PA) ensures the quality of products and processes to achieve maximum effectiveness and continued stakeholder confidence.

Quality is ensured through physical examination, measuring, test, process monitoring, and/or other methods as required to determine and control the quality of all deliverable airborne and ground equipment products and services. Independent verification includes mandatory inspection points; witnessing, monitoring, and surveillance; statistical methods; data analysis; trending; sampling plans; work instruction, build, and test documentation review; associated data reviews; process control; audit; acceptance tooling; and/or other techniques suited to the products or processes being verified. Responsibilities are defined in the following paragraphs.

Audit—Documented procedures are established and maintained for planning, performing, reporting, and follow-up of internal audits. The Astronautics Operations Internal Audit Program coordinates with other audit organizations including corporate internal audit and customer audit teams to maximize effectiveness.

The Management System Assessment organization has responsibility for management and oversight of the Internal Audit Program. Auditors are independent of the person responsible for taking corrective action.

Corrective/Preventative Action—The corrective and preventative action process (Fig. B.1.1-2) ensures visibility and resolution of anomalous conditions affecting or potentially affecting products, processes, or systems. This process encompasses customer concerns, internal activities, and supplier or subcontractor issues. The corrective and preventative action process ensures that problems are identified and documented, root cause is determined and recorded, corrective action is identified and reviewed for appropriateness, and corrective action is implemented and verified for effectiveness.

Design Reviews—Product Assurance participates at an appropriate level in conceptual, preliminary, critical, and drawing-level design reviews. The design review activity provides requirements flowdown, critical characteristics identification, acceptance strategy development, and method analysis. This ensures inspectability and acceptability of detailed drawings and specifications, and promotes producibility (Fig. B.1.1-1).

Parts, Materials, and Processes (PMP) Control Board (PMPCB)—The PMPCB provides the method for ensuring the use of proven parts, materials, and processes across the program. The board is the primary channel of communication for the interchange of PMP information analyses. The PMPCB

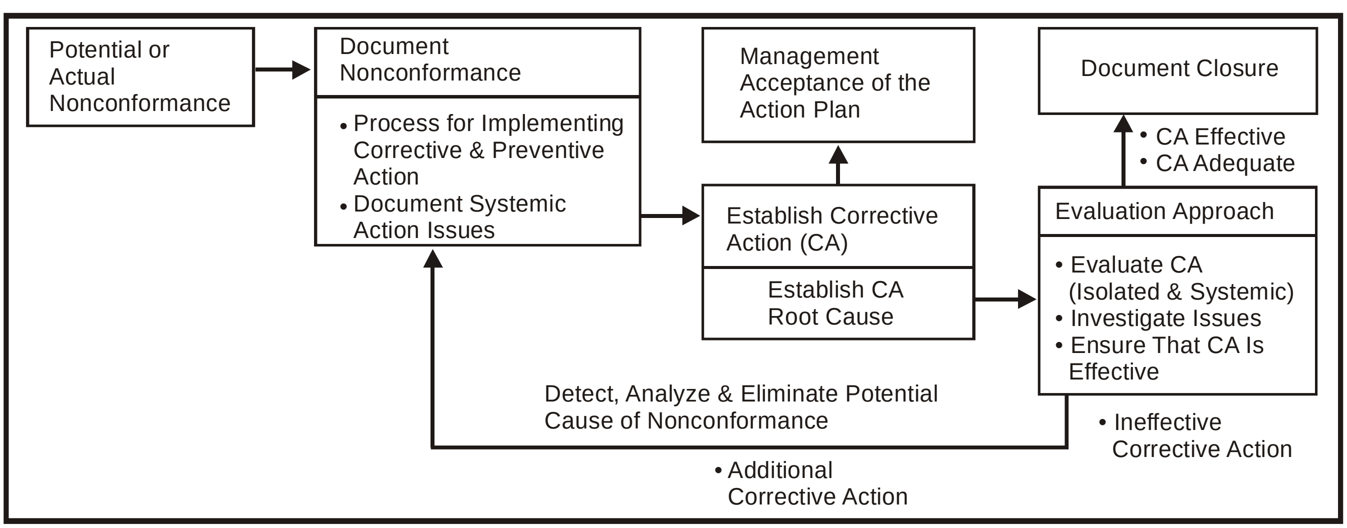


Figure B.1.1-2 Corrective and Preventive Action Process provides direction for procurement activities to support program schedules. This includes direction to order parts, assign priorities, and providence of device lot failure mitigation.

Change Control—All engineering changes are managed and approved by an engineering review board (ERB) and implemented by the Change Control Board. All preliminary changes are presented to a combined product support team (PST) and change integration board (CIB). This multidiscipline team develops the detailed scope of the change and provides inputs for scheduling, process planning, material planning, and configuration management. The PST and CIB provide status of each change through final engineering release and coordinate release with the product schedule and delivery needs. Product Assurance Engineering participates in this process, ensuring each change yields inspectable and acceptable products of known and controlled configuration.

Work Instructions—All work affecting quality is prescribed in documented instructions of a type appropriate to the circumstances. Work instructions encompass purchasing, handling, machining, assembling, fabricating, processing, inspecting, testing, modifying, installing, and any other treatment of product, facilities, standards, or equipment. Preparations and maintenance of work instructions and manufacturing processes are monitored as a function of the quality program.

Supplier Quality—A supplier rating system is maintained to monitor the performance of suppliers. Hardware is only procured from approved suppliers with acceptable ratings. An assessment of hardware criticality and supplier performance is performed to ensure the quality of supplier-built hardware. Based on this assessment, quality source representatives are assigned to certain suppliers and critical hardware to ensure hardware conformance to requirements during suppliers' operation through shipment.

Acceptance Status—Objective evidence of compliance to requirements of contract, specifications, configuration, and process control is maintained and made available to hardware acceptance teams at all manufacturing and test facilities. Product Assurance provides a positive system for identification of the inspection and acceptance status of products. This is accomplished by stamping, tags, routing cards, move tickets, build records, and/or other control devices. The overall product acceptance program is controlled by adherence to a program Product Acceptance Plan which details acceptance processes at Harlingen, San Diego, Denver, and suppliers.

Identification and Stamp Control—Inspection, fabrication, workmanship (including physical or electronic stamps), signatures, acceptance, and status markings are controlled and traceable to the individuals performing those functions.

Nonconforming Hardware—When material is initially found to be nonconforming, it is examined by preliminary material review (PMR) certified personnel. Assistance by inspection, manufacturing, and engineering personnel is often necessary to determine if the nonconformance can be eliminated through rework, scrapping, or by returning hardware to the supplier. If none of these criteria can be met, the material is referred to the Material Review Board (MRB) for disposition. Product Assurance ensures that all nonconforming material is identified and controlled to preclude its subsequent use in deliverable items without proper disposition. Quality Engineering chairs the MRB to determine appropriate disposition of nonconforming material. The board includes a representative from the Engineering organization, who is responsible for product design. All MRB members are required to be certified and approved by program Product Assurance.

Software Quality—The Software Quality program is designed to ensure software products (code and documentation) are compliant with program, corporate, and customer requirements. Specifically, the program provides the methods necessary to:

1) Ensure approved processes and procedures are followed in design, development and testing of software products;

2) Ensure requirements are clear, quantifiable, and testable;
3) Ensure subcontracted software is developed according to approved processes and procedures;
4) Ensure software anomaly reports (SAR) are tracked to closure and only approved anomalies are incorporated into controlled baselines.

The software development process is designed to build quality into the software and documentation and to maintain levels of quality throughout the life cycle of the software. This includes independent technical evaluations, software testing, documentation verification, and management reviews necessary to achieve this goal.

Record Retention—All Product Assurance records are retained as required by contract, policy and procedures, and specific program direction. A secure product assurance data center is used as the central repository for Atlas quality related data.

Training and Certification—The Certification Board is responsible for ensuring integrity in product development, test, and operations. The board ensures that personnel requiring special skills in fabrication, handling, test, maintenance, operations, and inspection of products have been trained and are qualified to ensure their capability to perform critical functions. Board responsibilities include certification of individuals and crews and oversight of offsite certification boards.

Metrics—Metrics are maintained to provide a continuous assessment of program performance, to control and/or reduce program costs, to ensure continued mission success, and to drive the program toward improvement. Examples of metrics maintained are:

1) Nonconformances per 1,000 hours,
2) MRBs per 1,000 hours,
3) Escapements,
4) Recurring nonconformances,
5) Supplier liability,
6) Foreign object incidents,
7) Aging of nonconformance documents.

Calibration—A calibration system is maintained and documented to ensure that supplies and services presented for acceptance conform with prescribed technical requirements. This system applies to adequacy of standards, environmental control, intervals of calibration, procedures, out-of-tolerance evaluation, statuses, sources, application and records, control of subcontractor calibrations, and storage and handling.

Acceptance Tooling—When production jigs, fixtures, tooling masters, templates, patterns, test software, and other devices are used as a method for acceptance, they are proven for accuracy before release and at subsequently periodic intervals to ensure that their accuracies meet or exceed product requirements.

B.1.3 Launch Site Product Assurance Operations

Launch Operations Product Assurance supports launch sites during launch vehicle processing, ground systems maintenance and installations, and checkout of modifications.

Product Assurance is the focal point for coordination of facility issues with Vandenberg Air Force Base and Cape Canaveral Air Force Station Product Assurance personnel. In cases where products are to be shipped with open engineering or open work, Product Assurance provides the necessary coordination with site personnel. This single-point contact between Denver and the sites ensures clear communication and provides a filter to minimize open items.

Launch processing oversight is conducted by the assignment of a vehicle-peculiar mission success engineering specialist. The engineering specialist ensures that vehicle noncompliance issues have resolution and acts as the customer coordinator on vehicle certificates of conformance and flight constraint closeouts. This specialist follows the assigned vehicle throughout the production and launch sequence and reports on vehicle readiness in a series of program reviews (Fig. B.1.1-1).

All launch site operations are an extension of factory operations and are covered by the same requirements for reporting, control, and problem resolution.

B.1.3.1 Payload Operations—Payload Operations is responsible for coordination and implementation of integrated spacecraft and launch vehicle tasks. The system engineer is responsible for integration activities, ensuring compliance to specifications and customer direction.

For launch vehicle activities, Payload Operations responds as the customer’s agent in coordinating integrated activities and providing support as requested by the customer. Lockheed Martin personnel at the launch site participate in the total quality system process (oversight, engineering, and inspection) during payload integration at commercial and government processing facilities. This effort is inclusive of mating to payload adapters, encapsulation, radio frequency (RF) checkout, integration testing, and transportation to the launch pad.

B.1.3.2 Safety Operations—Safety operations include system safety engineering efforts and launch site operations safety support. To assure safety of personnel and hardware, our Safety Operations Group provides onsite support from hardware receipt through launch. Operational support includes consultation/coordination of industrial, material handling, fire protection, and environmental/chemical safety requirements; enforcement of Lockheed Martin as well as Range Safety policy; and oversight of selected activities.

System Safety reviews spacecraft designs, assesses launch vehicle/spacecraft interfaces, and evaluates mission-specific ground processing operations to identify accident risks and develop the appropriate hazard controls. System Safety is responsible for coordination of technical issues with the $45^{th}/30^{th}$ Space Wing and timely completion of the Range Safety review and approval process (refer to Section 4.3).

APPENDIX C—SPACECRAFT DATA REQUIREMENTS

The items listed in this appendix are representative of the information required for spacecraft integration and launch activities. Additional information may be required for specific spacecraft.

C.1 INTERFACE CONTROL DOCUMENT INPUTS

Table C.1-1 indicates the spacecraft information required to assess the spacecraft's compatibility with the Atlas launch vehicle. Data usually are provided by the customer in the form of an interface requirements document (IRD) and are the basis for preparing the interface control document (ICD). Shaded items should be provided for a preliminary compatibility assessment, while all items should be completed for a detailed assessment. The shaded items are typically supplied for the spacecraft before a proposal is offered for Atlas Launch Services. These lists are generalized and apply to any candidate mission. If Lockheed Martin has experience with the satellite bus or satellite contractor, less information can be provided initially (assuming the spacecraft contractor is willing to use a "same as mission _____" designation for purposes of assessing preliminary compatibility. A complete IRD is typically supplied within 30 days of contract signing.

Tables C.1-2 through C.1-7 indicate spacecraft data required after contract signature to start integration of the spacecraft. The asterisks in these tables indicate data desired at an initial meeting between Lockheed Martin and the customer. These data will provide the detailed information required to fully integrate the spacecraft, determine such items as optimum mission trajectory, and verify compatibility of launch vehicle environments and interfaces.

Table C.1-1 Spacecraft Information Worksheet

For a Preliminary Compatibility Assessment, All Shaded Items Should Be Completed. For a Detailed Compatibility Assessment, All Items Should Be Completed.

Spacecraft Name:	Spacecraft Manufacturer:
Spacecraft Owner:	Spacecraft Model No.:
Name of Principal Contact:	Number of Launches:
Telephone Number:	Date of Launches:
Date:	

Spacecraft Design Parameter	**SI Units**	**English Units**
TRAJECTORY REQUIREMENTS		
Satellite Mass	______ kg	______ lbm
Minimum Satellite Lifetime	______ yr	______ yr
Final Orbit Apogee	______ km	______ km
Final Orbit Perigee	______ km	______ km
Final Orbit Inclination	______ deg	______ deg
Propulsion-Propellant Type, Orbit Insertion		
Propulsion-Propellant Type, Stationkeeping		
Propulsion-Multiple Burn Capability (Y/N)		
Propulsion-Propellant Mass	______ kg	______ lbm
Propulsion-Effective I_{sp}	______ s	______ s
Maximum Apogee Allowable	______ km	______ nmi
Minimum Perigee Allowable	______ km	______ nmi
Argument of Perigee Requirement	______ deg	______ deg
Right Ascension of Ascending Node Requirement	______ deg	______ deg
Apogee Accuracy Requirement	______ km	______ km
Perigee Accuracy Requirement	______ km	______ km
Inclination Accuracy Requirement	______ deg	______ deg
Argument of Perigee Accuracy Requirement	______ deg	______ deg
Right Ascension of Ascending Node Accuracy Requirement	______ deg	______ deg

Table C.1-1 (cont)

Spacecraft Design Parameter	SI Units	English Units
MECHANICAL INTERFACE		
Spacecraft Mechanical Drawing (Launch Configuration)		
Spacecraft Effective Diameter	______ mm	______ in.
Spacecraft Height	______ mm	______ in.
Spacecraft/Launch Vehicle Interface Diameter	______ mm	______ in.
Payload Sep System Supplier (Spacecraft or Launch Veh)		
Payload Adapter Supplier (Spacecraft or Launch Vehicle)		
Maximum Spacecraft Cross-Sectional Area	______ m^2	______ ft^2
Number & Size Payload Fairing Access Doors	______ mm x mm	______ in. x in.
Preseparation RF Transmission Requirement	______ band	______ band
ELECTRICAL INTERFACE		
Spacecraft Electrical Drawing		
Number of Launch Vehicle Signals Required		
Number of Separation Discretes Required		
Number of Umbilicals & Pins/Umbilical		
Curve of Spacecraft-Induced Electric Field Radiated Emissions	______ dBµ V/m	______ MHz
Curve of Spacecraft-Radiated Susceptibility	______ dBµ V/m	______ MHz
Number of Instrumentation Analogs Required		
THERMAL ENVIRONMENT		
Prelaunch Ground Transport Temperature Range	______ °C	______ °F
Prelaunch Launch Pad Temperature Range	______ °C	______ °F
Maximum Prelaunch Gas Impingement Velocity	______ m/s	______ ft/s
Maximum Ascent Heat Flux	______ W/m^2	______ $Btu/hr\text{-}ft^2$
Maximum Free-Molecular Heat Flux	______ W/m^2	______ $Btu/hr\text{-}ft^2$
Maximum Fairing Ascent Depressurization Rate	______ mbar/s	______ psi/s
Spacecraft Vented Volume(s)	______ m^3	______ ft^3
Spacecraft Vent Area(s)	______ cm^2	______ in^2
Prelaunch Relative Humidity Range	______ %	______ %
Preseparation Spacecraft Power Dissipation	______ W	______ Btu/hr
Maximum Free-Stream Dynamic Pressure	______ mbar	______ psi
DYNAMIC ENVIRONMENT		
Maximum Allowable Flight Acoustics	______ dB OA	______ dB OA
Allowable Acoustics Curve		
Maximum Allowable Sine Vibration	______ G_{RMS}	______ G_{RMS}
Allowable Sine Vibration Curve	______ G_{RMS}	______ G_{RMS}
Maximum Allowable Shock	______ g	______ g
Allowable Shock Curve		
Maximum Acceleration (Static + Dynamic) Lateral	______ g	______ g
Maximum Acceleration (Static + Dynamic) Longitudinal	______ g	______ g
Fundamental Natural Frequency—Lateral	______ Hz	______ Hz
Fundamental Natural Frequency—Longitudinal	______ Hz	______ Hz
cg—Thrust Axis (Origin at Separation Plane)	______ mm	______ in.
cg—Y Axis	______ mm	______ in.
cg—Z Axis	______ mm	______ in.
cg Tolerance—Thrust Axis	______ ±mm	______ ±in.
cg Tolerance—Y Axis	______ ±mm	______ ±in.
cg Tolerance—Z Axis	______ ±mm	______ ±in.
Fundamental Natural Frequency—Longitudinal	______ Hz	______ Hz
cg—Thrust Axis (Origin at Separation Plane)	______ mm	______ in.

Table C.1-1 (concl)

Spacecraft Design Parameter	SI Units	English Units
CONTAMINATION REQUIREMENTS		
Fairing Air Cleanliness	______ Class	______ Class
Maximum Deposition on Spacecraft Surfaces	______ mg/m^2	______ mg/m^2
Outgassing—Total Weight Loss	______ %	______ %
Outgassing—Volatile Condensable Material Weight Loss	______ %	______ %
SPACECRAFT DESIGN SAFETY FACTORS		
Airborne Pressure Vessel Burst Safety Factor		
Airborne Pressure System Burst Safety Factor		
Structural Limit (Yield) Safety Factor		
Structural Ultimate Safety Factor		
Battery Burst Safety Factor		
SPACECRAFT QUALIFICATION TEST PROGRAM		
Acoustic Qualification	______ +dB	______ +dB
Sine Vibration Qualification Safety Factor		
Shock Qualification Safety Factor		
Loads Qualification Safety Factor		
ORBIT INJECTION CONDITIONS		
Range of Separation Velocity	______ m/s	______ ft/s
Max Angular Rate at Separation-Roll	______ rpm	______ rpm
Max Angular Rate Uncertainty-Roll	______ ±rpm	______ ±rpm
Max Angular Rate at Separation—Pitch & Yaw	______ rpm	______ rpm
Max Angular Rate Uncertainty—Pitch & Yaw	______ ±rpm	______ ±rpm
Max Angular Acceleration	______ rad/s^2	______ rad/s^2
Max Pointing Error Requirement	______ deg	______ deg
Max Allowable Tip-Off Rate	______ deg/s	______ deg/s
Coefficients of Inertia—Ixx (x=Thrust Axis)	______ kg m^2	______ slug ft^2
Coefficients of Inertia—Ixx Tolerance	______ ±kg m^2	______ ±slug ft^2
Coefficients of Inertia—Iyy	______ kg m^2	______ slug ft^2
Coefficients of Inertia—Iyy Tolerance	______ ±kg m^2	______ ±slug ft^2
Coefficients of Inertia—Izz	______ kg m^2	______ slug ft^2
Coefficients of Inertia—Izz Tolerance	______ ±kg m^2	______ ±slug ft^2
Coefficients of Inertia—Ixy	______ kg m^2	______ slug ft^2
Coefficients of Inertia—Ixy Tolerance	______ ±kg m^2	______ ±slug ft^2
Coefficients of Inertia—Iyz	______ kg m^2	______ slug ft^2
Coefficients of Inertia—Iyz Tolerance	______ ±kg m^2	______ ±slug ft^2
Coefficients of Inertia—Ixz	______ kg m^2	______ slug ft^2
Coefficients of Inertia—Ixz Tolerance	______ ±kg m^2	______ ±slug ft^2

Table C.1-2 Mission Requirements

Type of Data	Scope of Data
Number of Launches*	
Frequency of Launches* Spacecraft Orbit Parameters Including Tolerances (Park Orbit, Transfer Orbit)*	• Apogee Altitude • Perigee Altitude • Inclination • Argument of Perigee • RAAN
Launch Window Constraints • Preseparation Function*	• Prearm • Arm • Spacecraft Equipment Deployment Timing & Constraints • Acceleration Constraints (Pitch, Yaw, Roll) • Attitude Constraints • Spinup Requirements
Separation Parameters (Including Tolerances)*	• Desired Spin Axis • Angular Rate of Spacecraft • Orientation (Pitch, Yaw & Roll Axis) • Acceleration Constraints
Any Special Trajectory Requirements	• Boost Phase • Coast Phase • Free Molecular Heating Constraints • Thermal Maneuvers • Separation Within View of Tlm & Tracking Ground Station • Telemetry Dipout Maneuvers
Note: * Information Desired at Initial Meeting Between Lockheed Martin & Customer after Contract Award	

Table C.1-3 Spacecraft Characteristics

Type of Data	Scope of Data
Configuration Drawings* Apogee Kick Motor*	• Drawings Showing the Configuration, Shape, Dimensions & Protrusions into the Mounting • Adapter (Ground Launch & Deployment Configurations) • Coordinates (Spacecraft Relative to Launch Vehicle) • Special Clearance Requirements • Manufacturer's Designation • Thrust • Specific Impulse • Burn Action Time • Propellant Offload Limit
Mass Properties (Launch & Orbit Configurations)*	• Weight—Specify Total, Separable & Retained Masses • Center of Gravity—Specify in 3 Orthogonal Coordinates Parallel to the Booster Roll, Pitch & Yaw Axes for Total, Separable & Retained Masses • Changes in cg Due to Deployment of Appendages • Propellant Slosh Models
Moments of Inertia (Launch & Orbit Configurations)	• Specify About the Axes Through the Spacecraft cg That Are Parallel to the Atlas Roll, Pitch & Yaw Axes for Total, Separable, & Retained Masses
Structural Characteristics	• Spring Ratio of Structure, Elastic Deflection Constants, Shear Stiffness, Dynamic Model, Bending Moments & Shear Loads at Atlas/Centaur/Spacecraft Interface & Limitations to Include Acoustic, Shock, Acceleration, Temperature & Bending Moments

Table C.1-3 (cont)

Type of Data	Scope of Data
Dynamic Model for 3-D Loads Analysis	• Generalized Stiffness Matrix (Ref Paragraph C.3.2 for Details) • Generalized Mass Matrix • Description of the Model, Geometry & Coordinate System • Loads Transformation Matrix • Note: Models Must Include Rigid Body & Normal Modes
Handling Constraints	• Spacecraft Orientation During Ground Transport • Spacecraft Handling Limits (e.g., Acceleration Constraints)
Spacecraft Critical Orientation During:	• Location & Direction of Antennas Checkout, Prelaunch & Orbit • Location, Look Angle & Frequency of Sensors • Location & Size of Solar Arrays
Safety Items	• General Systems Description • Basic Spacecraft Mission • Prelaunch Through Launch Configuration • Orbital Parameters • Functional Subsystems • Hazardous Subsystems • Ground Operations Flow • Flight Hardware Descriptions (Safety-Oriented) • Structural/Mechanical Subsystems • Propellant/Propulsion Subsystems • Pressurized Subsystems • Ordnance Subsystems • Electrical & Electronic Subsystems • Nonionizing Radiation Subsystems (RF/Laser) • Ionizing Radiation Subsystems • Hazardous Materials
	• Thermal Control Subsystems • Acoustical Subsystems • Note: Hazard Identification/Controls/Verification Method Summaries for Each Subsystem • GSE Descriptions • Mechanical GSE • Propellant/Propulsion GSE • Pressure GSE • Ordnance GSE • Electrical GSE • RF/Laser GSE • Ionizing Radiation GSE • Hazardous Materials GSE • Note: Hazard Identification/Controls/Verification Method Summaries for Each Item • Ground Operations • Hazardous Ground Operations • Procedures • Transport Configuration
Note: For Each Data Submittal—Identify Each Item/Operation Applicable to PPF, HPF, or Launch Site	

Table C.1-3 (concl)

Type of Data	Scope of Data
Thermal Characteristics	• Spacecraft Thermal Math Model (Ref Sect. C.3.3) • Emissivity • Conductivity • Resistivity • Thermal Constraints (Maximum & Minimum Allowable Temperature) • Heat Generation (e.g., Sources, Heat, Time of Operation)
Contamination Control	• Requirements for Ground-Supplied Services • In-Flight Conditions (e.g., During Ascent & After PLF Jettison) • Surface Sensitivity (e.g., Susceptibility to Propellants, Gases, & Exhaust Products)
RF Radiation	• Characteristics (e.g., Power Levels, Frequency, & Duration for Checkout & Flight Configuration) • Locations (e.g., Location of Receivers & Transmitters on Spacecraft) • Checkout Requirements (e.g., Open-Loop, Closed-Loop, Prelaunch, Ascent Phase)
Note: * Information Desired at Initial Meeting Between Lockheed Martin & Customer After Contract Award	

Table C.1-4 Aerospace Vehicle Equipment (AVE) Requirements (Mechanical)

Type of Data	Scope of Data
Mechanical Interfaces *	• Base Diameter of Spacecraft Interface* • Structural Attachments at Spacecraft Interface* • Required Accessibility to Spacecraft in Mated Condition* • Extent of Equipment Remaining with Adapter After Spacecraft Separation* • Degree of Environmental Control Required • Spacecraft Pressurization, Fueling System Connector Type & Location; Timeline for Pressure/Fuel System Operation • Spacecraft/Adapter Venting Requirements
PLF Requirements	• Heating Constraints • Venting Characteristics (e.g., Quantity, Timing & Nature of Gases Vented from P/L) • RF Reradiation System (RF Band, Spacecraft Antenna Location, etc) • PLF Separation (e.g., Altitude, Cleanliness, Shock, Aeroheating & Airload Constraints) • Acoustic Environment Constraints • Special Environmental Requirements
Preflight Environment	• PLF Separation (e.g., Altitude, Cleanliness, Shock, Aeroheating & Airload Constraints) • Acoustic Environment Constraints • Special Environmental Requirements • Requirements – Cleanliness – Temperature & Relative Humidity – Air Conditioning – Air Impingement Limits • Monitoring & Verification Requirements
Umbilical Requirements	• Separation from Launch Vehicle • Flyaway at Launch • Manual Disconnect (Including When)
Materials	• Special Compatibility Requirements • Outgassing Requirements
Note: * Information Desired at Initial Meeting Between Lockheed Martin & Customer After Contract Award	

Table C.1-5 Aerospace Vehicle Equipment (AVE) Requirements (Electrical)

Type of Data	Scope of Data
Power Rqmts (Current, Duration, Function Time & Tolerances)*	• 28-Vdc Power • Other Power • Overcurrent Protection
Command Discrete Signals*	• Number* • Sequence • Timing (Including Duration, Tolerance, Repetition Rate, etc) • Voltage (Nominal & Tolerance) • Frequency (Nominal & Tolerance) • Current (Nominal & Tolerance) • When Discretes Are for EED Activation, Specify Minimum, Maximum & Nominal Fire Current; Minimum & Maximum Resistance; Minimum Fire Time; Operating Temperature Range & Manufacturer's Identification of Device
Other Command & Status Signals	• Status Displays • Abort Signals • Range Safety Destruct • Inadvertent Separation Destruct
Ordnance Circuits	• Safe/Arm Requirements
Telemetry Requirements*	• Quantity of Spacecraft Measurements Required To Be Transmitted by Atlas Telemetry & Type (e.g., Temperature, Vibration, Pressure, etc); Details Concerned with Related System Including Operating Characteristics (Response Definition of System) & Locations & Anticipated Time of Operation • Impedance, Capacitance, Operating Range & Full-Scale Range of Each Measurement • Signal Conditioning Requirements (e.g., Input Impedance, Impedance Circuit Load Limits, Overcurrent Protection & Signal-to-Noise Ratio) • Discrete Events (Bilevel) • Analog Measurements • Transducers Required To Be Furnished by LV Contractor • Minimum Acceptable Frequency Response for Each Measurement • Minimum Acceptable System Error for Each Measurement (Sampling Rate Is Also Governed by This Requirement) • Period of Flight for Which Data from Each Measurement Are of Interest (e.g., from Liftoff to Space Vehicle Separation)* • Atlas Flight Data Required by Spacecraft Contractor
Bonding	• Bonding Requirements at Interface (MIL-B-5087, Class R for LV) • Material & Finishes at Interface (for Compatibility with LV Adapter)
EMC	• Test or Analyze Spacecraft Emissions & Susceptibility • EMC Protection Philosophy for Low-Power, High-Power & Pyrotechnic Circuits • LV & Site Emissions (Provided by Lockheed Martin)
Grounding Philosophy	• Structure (e.g., Use of Structural As Ground & Current Levels) • Electrical Equipment (e.g., Grounding Method for Signals & Power Supplies) • Single-Point Ground (e.g., Location & Related Equipment)
Interface Connectors*	• *Connector Item (e.g., Location & Function) • Connector Details • Electrical Characteristics of Signal on Each Pin
Shielding Requirements	• Each Conductor or Pair • Overall • Grounding Locations for Termination
Note: *Desired for Initial Integration Meeting with Lockheed Martin After Contract Award	

Table C.1-6 Test Operations

Type of Data	Scope of Data
Spacecraft Launch Vehicle Integration	• Sequence from Spacecraft Delivery Through Mating with the LV • Handling Equipment Required • Lockheed Martin-Provided Protective Covers or Work Shields Required • Identify the Space Envelope, Installation, Clearance, & Work Area Requirements • Any Special Encapsulation Requirements Support Services Required
Spacecraft Checkout AGE & Cabinet Data	• List of All AGE & Location Where Used (e.g., Storage Requirements on the Launch Pad) • Installation Criteria for AGE Items: – Size & Weight – Mounting Provisions – Grounding & Bonding Requirements – Proximity to the Spacecraft When In Use – Period of Use – Environmental Requirements – Compatibility with Range Safety Requirements & LV Propellants – Access Space to Cabinets Required for Work Area, Door Swing, Slideout Panels, etc – Cable Entry Provisions & Terminal Board Types in Cabinets &/or Interface Receptacle Locations & Types – Power Requirements & Characteristics of Power for Each Cabinet
Spacecraft Environmental Protection (Preflight)	• Environmental Protection Requirements by Area, Including Cleanliness Requirements: – Spacecraft Room – Transport to Launch Pad – Mating – Inside PLF – During Countdown • Air-Conditioning Requirements for Applicable Area (Pad Area) by: – Temperature Range – Humidity Range – Particle Limitation – Impingement Velocity Limit – Flow Rate • Indicate if Space Veh Is Not Compatible with LV Propellants & What Safety Measures Will Be Reqd • Environmental Monitoring & Verification Requirements
Space Access Requirements	• Access for Space Vehicle Mating & Checkout • Access During Transportation to the Launch Pad & Erection Onto the Atlas • Access for Checkout & Achieving Readiness Prior to Fairing Installation • Access After Fairing Installation; State Location, Size of Opening & Inside Reach Required • Access During the Final Countdown, if Any • AGE Requirements for Emergency Removal
Umbilicals	• Ground Servicing Umbilicals by Function & Location in Excess of Atlas/Centaur Baseline • Structural Support Requirements & Retraction Mechanisms • Installation (e.g., When & by Whom Supplied & Installed)
Commodities Required for Both Spacecraft, AGE & Personnel	• Gases, Propellants, Chilled Water & Cryogenics in Compl with Ozone-Depleting Chemicals Rqmts • Source (e.g., Spacecraft or LV) • Commodities for Personnel (e.g., Work Areas, Desks, Phones)
Miscellaneous	Spacecraft Guidance Alignment Requirements

Table C.1-6 (concl)

Type of Data	Scope of Data
Hardware Needs (Including Dates)*	• Electrical Simulators • Structural Simulators • *Master Drill Gage
Interface Test Requirements	• Structural Test • Fit Test • Compatibility Testing of Interfaces (Functional) • EMC Demonstration • LV/Spacecraft RF Interface Test • Environmental Demonstration Test
Launch Operations	• Detailed Sequence & Time Span of All Spacecraft-Related Launch Site Activities Including: AGE Installation, Facility Installation & Activities, Spacecraft Testing & Spacecraft Servicing • Recycle Requirements • Restrictions To Include Launch Site Activity Limitations, Constraints on Launch Vehicle Operations, Security Requirements & Personnel Access Limitations & Safety Precautions • Special Requirements Include Handling of Radioactive Materials, Security & Access Control • Support Requirements To Include Personnel, Communications & Data Reduction • Launch & Flight Requirements for Real-Time Data Readout, Postflight Data Analysis, Data Distribution, Postflight Facilities
Note: * Desired for Initial Integration Meeting with Lockheed Martin After Contract Award	

Table C.1-7 AGE/Facility Requirements (Electrical)

Type of Data	Scope of Data
Space Vehicle Electrical Conductor Data	S/C System Schematic Showing All Connectors Required Between S/C Equipment, & S/C Terminal Board Position or Receptacle Pin Assigned to Each Conductor; Electrical Characteristics of Each Connector Including Maximum End-to-End Resistance, Shielding, Capacitance & Spare Conductors
Electrical Power (AGE & Facility)	• Frequency, Voltage, Watts, Tolerance, Source • Isolation Requirements • Identify if Values Are Steady or Peak Loads • High-Voltage Transient Susceptibility
RF Transmission	• Antenna Requirements (e.g., Function, Location, Physical Characteristics, Beam Width & Direction & Line-of-Sight) • Frequency & Power Transmission • Operation
Cabling	• Any Cabling, Ducting, or Conduits To Be Installed in the Mobile Service Tower; Who Will Supply, Install, Checkout & Remove
Monitors & Controls	• Specify Which Signals from S/C Are To Be Monitored During Readiness & Countdown; Power Source (S/C, Atlas, Centaur) • Transmission Method (e.g., S/C Tlm, LV Tlm, Landline, or LV Readiness Monitor) • Location of Data Evaluation Center, Evaluation Responsibility, Measurement Limits & Go/No-Go Constraints; Identify Where in the Operational Sequence Measurements Are To Be Monitored & Evaluated; Specify Frequency & Duration of Measurements • Video Output Characteristics of Telepaks (if Available) for Closed-Loop Prelaunch Checkout at the Launch Pad; Data To Include Location & Type of Interface Connector(s), & Characteristics of Signal at Source; This Includes Voltage Level, Output Impedance, Output Current Limitation, Maximum Frequency of Data Train & Output Loading Requirements

C.2 SPACECRAFT DESIGN REQUIREMENTS

Table C.2-1 lists specific requirements that should be certified by analysis and/or test by the spacecraft agency to be compatible for launch with Atlas/Centaur. Lockheed Martin will work with the customer to resolve the incompatibility should the spacecraft not meet any of these requirements.

Table C.2-1 Spacecraft Design Elements to Be Certified by Analysis or Test

Spacecraft Design Requirement	Comment
Mechanical	
PLF Envelope (Appendix D)	
P/L Adapter Envelope (Appendix D)	
P/L Adapter Interface (Appendix E)	
Electrical	
• Two or Fewer Separation Commands (Section 4.1.3)	
• 16 or Fewer Control Commands (28-V Discretes or Dry Loop) (Section 4.1.3)	
• Instrumentation I/F, 2 or Fewer Inputs for S/C Separation Detection, 4 or Fewer Analog Inputs for General Use; 10 or Fewer Cmd Feedback Discretes, 2 or Fewer Serial Data I/F for Downlinking S/C Data (Section 4.1.3)	
• Two Umbilical Connectors at S/C Interface (Figure 4.1.3-1)	
Structure & Loads	
• Design Load Factors (Tables 3.2.1.1-1 and 3.2.1.1-2)	
• First Lateral Modes Above 10 Hz & First Axial Mode Above 15 Hz (Section 3.2.1.1)	
• S/C Mass vs cg Range (Appendix E)	
• Design FS per Applicable Range Safety Documentation & MIL-STD-1522 (or Submit Deviations for Review)	
Environment	
• S/C Test Requirements (Section 3.3)	
• Quasi-Sinusoidal Vibration (Figures 3.2.3-1 Through 3.2.3-3)	
• Acoustic Levels in the PLF (Figures 3.2.2-1 Through 3.2.2-5)	
• Shock Induced by PLF Jettison & P/L Separation (Figure 3.2.4-1)	
• P/L Compartment Pressures & Depressurization Rates (Figures 3.2.6-1 Through 3.2.6-4)	
• Gas Velocity Across S/C Components < 6.1 m/s (20 ft/s)	For Medium Fill S/C—4-m PLF
• Gas Velocity Across S/C Components 9.75 m/s (32 ft/s)	For Large Fill S/C—4-m PLF
• Gas Velocity Across S/C Components 10.67 m/s (35 ft/s)	5-m PLF
• Electric Fields (Figures 3.1.2.1-1 Through 3.1.2.2-3)	
• S/C Radiation Limit (Figure 3.1.2.4-1)	
• EM Environment at Launch Range TOR-95(5663)-1 & Section 3.1.2.3	
Safety	
• All S/C Propellant Fill & Drain Valves & All Pressurant Fill & Vent Valves Readily Accessible When S/C Is Fully Assembled & Serviced in Launch Configuration (Encapsulated & on Pad)	It Is Advisable To Accommodate Normal Servicing/Deservicing & Potential Emergency Backout Situation for New Spacecraft Design
• Requirements in Range Safety Regulation	
Miscellaneous	
• See Atlas Launch Services Facilities Guide for S/C Propellants & Specifications Available at Launch Site Fuel Storage Depot	
Note: Compliance with Ozone-Depleting Chemicals Regulation Is Required	

C.3 SPACECRAFT INTEGRATION INPUTS

Table C.3-1 provides a list of typical spacecraft inputs required for the integration process, the approximate need date, and a brief description of the contents. Further details on some items are provided in the following sections.

Table C.3-1 Spacecraft Inputs to Integration Process

Spacecraft Data Input	Typical Need Date	Comments
Interface Rqmts Doc	Program Kickoff	See Sect C.1
Initial Target Specification	Program Kickoff	S/C Weight, Target Orbit, Separation Attitude; See Sect C.3.4
Range Safety Mission Orientation Briefing Input	Program Kickoff	Top-Level Description of Spacecraft & Mission Design
CAD Model	30 days After Program Kickoff	See Sect C.3.1
Coupled Loads Model	6 mo Before Design Review	See Sect C.3.2
Thermal Models	5 mo Before Design Review	See Sect C.3.3
Preliminary Launch Windows	5 mo Before Design Review	Support Thermal Analysis; See Sect C.3.3
In-Flight Breakup Data	Program Kickoff	See Sect C.3.6.4.3
Intact Impact Breakup Data	Program Kickoff	See Sect C.3.6.4.2
Prelim S/C MSPSP	Program Kickoff	See Sect C.3.6.1
S/C EMI/EMC Cert Letter	6 mo Before Launch	See Sect C.3.5
S/C EED Analysis	6 mo Before Launch	See Sect C.3.5
Procedures Used on CCAFS	4 mo Before Launch	See Sect C.3.6.2
Procedures Used at Astrotech	2 mo Before S/C Arrival at Astrotech	See Sect C.3.6.2
Final Target Specification	90 days Before Launch	Date Depends on Mission Design; See Sect C.3.4
S/C Environ Qual Test Reports	As Available	See Table C.2-1 for Envir Qual Rqmts

C.3.1 Computer-Aided Design (CAD) Data Transfer Requirements

CAD data must be provided according to specified software formats. The Atlas program supports three UNIX-based CAD systems: Parametric Technology Corporation (PTC) Pro/Engineer (Pro/E), PTC CADDS 5, and Structural Dynamics Research Corporation (SDRC) I-DEAS Master Series. When CAD data do not come from a supported software platform, Lockheed Martin prefers to receive solid model data translated through the Standard for the Exchange of Product Model Data (STEP) converter. If this is not available, an Initial Graphics Exchange Specification (IGES) 4.0 or higher file from a three-dimensional (3-D) wireframe system or wireframe extracted from a solid model will suffice.

Prerequisites to Data Transfer—The following criteria should be met before transferring CAD data:

1) All entities (i.e., parts, assemblies) should be visible (unblanked) and fontless (e.g., no center lines, phantom lines, or thick lines);
2) Single part or model files containing 30,000 entities or more should be divided up into assemblies of smaller files all having the same origin,
3) Single CADDS 5 models (containing _pd, _fd, _td, vp_links, draw files, and execute routines) should be in the same subdirectory structure. The entire directory should then be transferred as a single compressed file;
4) Subfigures should not be included in the model unless they are included in the data transfer.

Data Transfer—Tape media (8-mm, ¼-in. data cartridge, and 4-mm DAT) is the preferred transfer method for all database files. Files should be compressed and transferred to tape using the UNIX operating system. The following tape archive retrieval (TAR) command syntax should be used: tar cvf/dev/rmt0 partfilenames. DOS- and Microsoft Windows-based 3 ½-in. disks can also be accommodated. The spacecraft contractor should verify that the tape files contain the desired results by reading the files back onto the originating CAD system from tape before transmittal to Lockheed Martin.

The file transfer protocol (FTP) method of data transfer is also feasible. An account can be established on a Lockheed Martin firewall server for this purpose. Once the account is set-up and a password is provided for access, up to 600 MB of data can be transmitted at one time. An alternative would involve the contractor providing similar access to one of their systems via a temporary account. In either case, the transfer type should be set to binary.

The following information must be sent with the CAD data file regardless of transfer method:

1) Name and phone number of the computer system administrator/operator;
2) Name and phone number of the contact person who is familiar with the model in case problems or questions arise;
3) Spacecraft axis and coordinate system;
4) Spacecraft access requirements for structure not defined on CAD model (i.e., fill and drain valve locations);
5) Multiview plot of the model;
6) File size (MB);
7) Uudecode (UNIX-based) information, if applicable.

C.3.2 Coupled-Loads Analysis Model Requirements

The customer-supplied dynamic mathematical model of the spacecraft should consist of generalized mass and stiffness matrices, and a recommended modal damping schedule. The desired format is Craig-Bampton, constrained at the Centaur interface in terms of spacecraft modal coordinates and discrete Centaur interface points. The spacecraft dynamic model should have an upper frequency cutoff of 90 to 100 Hz. The output transformation matrices (OTM) should be in the form that, when multiplied by the spacecraft modal and interface generalized coordinate responses, will recover the desired accelerations, displacements, or internal loads. One of the OTMs should contain data that will allow calculation of loss of clearance between the payload fairing (PLF) and critical points on the spacecraft. Typically, the size of the OTMs is 200 to 500 rows for accelerations, 50 to 200 rows for displacements, and 300 to 1,000 rows for internal loads.

C.3.3 Spacecraft Thermal Analysis Input Requirements

Spacecraft geometric and thermal mathematical models are required to perform the integrated thermal analysis. These models should be delivered electronically or on a computer diskette with printed listings of all the files. The geometric mathematical model (GMM) and thermal mathematical model (TMM) size should be less than 600 nodes each.

The preferred GMM format is Thermal Radiation Analysis System (TRASYS) input format. Alternate formats are ESABASE or NEVADA input formats. The documentation of the GMM should include illustrations of all surfaces at both the vehicle and component levels, descriptions of the surface optical properties, and the correspondences between GMM and TMM nodes.

The preferred TMM format is System-Improved Numerical Differencing Analyzer (SINDA). The TMM documentation should include illustrations of all thermal modeling; detailed component power dissipation’s for prelaunch, ascent, and on-orbit mission phases; steady-state and transient test case boundary conditions, output to verify proper conversion of the input format to Lockheed Martin analysis codes; maximum and minimum allowable component temperature limits; and internal spacecraft convection and radiation modeling.

In addition to the TMM and GMM, launch window open and close times for the entire year are required inputs to the integrated thermal analysis.

C.3.4 Target Specifications

Target specifications normally include the final mission transfer orbit (apogee and perigee radius, argument of perigee, and inclination), spacecraft weight, and launch windows. The final target specification is due to Lockheed Martin 90 days before launch for missions incorporating minimum residual shutdown (MRS) or in-flight retargeting (IFR), and 60 days before launch for guidance commanded-shutdown geosynchronous transfer orbit (GTO) missions.

C.3.5 Spacecraft Electromagnetic Interference (EMI) and Electromagnetic Compatibility (EMC) Certification Letter and Electroexplosive Device (EED) Analysis

A final confirmation of spacecraft transmitter and receiver parameters, and emission and susceptibility levels of electronic systems is required 6 months before launch. This includes consideration of emissions from such electronic equipment as internal clocks, oscillators, and signal or data generators; and likelihood of electronics and items such as electroexplosive devices (EED) to cause upset, damage, or inadvertent activation. These characteristics are to be considered according to MIL-STD-1541 requirements to assure that appropriate margins are available during launch operations. Lockheed Martin will use the spacecraft data to develop a final analysis for the combined spacecraft/launch vehicle and site environment.

C.3.6 Safety Data

To launch from CCAFS on the Eastern Range or VAFB on the Western Range, spacecraft design and ground operations must meet the applicable launch-site safety regulations. Refer to Section 4.3.1 for a listing of these regulations. Mission-specific schedules for development and submittal of the spacecraft safety data will be coordinated in safety working group meetings during the safety integration process. Refer to Section 4.3.2 for additional information on this process.

C.3.6.1 Missile System Prelaunch Safety Package (MSPSP)—The MSPSP is the data package that describes in detail the hazardous and safety-critical spacecraft systems/subsystems, their interfaces, and the associated ground support equipment (GSE). In addition, the SC MSPSP provides verification of compliance with the applicable Range Safety requirements. The SC MSPSP must be approved by Range Safety before the arrival of spacecraft elements at the launch site.

C.3.6.2 Spacecraft Launch Site Procedures—Before any procedures are performed at the launch site, hazardous spacecraft procedures must be approved by the Range Safety Office and/or the safety organization at the appropriate spacecraft processing facility (e.g., Astrotech, NASA, DoD). Since the approving authority must also concur with the nonhazardous designation of procedures, all spacecraft launch-site procedures must be submitted for review. Lockheed Martin's System Safety group is the point of contact for submittal/coordination of all spacecraft data (refer to Section 4.3.2)

C.3.6.3 Radiation Protection Officer (RPO) Data—Permission must be received from the Range RPO before spacecraft radio frequency (RF) emissions are allowed at the launch complex. The required RPO data includes descriptions of the equipment involved, the procedures that will be used, and information on the personnel who will be running the procedures.

C.3.6.4 Spacecraft Breakup Data Requirements—The spacecraft data described in the following three subsections is required for the Atlas program to complete mission-specific analyses that satisfy 45th Space Wing/SEOE requirements for submitting a request for Range Safety Flight Plan Approval (FPA).

C.3.6.4.1 Inadvertent Spacecraft Separation and Propulsion Hazard Analysis—This data set is related to inadvertent separation of the spacecraft during early ascent and the potential for launch area hazards that could exist in the event spacecraft engine(s) fire. Typical spacecraft propulsion system data provided by the customer include the maximum tanked weight, maximum loaded propellant weight, maximum axial thrust (all motors), and maximum resultant specific impulse.

***C.3.6.4.2 Intact Impact Analysis*—**This data set is related to the ground impact of the spacecraft. The intact impact analysis assumes ground impact of a fully loaded, fueled, intact spacecraft. It also assumes propellants will combine and explode. Typical spacecraft data provided by the customer include the types and weights of explosive propellants; estimates of the number of pieces of the spacecraft that could break off in an explosion; and the location, size, weight, and shape of each piece.

***C.3.6.4.3 Destruct Action Analysis*—**This data set is related to the flight termination system (FTS) destruction of the launch vehicle. The destruct action analysis assumes in-flight destruction of the vehicle by detonation of the Range Safety charge. Typical spacecraft data provided by the customer include an estimate of the number of spacecraft pieces that could break off because of commanded vehicle destruction and estimates of their size, weight, shape, and location on the spacecraft.

C.3.7 SPACECRAFT PROPELLANT SLOSH MODELING

Lockheed Martin performs a spacecraft and Centaur propellant slosh analysis as part of the launch vehicle's autopilot design. The spacecraft propellant tank geometry, propellant densities, and maximum and minimum tank fill levels are required to perform this analysis. The data other than the tank fill levels should be available as part of the ICD or MSPSP inputs.

APPENDIX D—PAYLOAD FAIRINGS

D.1 ATLAS 4-M PLF (LPF AND EPF)

The Atlas 4-m large payload fairing (LPF) and extended payload fairing (EPF) have a 4.2-m (165-in.) outer skin line diameter cylindrical section. The major sections of these fairings are the boattail, the cylindrical section, and the conical section that is topped by a spherical cap (Fig. D.1-1 and D.1-2). The EPF was developed to support launch of larger volume spacecraft by adding a 0.9-m (36-in.) long cylindrical plug to the top of the cylindrical section of the large payload fairing. All of these sections consist of an aluminum skin, stringer, and frame construction with vertical, split-line longerons that allow the fairing to separate into bisectors for jettison. Electrical packages required for the fairing separation system are mounted on the internal surface of the boattail. Ducting for the Centaur upper-stage hydrogen tank venting system and cooling ducts for the equipment module packages are also attached to the boattail.

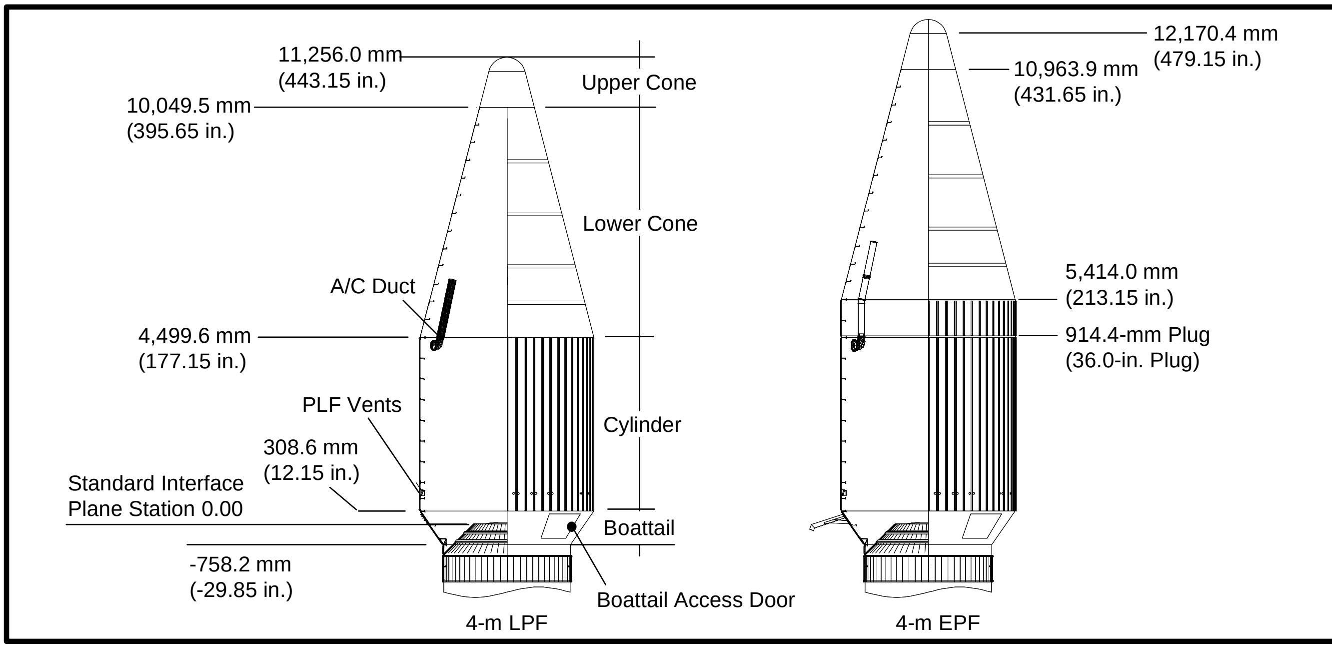


Figure D.1-1 Atlas 4-m Payload Fairings

Static Payload Envelope—The usable volume for a spacecraft inside the payload fairing is defined by the static payload envelope. The Atlas 4-m payload fairing provides a 3,750-mm (147.64-in.) diameter envelope in the cylindrical section with additional volume available in the conical section of the payload fairing. On a mission-specific basis, the envelope may be increased to a 3,850-mm (151.57-in.) diameter in localized areas by modifying portions of the payload fairing structure. This envelope represents the maximum allowable spacecraft static dimensions (including)

Figure D-1.2 Atlas 4-m Payload Fairing manufacturing tolerances) relative to the spacecraft/payload adapter interface. These envelopes include allowances for payload fairing static tolerances and misalignments, payload fairing and spacecraft dynamic deflections, and payload fairing out-of-round conditions, and were established to insure that a minimum 25-mm (1-in.) clearance between the spacecraft and the payload fairing is maintained. These envelopes were developed and are applicable for spacecraft that meet the stiffness and load requirements discussed in Section 3.2.1. Clearance layouts and analyses are performed for each spacecraft configuration and, if necessary, critical clearance locations are measured after the spacecraft is encapsulated inside the fairing to ensure positive clearance during flight. Detailed views of the static payload envelope for the LPF and EPF are shown in Figures D.1-3 to D.1-5.

For customers that request a dynamic payload envelope, the static payload envelopes shown in Figures D.1-3 and D.1-4 can be conservatively used for preliminary design purposes. These envelopes meet the requirements for dynamic payload envelopes of the Evolved Expendable Launch Vehicle Standard Interface Specification. The static payload envelopes were based on a combination of flight, jettison, and ground handling conditions, and the spacecraft dynamic deflections are only a consideration during flight conditions. Mission-specific modifications to these envelopes, either on a static or dynamic basis, are dependent upon the spacecraft configuration and dynamic behavior, and are considered based on analysis performed for each mission.

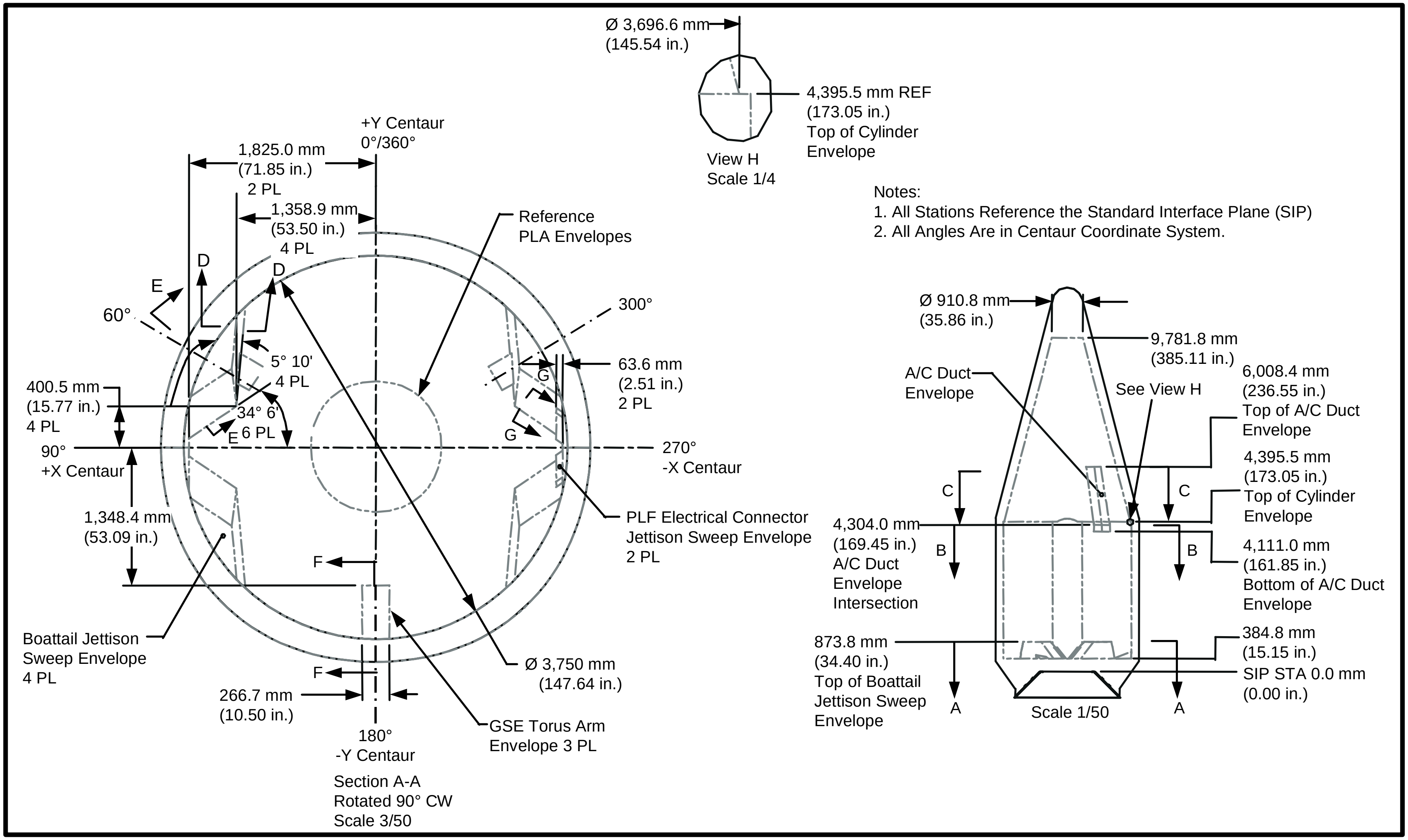


Figure D.1-3 Atlas 4-m LPF Static Payload Envelope

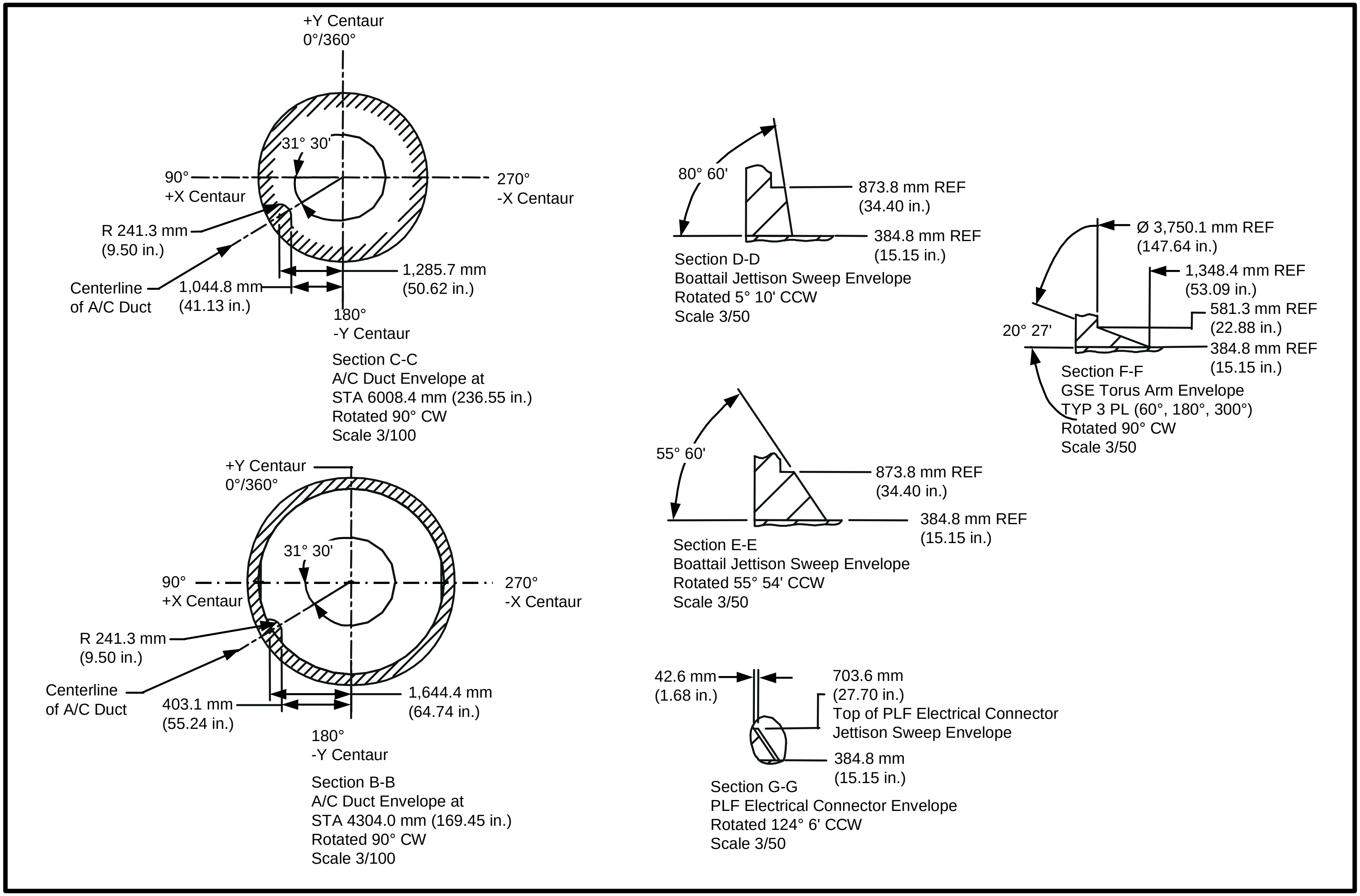


Figure D.1-3 (concl)

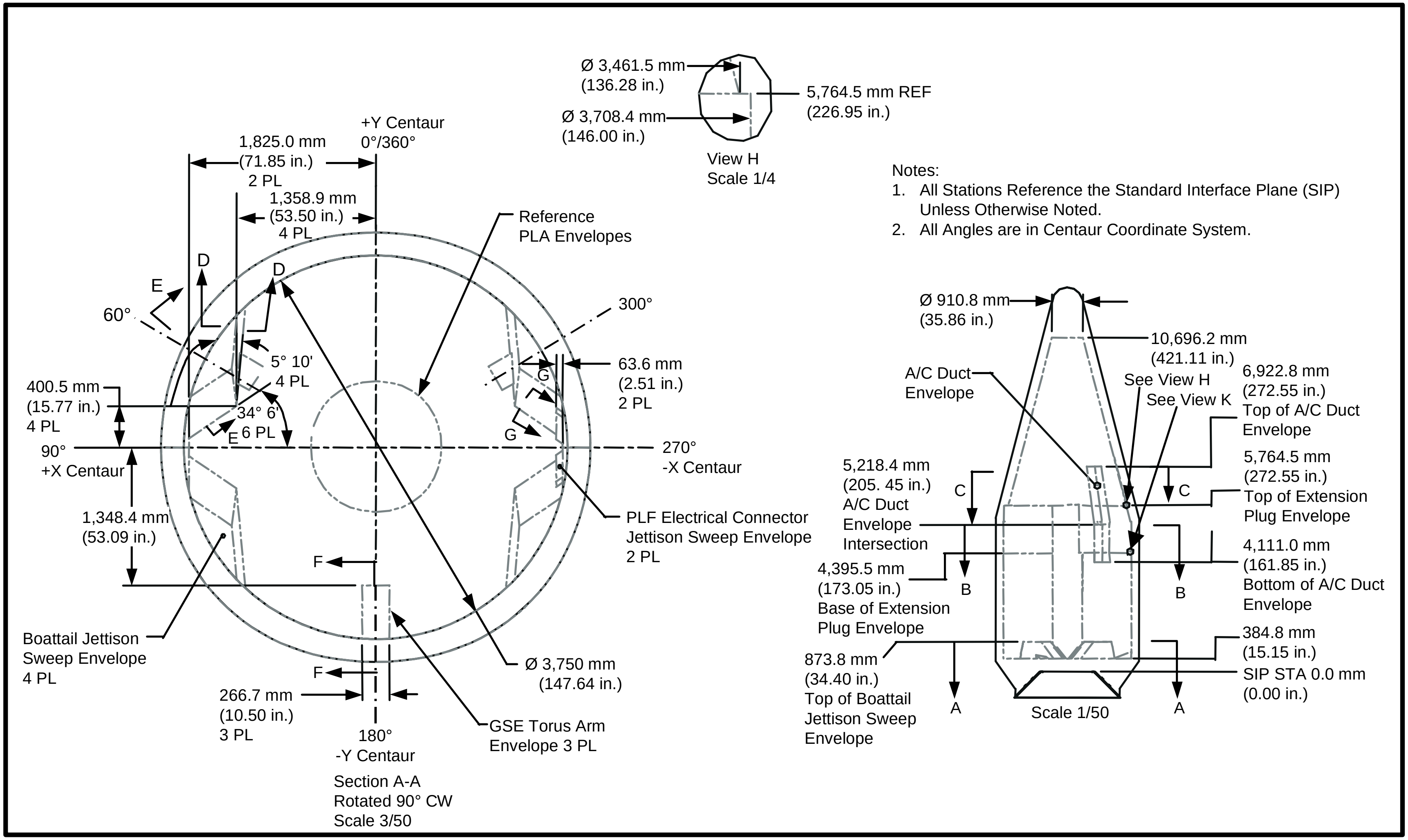


Figure D.1-4 Atlas 4-m EPF Static Payload Envelope

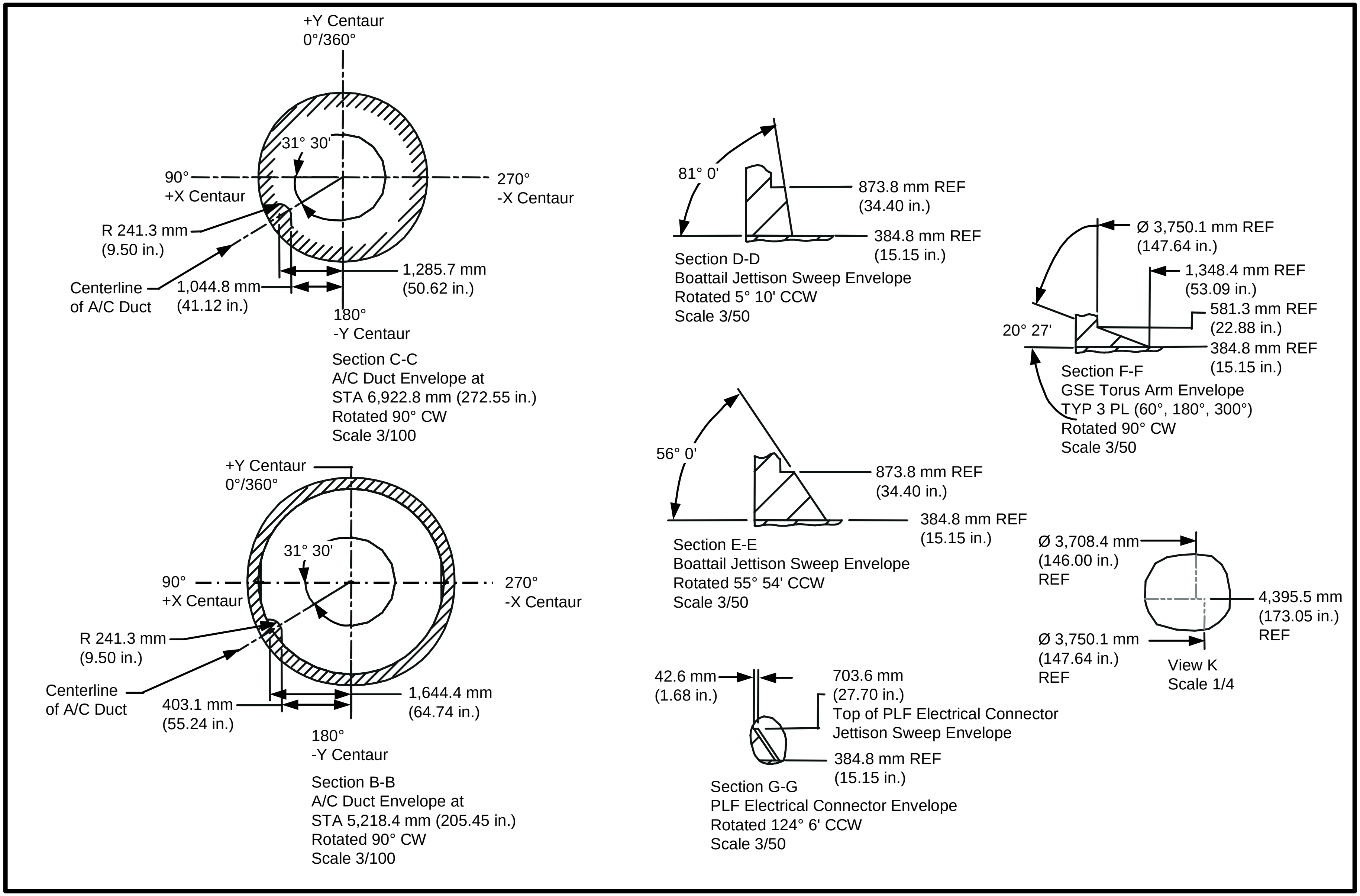


Figure D.1-4 (concl)

Payload Compartment Environmental Control—The Atlas payload fairing is designed to provide a suitable acoustic, thermal, electromagnetic, and contamination controlled environment for the payload. On the Atlas III and Atlas V vehicles, acoustic panels (Fig. D.1-6) are provided on the cylindrical section of the fairing to attenuate the sound pressure levels to acceptable limits (shown in Section 3.2.2). These panels may also be added to the Atlas IIAS vehicle as a mission-specific option. The heritage acoustic panels shown in Figure D.1-6 are being redesigned to be more producible and to reduce infringement on the payload envelope. The new panel design will not change acoustic attenuation characteristics or acoustic levels for the spacecraft.

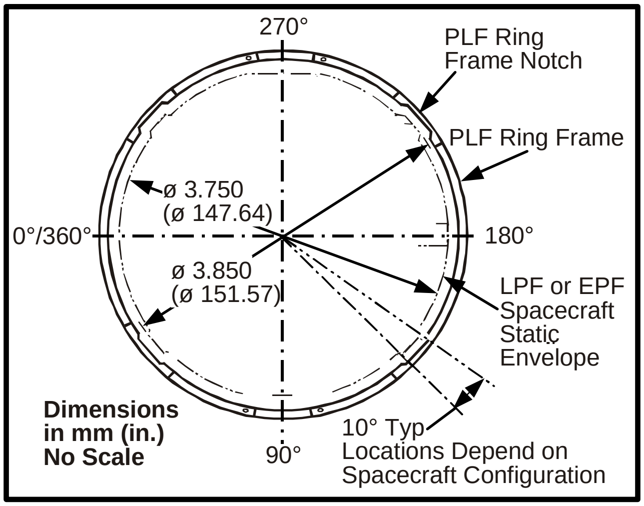


Figure D.1-5 Atlas 4-m PLF Static Payload Envelope, 3,850 mm-Diameter Modification

For thermal control, the external surface of the conical section fairing is insulated with cork to limit temperatures to acceptable values (Sections 3.1.1 and 3.2.5). Noncontaminating, low emittance thermal control coatings are used on fairing internal surfaces to reduce incidental heat fluxes to the spacecraft. The acoustic panels located in the cylindrical section of the payload fairing also serve to reduce heating in the payload compartment. Thermal shields (Fig. D.1-7) may be added in the conical section of the fairing to provide additional thermal control as a mission-specific option. During prelaunch activities,

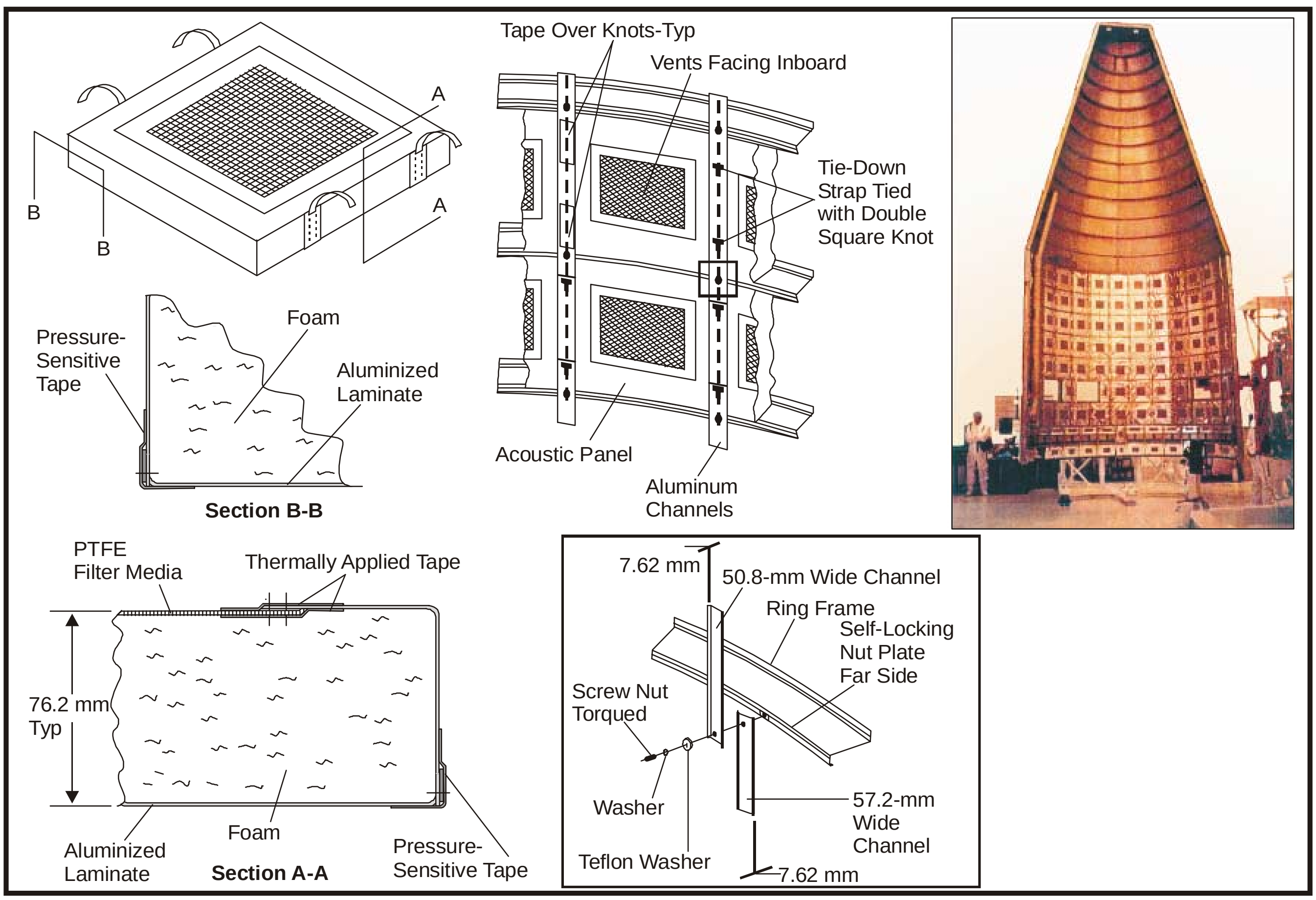


Figure D.1-6 Atlas 4-m PLF Acoustic Panels

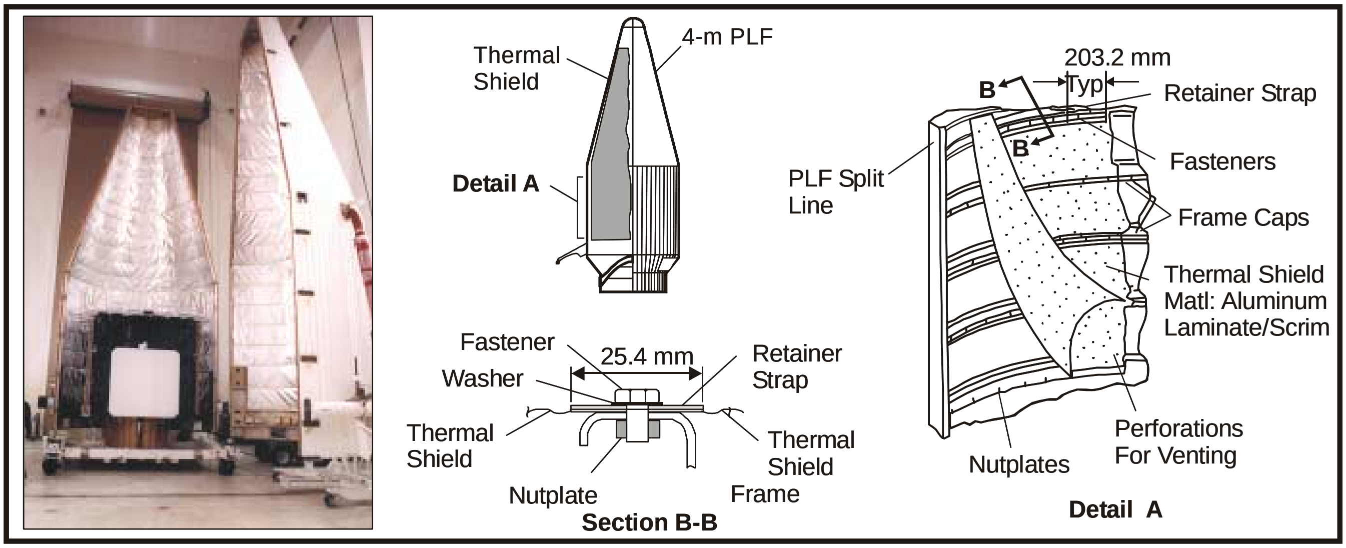


Figure D.1-7 Atlas 4-m PLF Thermal Shields

conditioned air is provided through the air-conditioning duct located in the upper cylindrical and lower conical portion of the fairing. This duct directs conditioned air to provide thermal and humidity control upward into the conical section to avoid direct impingement on the spacecraft. Vent holes and housings are mounted on the lower part of the cylindrical section for the LPF and EPF to allow air from the air-conditioning system to exit the fairing and to allow depressurization during ascent. A secondary environmental control system may be added to provide additional cooling or to direct cooling air to specific points on the payload. This mission-specific option has been used on several missions and design approaches developed for these past missions can be adapted for future applications.

The metallic construction of the fairing provides electromagnetic shielding for the spacecraft and serves to attenuate the external RF environment when it is in place during ground operations. Electrically conductive seal materials are used between mating surfaces on the payload fairing to preclude creating entry paths for RF signals.

The payload fairing is fabricated and operated according to requirements of the Atlas Contamination Control Plan described in Section 3.1.3. This plan establishes rigorous procedures to ensure that all hardware that comes into contact with the payload meets cleanliness requirements and may be tailored to meet mission-specific payload needs.

Payload Access—The four large doors in the boattail section of the 4-m payload fairing (Fig. D.1-8) provide primary access to Centaur forward adapter packages and the encapsulated spacecraft. Work platforms can be inserted into the payload compartment through these doors to allow access to spacecraft hardware near the aft end of the payload compartment. If additional access to the spacecraft is required, doors can be provided on a mission-unique basis on the cylindrical section of each payload fairing. The available sizes and allowable locations for these doors are shown in Figure D.1-8. Access is permitted from the time of payload encapsulation until close-out operations before Mobile Service Tower (MST) rollback at LC-36 or Mobile Launch Platform (MLP) transport at LC-41.

Payload RF Communications—A reradiating system allows payload RF telemetry transmission and command receipt communications after the payload is encapsulated in the spacecraft until time of launch. The airborne system consists of an antenna, a mounting bracket, and cabling inside the payload fairing. Reradiating antennas are available in S-, C-, and Ku-bands. The pick-up antenna is mounted on a bracket at a location appropriate for the spacecraft configuration (Fig. D.1-9). This antenna acquires the spacecraft RF signal and routes it via RF cabling to payload fairing T-0 disconnect. A cable runs from the T-0 disconnect to a junction box that routes the signal to a customer-specified location.

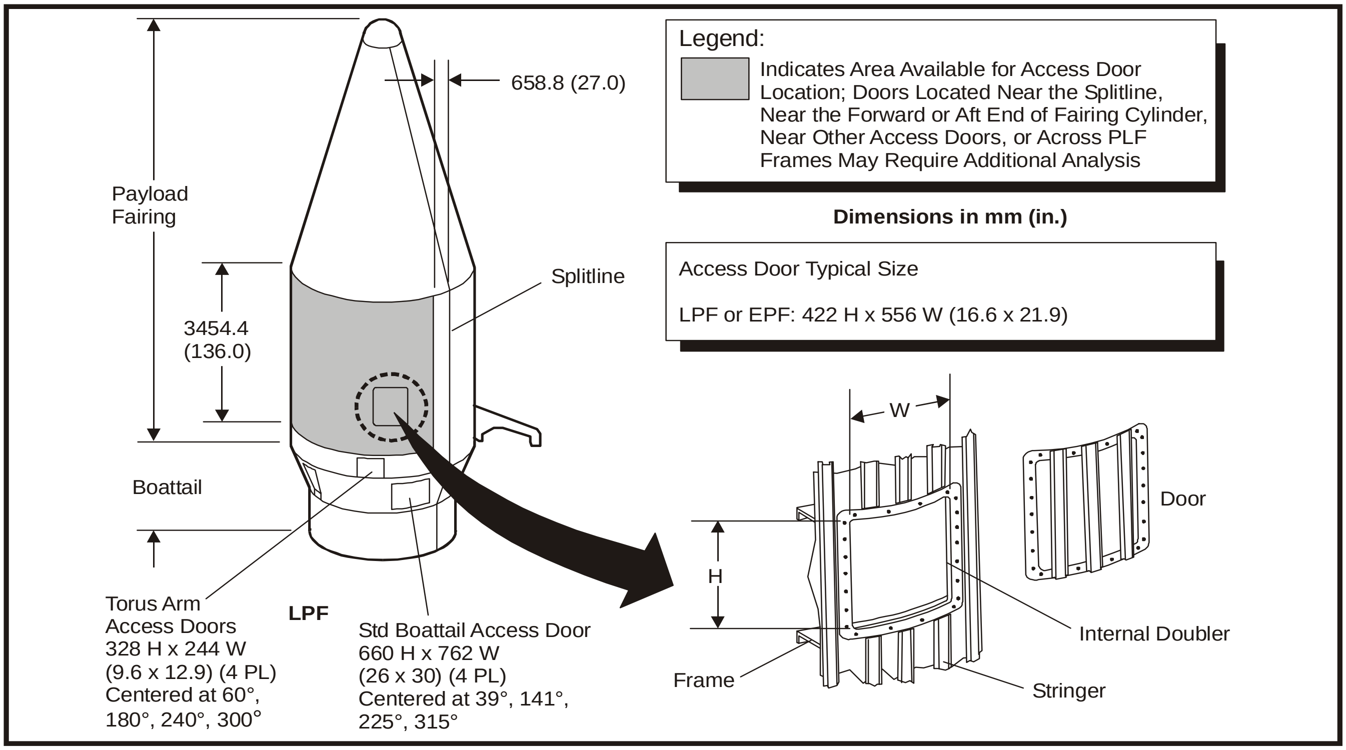


Figure D.1-8 Atlas 4-m PLF Access Doors

Customer Logo—A customer-specified logo may be placed on the cylindrical section of the payload fairing. Logos up to 3.05 x 3.05 m (10 x 10 ft.) are provided as a standard service. The area of the payload fairing reserved for customer logos is shown in Figure D.1-10. The Atlas program will work with the customer and provide layouts of the logo on the launch vehicle to assist in determining their proper size and location.

Figure D.1-9 Atlas 4-m PLF RF Reradiate Antenna Installation

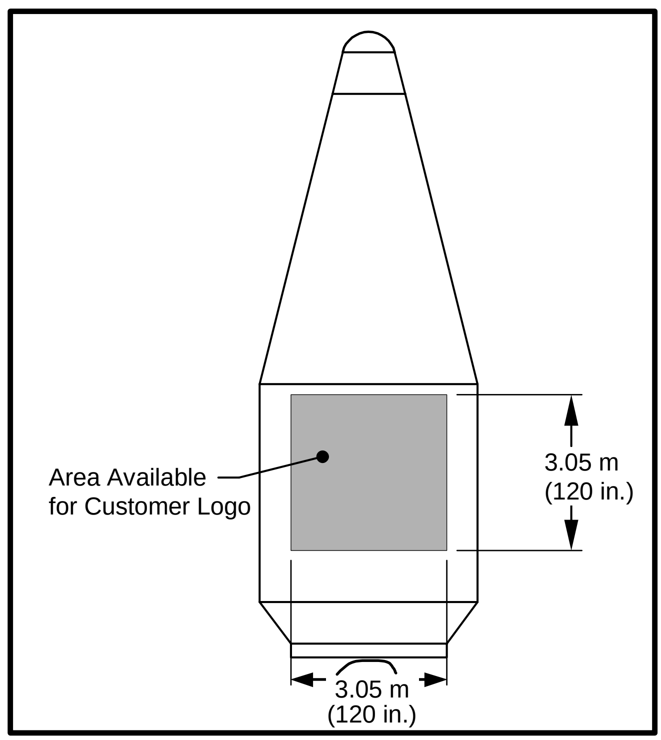


Figure D.1-10 Atlas 4-m PLF Customer Logo Provisions

D.2 ATLAS 5-M PLF (SHORT AND MEDIUM)

The 5-m diameter payload fairing (PLF) was developed along with the increased launch vehicle performance to accommodate growing spacecraft needs. Atlas' 5-m short- and medium-length payload fairings (PLF) have a 5.4-m (213.6-in.) outer skin line diameter cylindrical section. This fairing is a bi-sector fairing with a composite structure made from sandwich panels with composite facesheets and a vented aluminum honeycomb core. This fairing and its associated separation system are derived from an existing flight-proven system. There are two major components of the fairing. The lower section is the base module (BM) that encapsulates the Centaur upper stage. The upper section is the common payload module (CPM) that encapsulates the spacecraft and consists of a cylindrical section that transitions into a constant radius ogive nose section topped by a spherical nose cap (Fig. D.2-1). The ogive shape minimizes aerodynamic drag and buffet. For the 5-m medium payload fairing, a 2,743-mm (108-in.) lower payload module (LPM) is added to the base of the common payload module to increase the available payload volume. The fairing interfaces with the launch vehicle at the fixed conical boattail that is attached to the launch vehicle first stage. Clearance losses for payloads are minimized by the Centaur forward-load reactor (CFLR) system that stabilizes the top of the Centaur, thereby reducing the relative motion between the PLF and payload. The PLF sections provide mounting provisions for various secondary systems. Payload compartment cooling system provisions are in the ogive section of the fairing. Electrical packages required for the fairing separation system are mounted on the internal surface of the fairing.

Static Payload Envelope—The useable volume for a spacecraft inside the payload fairing is defined by the static payload envelope. The Atlas 5-m payload fairing provides a 4,572-mm (180-in.) diameter envelope in the cylindrical section with additional volume available in the ogive section of the payload fairing. This envelope represents the maximum allowable spacecraft static dimensions (including manufacturing tolerances) relative to the spacecraft/payload adapter interface. These envelopes were established to insure that a minimum 25-mm (1-in.) clearance between the spacecraft and the payload fairing is maintained and include allowances for payload fairing static tolerances and misalignments, spacecraft-to-payload fairing dynamic deflections, and payload fairing out-of-round conditions. These envelopes were developed and are applicable for spacecraft that meet the stiffness and load requirements discussed in Section 3.2.1. Clearance layouts and analyses are performed for each spacecraft configuration and, if necessary, critical clearance locations are measured after the spacecraft is encapsulated inside the fairing to ensure positive clearance during flight. Detailed views of the static payload envelope for the 5-m short and 5-m medium payload fairings are shown in Figure D.2-2.

For customers requesting a dynamic payload envelope, the static payload envelopes shown in Figure D.2-2 can be conservatively used for preliminary design purposes. These envelopes meet the requirements for dynamic payload envelopes of the Evolved Expendable Launch Vehicle Standard Interface Specification. The static payload envelopes were based on a combination of flight, jettison, and ground handling conditions, and the spacecraft dynamic deflections are only a consideration during flight conditions. Mission-specific modifications to these envelopes, either on a static or dynamic basis, are dependent upon the spacecraft configuration and dynamic behavior and are considered based on analysis performed for each mission.

Payload Compartment Environmental Control—The Atlas 5-m payload fairing is designed to provide a suitable acoustic, thermal, electromagnetic, and contamination controlled environment for the payload. Fairing acoustic protection (FAP) (Fig. D.2-3) is provided as a standard service to attenuate the sound pressure levels to acceptable limits (Section 3.2.2).

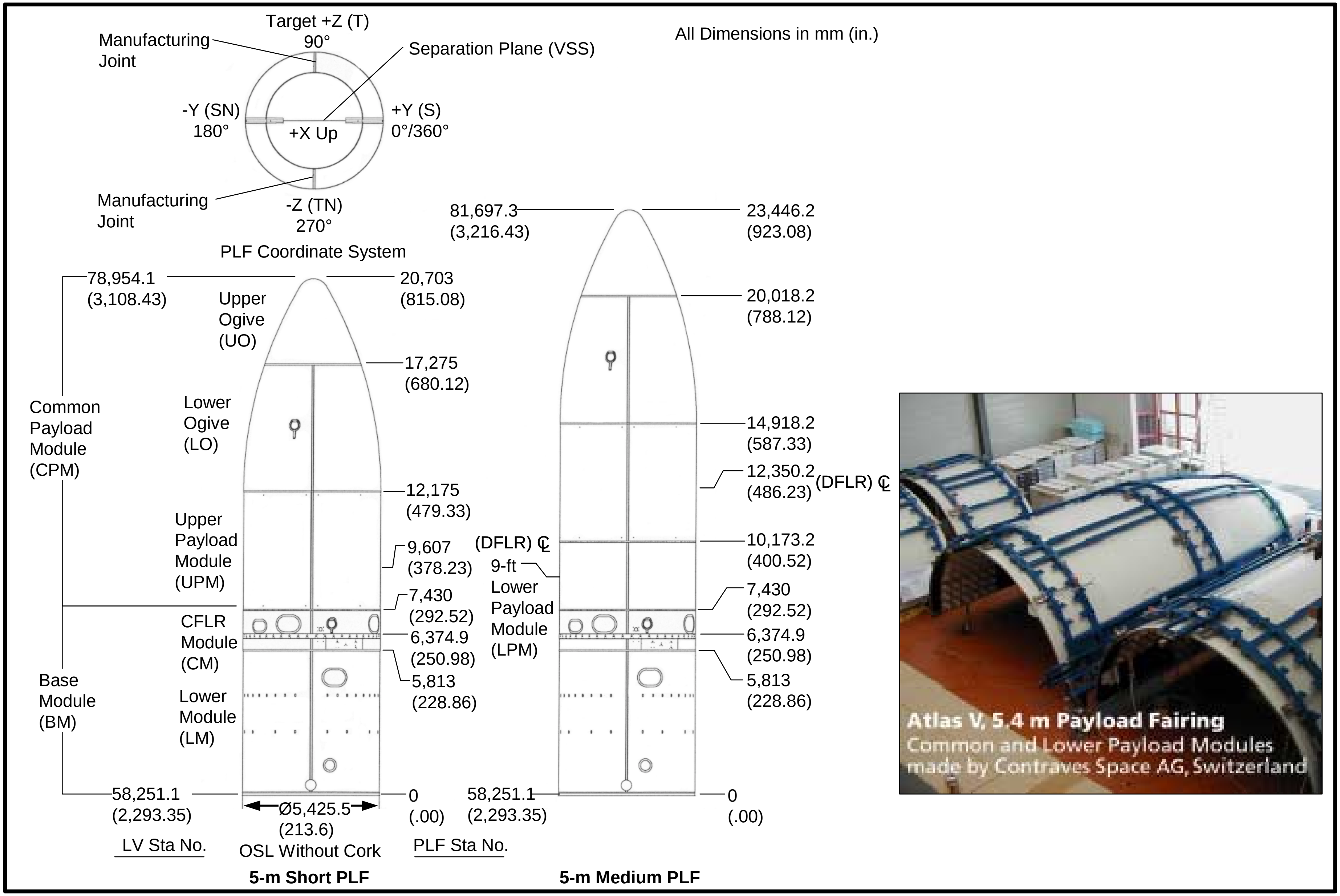


Figure D.2-1 Atlas 5-m Payload Fairing Configurations

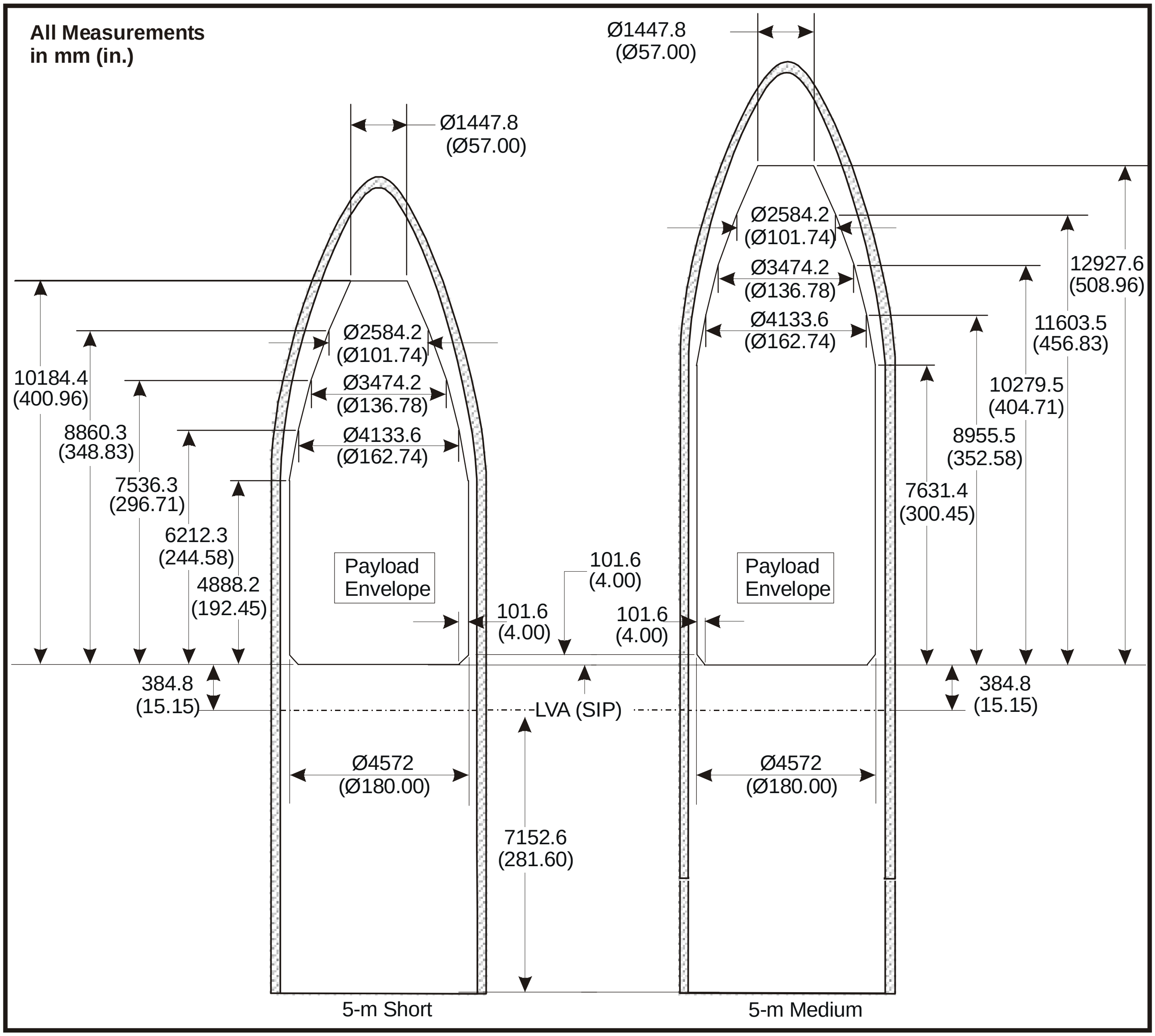


Figure D.2-2 Atlas 5-m PLF Static Payload Envelope

For thermal control, the external surface of the fairing is insulated with cork and painted white to limit temperatures to acceptable values (Sections 3.1.1 and 3.2.5). The fairing acoustic protection panels also serve to reduce heating in the payload compartment. During prelaunch activities, conditioned air is provided through the air-conditioning inlet located in the ogive section of the fairing. This inlet directs conditioned air to provide thermal and humidity control for the payload compartment and prevents direct impingement of this flow on the spacecraft. Vent ports and vent port assemblies are mounted in the mid-section of the base module for air from the air-conditioning system to exit the fairing and to allow depressurization during ascent. A secondary environmental control system may be added as a mission-specific option to provide additional cooling or to direct cooling air to specific points on the payload.

Electrically conductive paint is used on the outside of the payload fairing to prevent electrostatic build-up on the payload fairing.

The payload fairing is fabricated and operated according to the requirements of the Atlas Contamination Control plan described in Section 3.1.3. This plan establishes rigorous procedures to

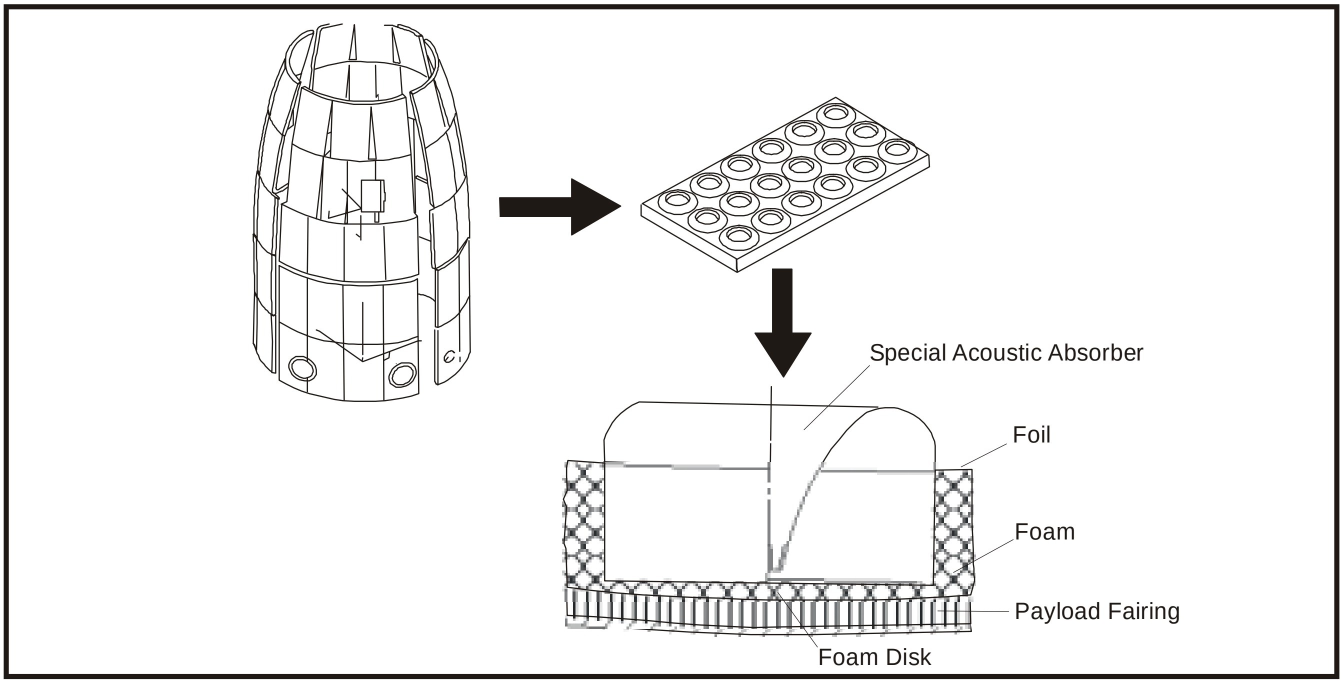


Figure D.2-3 Atlas 5-m Fairing Acoustic Protection

ensure that all hardware that comes into contact with the payload meets cleanliness requirements and may be tailored to meet mission-specific payload needs.

Payload Access—The 5-m payload fairings have four large doors in the base module portion of the payload fairing to provide primary access to the Centaur forward adapter packages and the encapsulated spacecraft. The doors provide an opening of approximately 600 x 900 mm (24 x 36 in.). Work platforms can be inserted through these doors onto the CFLR deck to allow access to spacecraft hardware near the aft end of the payload module. If additional access to the spacecraft is required, additional doors can be provided on a mission-specific basis on the cylindrical and ogive section of the payload fairing. The available sizes and allowable locations for these doors are shown in Figures D.2-4 and D.2-5. Access is permitted from the time of payload encapsulation until close out operations before mobile launch platform (MLP) transport at LC-41.

Payload RF Communications—A reradiating system allows payload RF telemetry transmission and command receipt communications after the payload is encapsulated inside the spacecraft until the time of launch. The airborne system consists of an antenna, a mounting bracket, and cabling inside the payload fairing. Reradiating antennas are available in the S, C, and Ku bands. The pickup antenna is mounted on a bracket at a location appropriate for the spacecraft configuration. This antenna acquires the spacecraft RF signal and routes it via RF cabling to the payload fairing T-0 disconnect. A cable runs from the T-0 disconnect to a junction box that routes the signal to a customer-specified location.

Customer Logo—A customer-specified logo may be placed on the cylindrical section of the payload fairing. Logos up to 3.05 x 3.05 m (10 x 10 ft.) are provided as a standard service. The area of the payload fairing reserved for customer logos is shown in Figure D.2-6. The Atlas program will work with the customer and provide layouts of the logo on the launch vehicle to assist in determining their proper size and location.

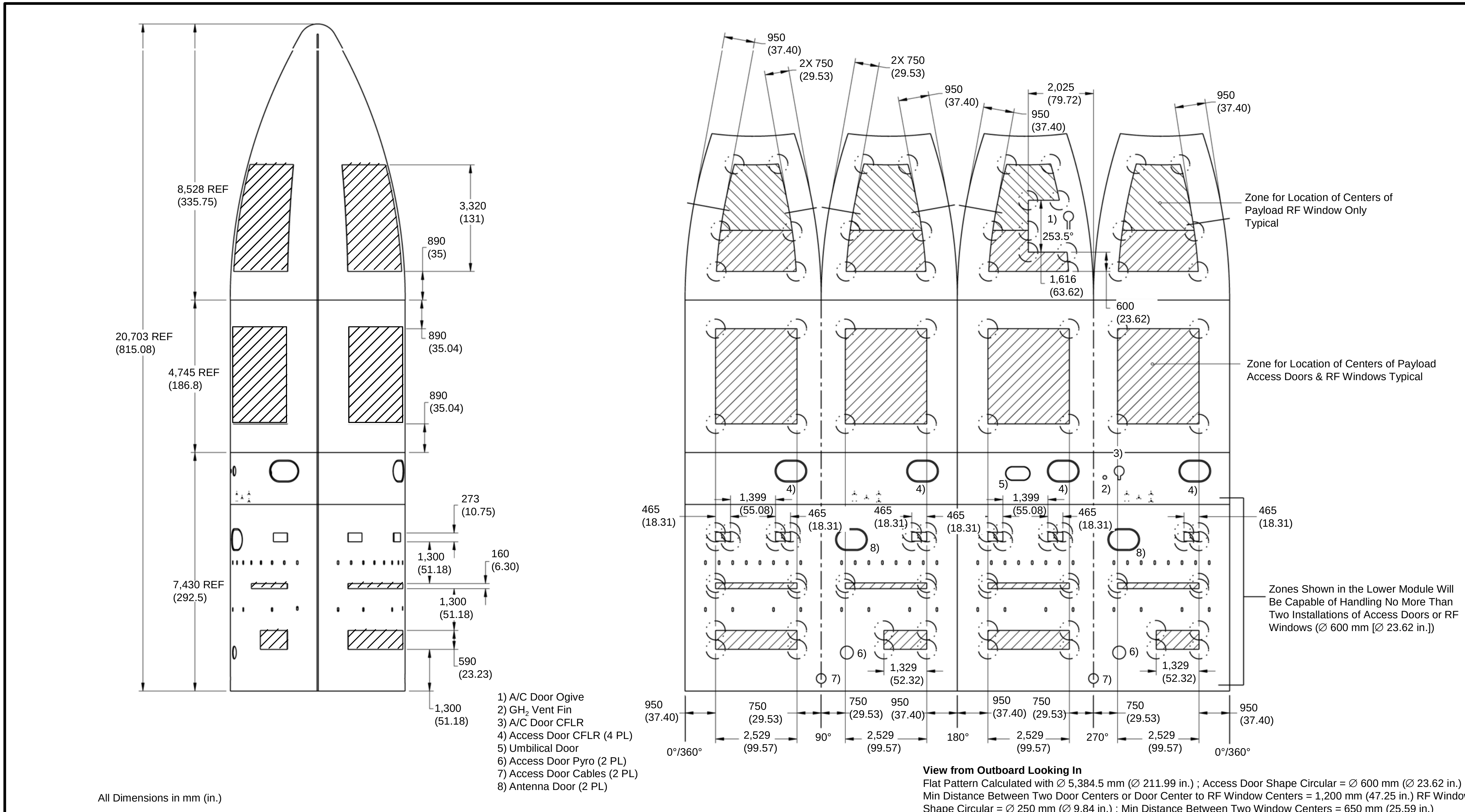


Figure D.2-4 Atlas 5-m Short PLF Mission-Specific Access Doors

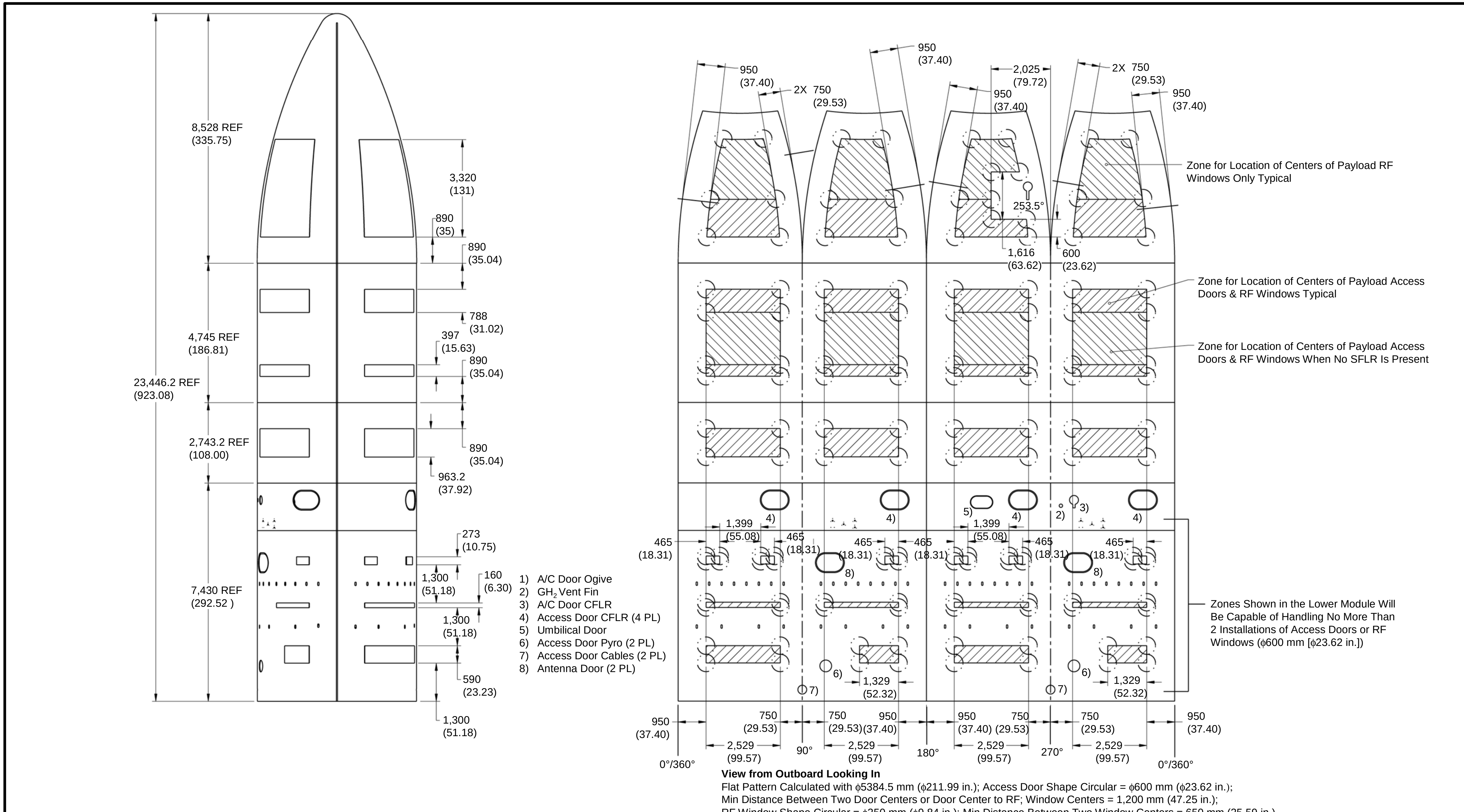


Figure D.2-5 Atlas 5-m Medium PLF Mission-Specific Access Doors

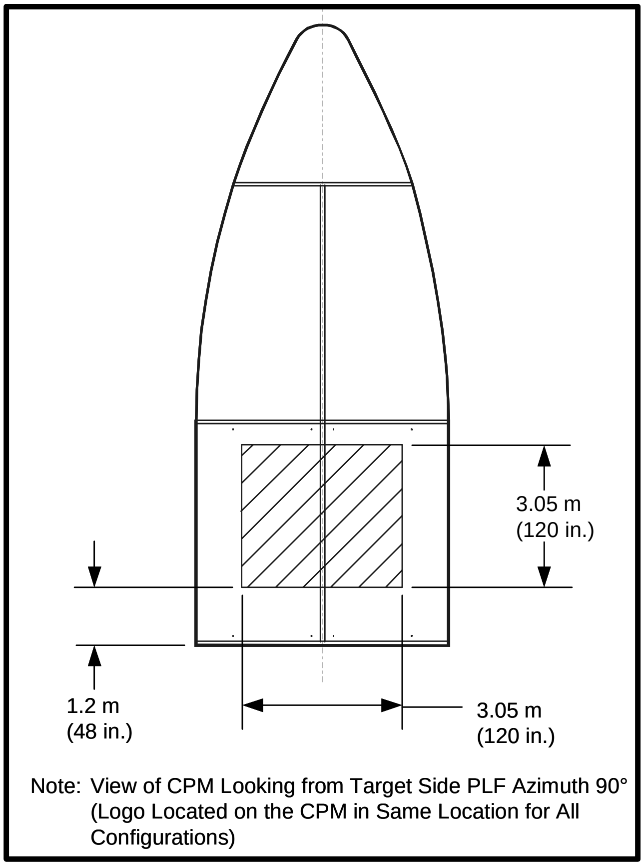


Figure D.2-6 Atlas 5-m PLF Customer Logo Provisions

APPENDIX E—PAYLOAD ADAPTERS

E.1 ATLAS STANDARD INTERFACE PLANE

The Atlas standard interface plane (SIP) provides a standardized bolted interface for Atlas and customer-provided payload adapters. This interface consists of a machined ring that contains 121 boltholes on a 1,570-mm (62.010-in.) diameter. For vehicles using a 4-m payload fairing, this plane is at the top of the Atlas forward adapter. For vehicles using a 5-m payload fairing, this plane is at the top of an Atlas-provided C22 launch vehicle adapter that is mounted on top of the forward adapter (Fig. E.1-1). The C22 adapter is standard with the 5-m fairing to allow launch vehicle ground support equipment (GSE) interfaces and is designed to provide an interface that is identical to that provided by the forward adapter. In this configuration, cost and performance impacts of the C22 launch vehicle adapter are considered to be a part of the basic launch vehicle service and are not counted against payload systems weight.

For customers that provide their own adapter, the standard interface plane is the interface point between the spacecraft-provided hardware and the launch vehicle. If a customer-provided spacecraft adapter is used with a 4-m payload fairing, it must provide interfaces for ground handling, encapsulation, and transportation equipment. In particular, there need to be provisions for torus arm fittings and an encapsulation diaphragm unless a launch vehicle-supplied intermediate adapter is used. Information on these interface requirements is in Atlas Specification S/M-00-025, “Torus Arm And Ground Transport Vehicle Interface Control Document for Payloads Using the 4-meter Payload Fairing,” which is available on request to Atlas customers.

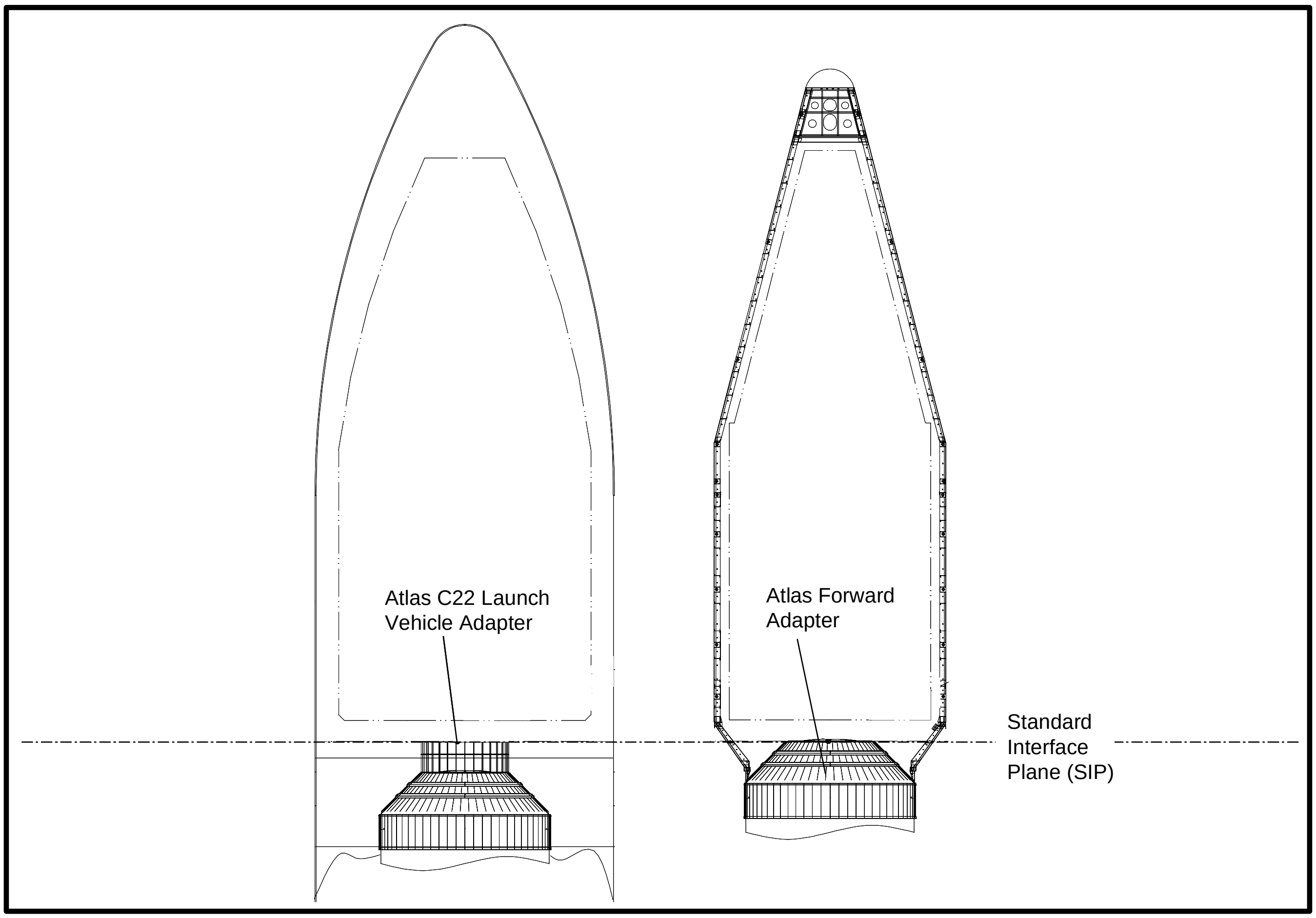


Figure E.1-1 Atlas Standard Interface Plane

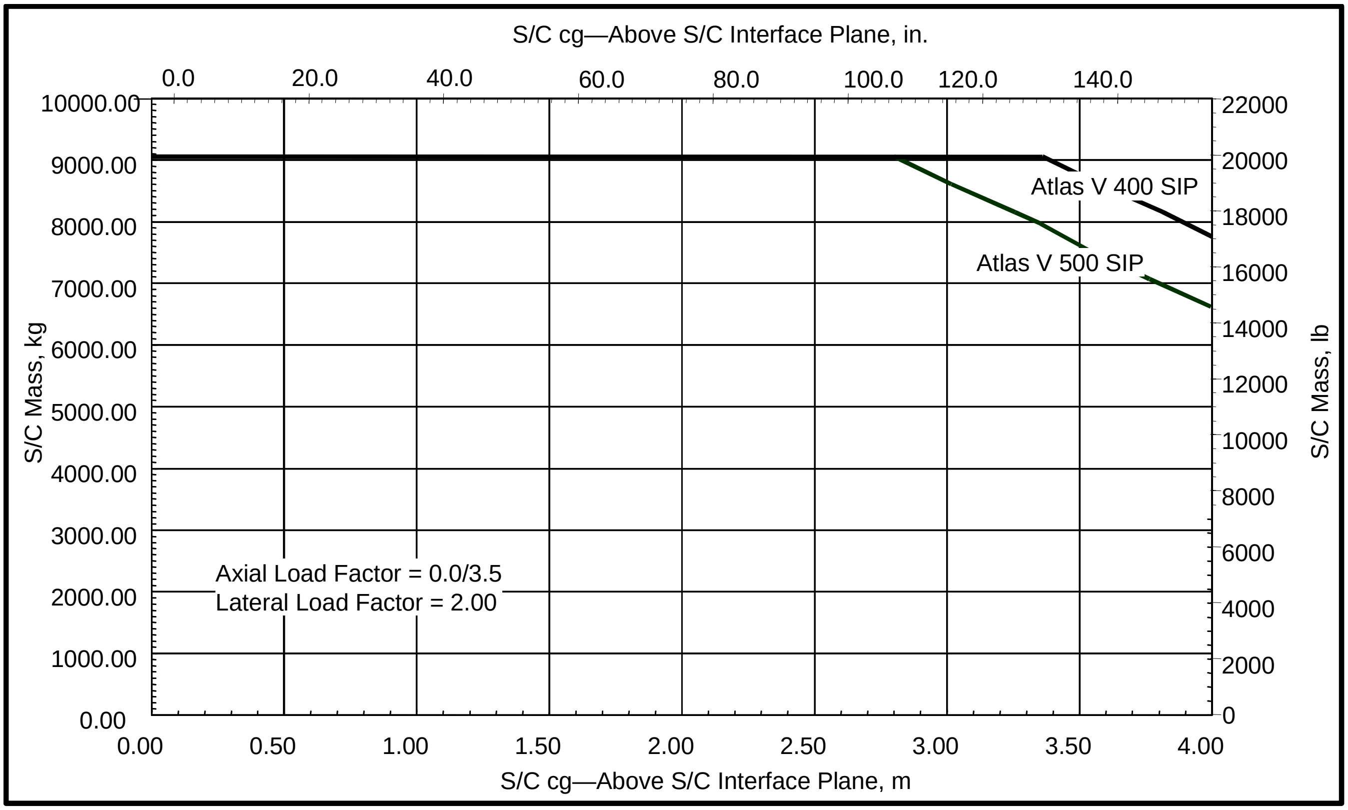


Figure E.1-2 Atlas Standard Interface Plane Structural Capability

Standard Interface Plane Structural Capabilities—Allowable spacecraft weights and longitudinal centers of gravity (cg) for the standard interface plane are shown in Figure E.1-2. These spacecraft mass and cg capabilities were determined using generic spacecraft interface ring geometry and quasi-static load factors shown in Section 3.2.1. Actual spacecraft design allowables may vary depending on interface ring stiffness and results of spacecraft mission-specific coupled loads analyses. Coordination with the Atlas program is required to define appropriate structural capabilities for spacecraft designs that exceed these generic allowables.

Standard Interface Plane Definition—Configuration and dimensional requirements for the Atlas standard interface plane are shown in Figure E.1-3. The hole pattern for this interface is controlled by Atlas-provided tooling. This tooling is made available to customers for fabrication of their matching hardware as a part of mission-integration activities.

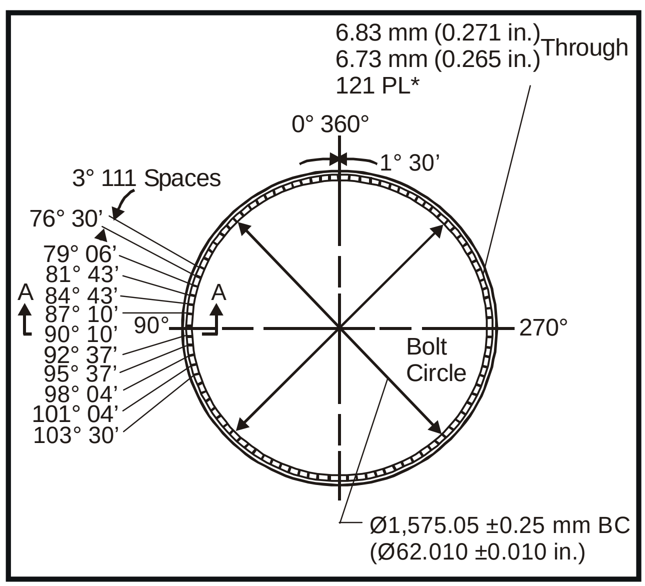


Figure E.1-3 Atlas Standard Interface Plane Dimensional Requirements

Static Payload Envelope—Usable volume for the spacecraft relative to the standard interface plane is defined by the static payload envelope. This envelope represents the maximum allowable spacecraft static dimensions (including manufacturing tolerances) relative to the spacecraft/payload adapter interface. This envelope is designed to allow access to mating components for integration and installation operations and to ensure positive clearance between launch vehicle and spacecraft-provided hardware. Clearance layouts are performed for each spacecraft configuration, and if necessary, critical clearance locations are measured during spacecraft-to-launch-vehicle mate operations to ensure positive clearances. Detailed views of the static payload envelope for the standard interface plane are shown in Figure E.1-4.

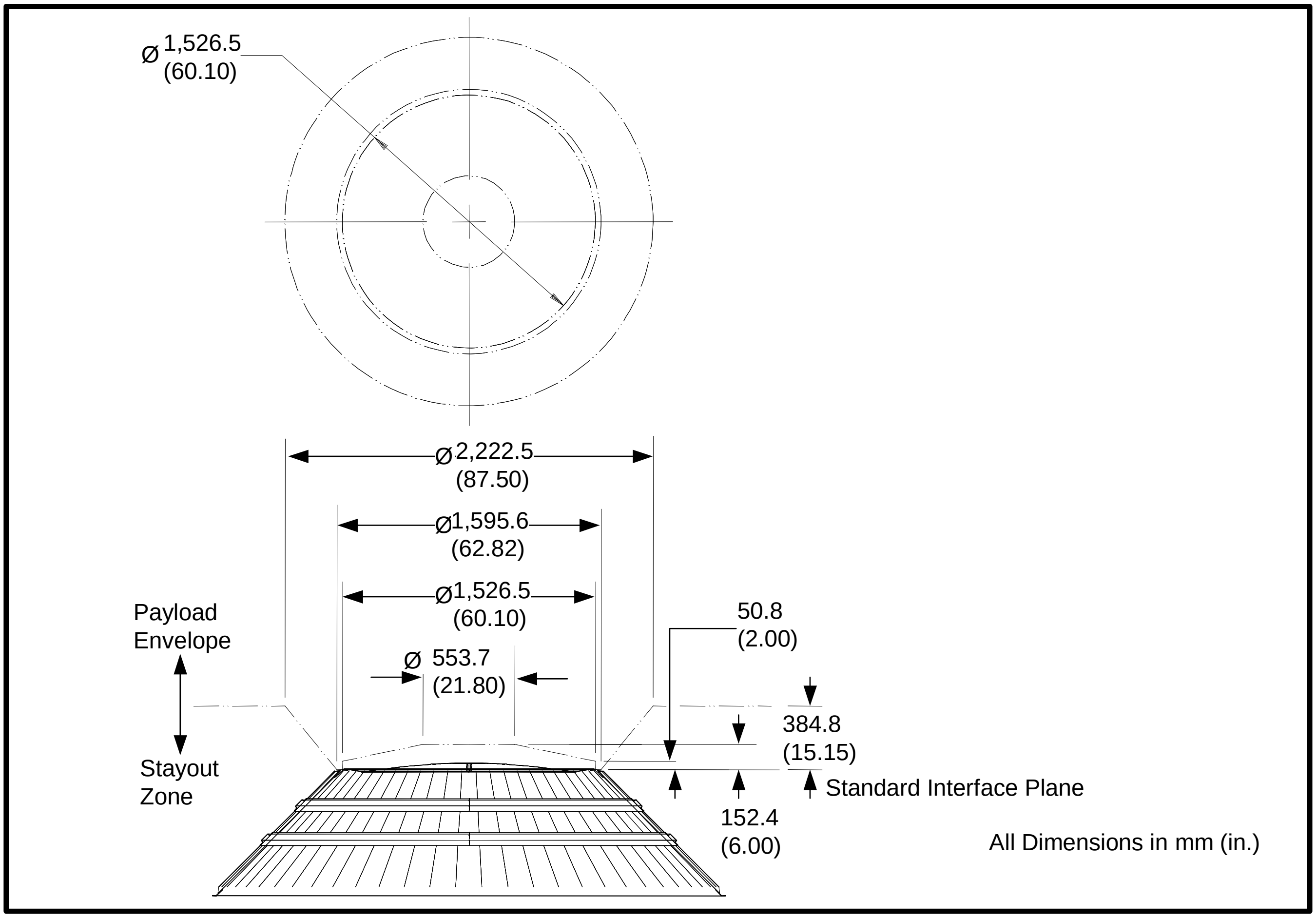


Figure E.1-4 Atlas Standard Interface Plane Static Payload Envelope

E.2 ATLAS LAUNCH VEHICLE ADAPTERS

Atlas launch vehicles adapters were developed to provide a common interface for launch vehicle required ground support equipment that interfaces with payload adapter systems. Atlas launch vehicle adapter characteristics are summarized in Table E.2-1. The launch vehicle adapter is a machined aluminum component in the form of an integrally stiffened cylinder and is available in heights from 330.2 mm (13.00 in.) to 736.6 mm (29.00 in.). Standard configurations available are the C13 (330.2-mm [13.00-in.] high), C15 (384.8-mm [15.15-in.] high), and C22 (558.8-mm [22-in.] high) adapters (Fig. E.2-1).

Table E.2-1 Atlas Launch Vehicle Adapter Characteristics

Atlas Launch Vehicle Adapters		
Construction	Integrally Machined Aluminum Construction	
Mass Properties		
C13	29 kg	64 lb
C15	32 kg	71 lb
C22	44 kg	97 lb
Payload Capability	Figure E.2-2	
	8,000 kg at 3.3 m	17,640 lb at 130 in.

On the Atlas V 500 vehicle, a C22 launch vehicle adapter is mounted to the top of the Atlas forward adapter and provides an interface surface and hole-pattern at its forward end that is compatible with standard interface plane requirements. The C22 adapter is standard with the 5-m payload fairing to allow clearance for launch vehicle ground support equipment. In this configuration, cost and performance impacts of the C22 launch vehicle adapter are considered to be a part of the basic launch vehicle service and are not counted against payload systems weight.

The Atlas program has adapted a modified version of the launch vehicle adapter as a component of Atlas V standard payload adapters. This allowed creation of a modular series of payload adapters that have common interfaces to launch vehicle flight hardware and ground support equipment on a component that is separate from the mission-specific requirements of the spacecraft.

For customers that provide their own payload adapter and payload separation system, launch vehicle adapters are available as a mission-specific option. This allows the customer to raise the position of the standard interface plane relative to the launch vehicle for additional clearance or to take advantage of standard GSE interfaces that are built into the launch vehicle adapter. The Atlas program is also developing lighter weight versions of these adapters, with reduced structural capability, for lighter weight spacecraft. Information on these adapters is available to customers on request.

Launch Vehicle Adapter Structural Capabilities—Allowable spacecraft weights and longitudinal centers of gravity for launch vehicle adapters are shown in Figure E.2-2. These spacecraft

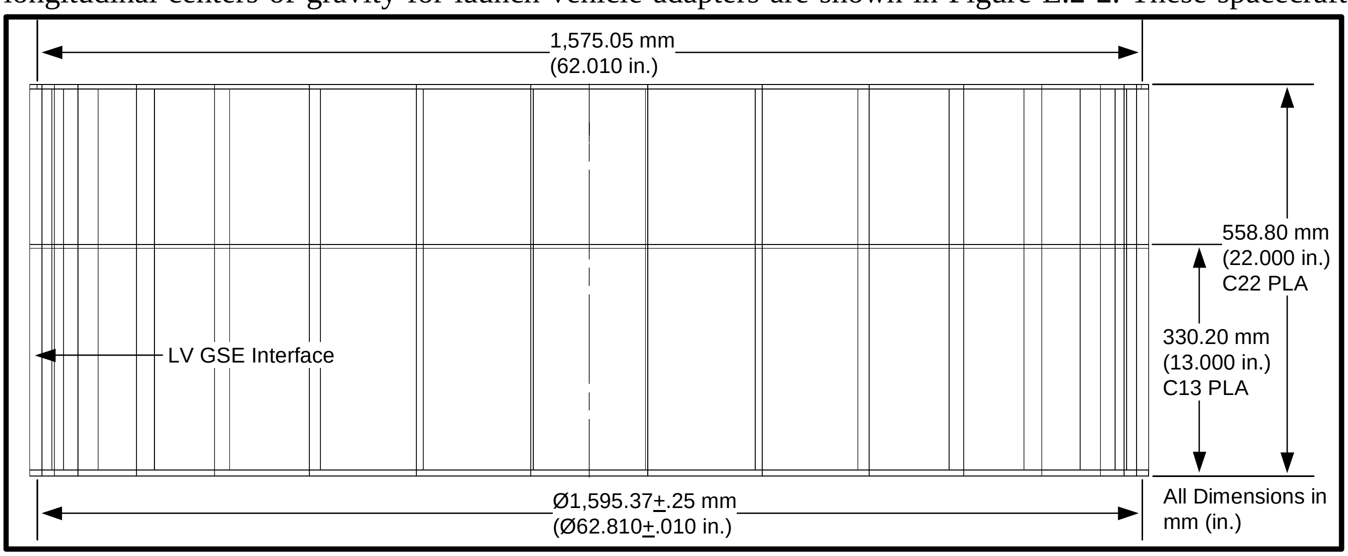


Figure E.2-1 Atlas Launch Vehicle Adapters

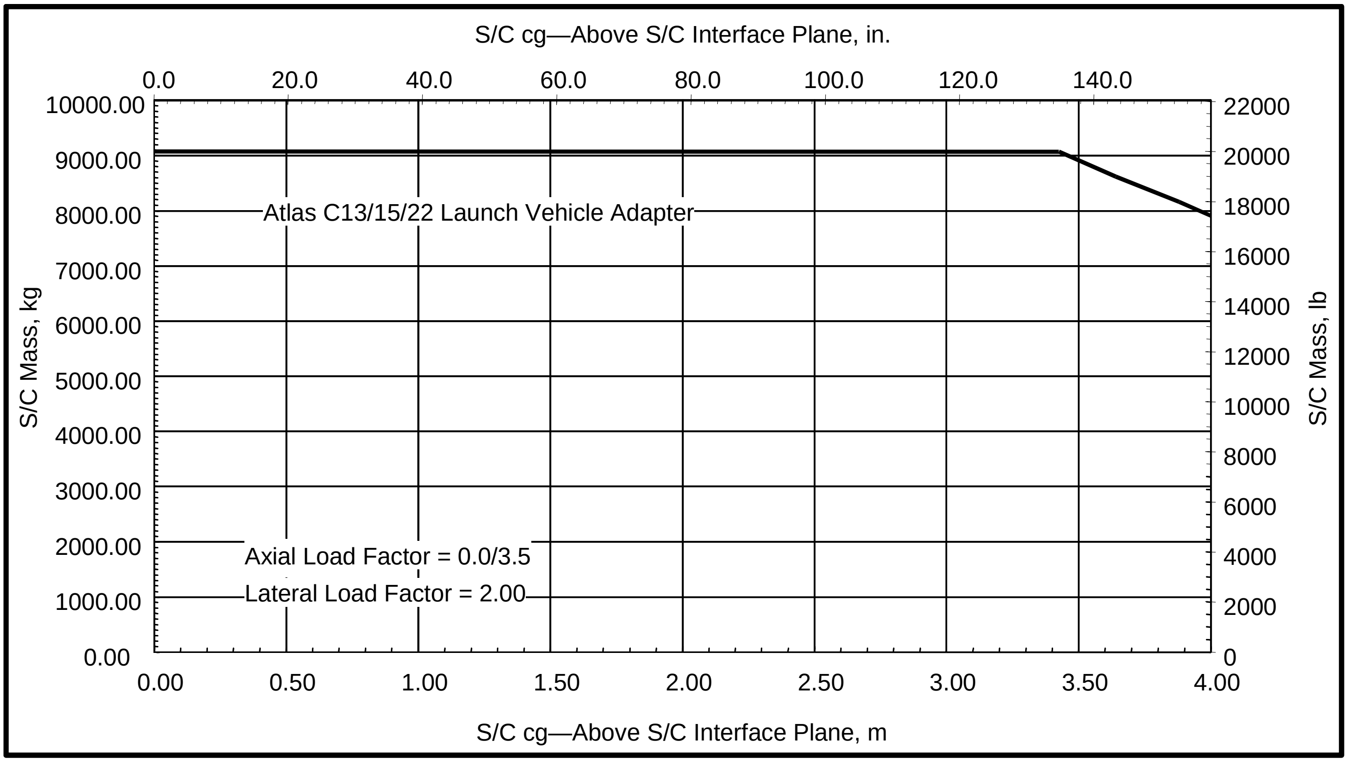


Figure E.2-2 Atlas Launch Vehicle Adapter Structural Capability

mass and center of gravity capabilities were determined using generic spacecraft interface ring geometry and quasi-static load factors shown in Section 3.2.1. Actual spacecraft design allowables may vary depending on interface ring stiffness and results of spacecraft mission-specific coupled loads analyses. Coordination with the Atlas program is required to define appropriate structural capabilities for spacecraft designs that exceed these generic allowables.

Launch Vehicle Adapter Interfaces—Configuration and dimensional requirements for Atlas launch vehicle adapters interface are shown in Figure E.2-3. The hole pattern for this interface is controlled by Atlas-provided tooling. This tooling is made available to customers for fabrication of their matching hardware as a part of mission integration activities. Alternative hole patterns for this interface can be incorporated on a mission-specific basis.

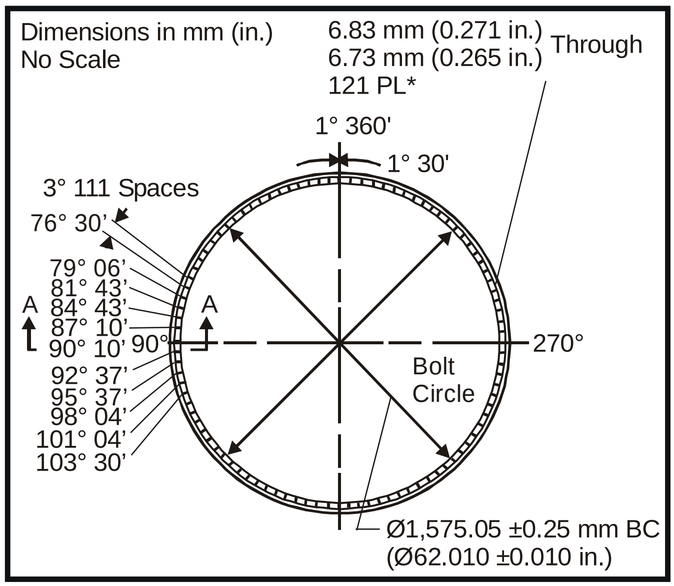


Figure E.2-3 Atlas Launch Vehicle Adapter Interface Requirements

E.3 ATLAS TYPE A PAYLOAD ADAPTER

The Atlas Type A payload adapter is designed to support spacecraft with an aft ring diameter of 937 mm (37 in.). Major characteristics of this payload adapter are summarized in Table E.3-1. This adapter is an aluminum skin, stringer, and frame construction with machined forward and aft rings that mate to the spacecraft and launch vehicle forward adapter (Fig. E.3-1). The forward ring has an outer diameter of 945.3 mm (37.215 in.) and forms the spacecraft separation plane. The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 121 holes that match up with the Atlas standard interface plane requirements. The nominal height of the Type A payload adapter is 762 mm (30.00 in.). The Type A payload adapter supports all hardware that directly interfaces with the spacecraft including the payload separation system, electrical connectors, and mission-specific options and includes all provisions for mating to the launch vehicle ground support equipment, including the torus arms and isolation diaphragm, used during ground processing operations.

Table E.3-1 Atlas Type A Payload Adapter Characteristics

Atlas Type A Payload Adapter		
Construction	Aluminum Skin/Stringer/Frame Construction	
Mass Properties	44 kg	97 lb
Payload Capability	Figure E.3-3	
	2,100 kg at 1.27 m	4,600 lb at 50 in.
P/L Separation Sys	PSS37	
Max Shock Levels	Section 3.2.4	
Clampband Preload—Installation	23.7 ± 0.1 kN	5,328 ± 22 lb
Clampband Preload—Flight	23.0 ± 0.5 kN	5,170 ± 112 lb
Separation Springs		
Number	4	
Force per Spring—Max	1 kN	225 lb

Payload Separation System—The Atlas Type A payload adapter uses a launch vehicle-provided Marmon-type clampband payload separation system. This separation system (Fig. E.3-2) consists of a clampband set, release mechanism, and separation springs. The clampband set consists of a clampband for holding the spacecraft and adapter rings together plus devices to extract, catch, and retain the

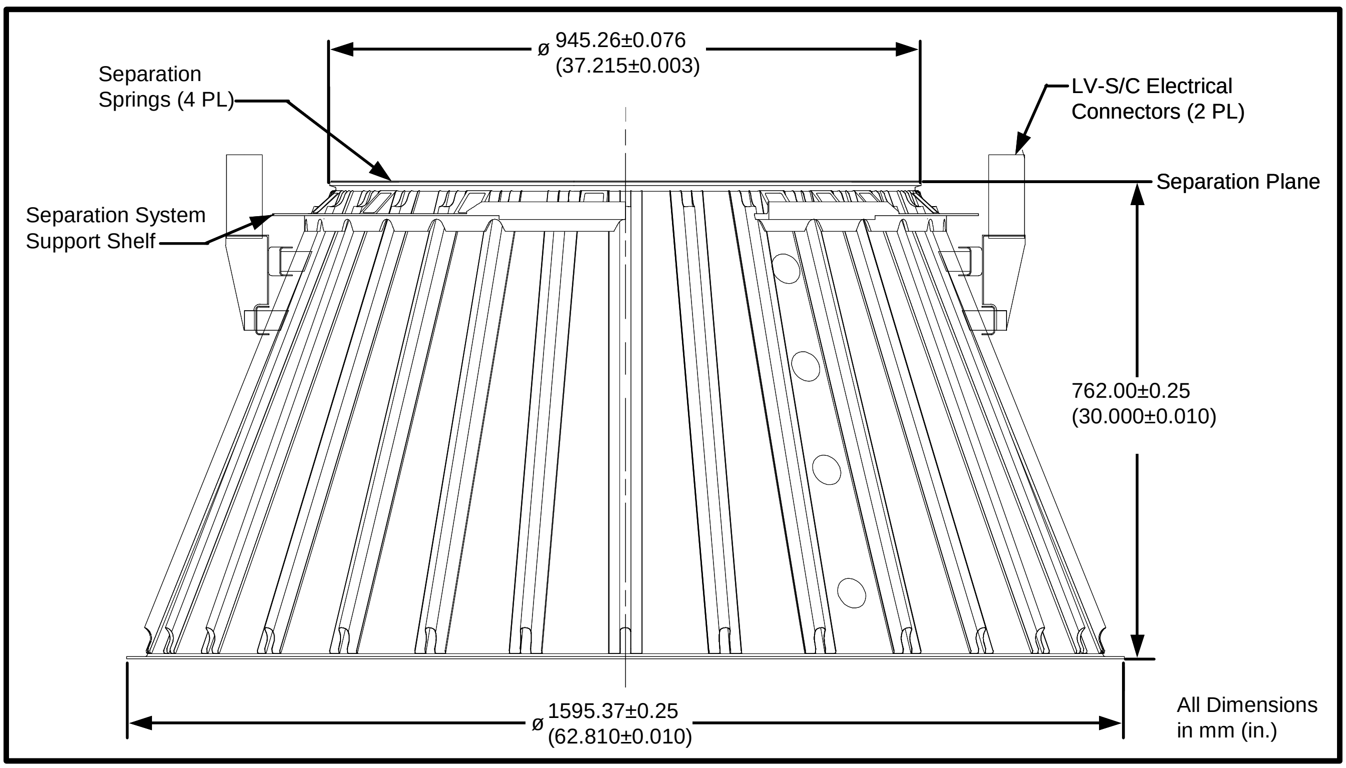


Figure E.3-1 Atlas Type A Payload Adapter

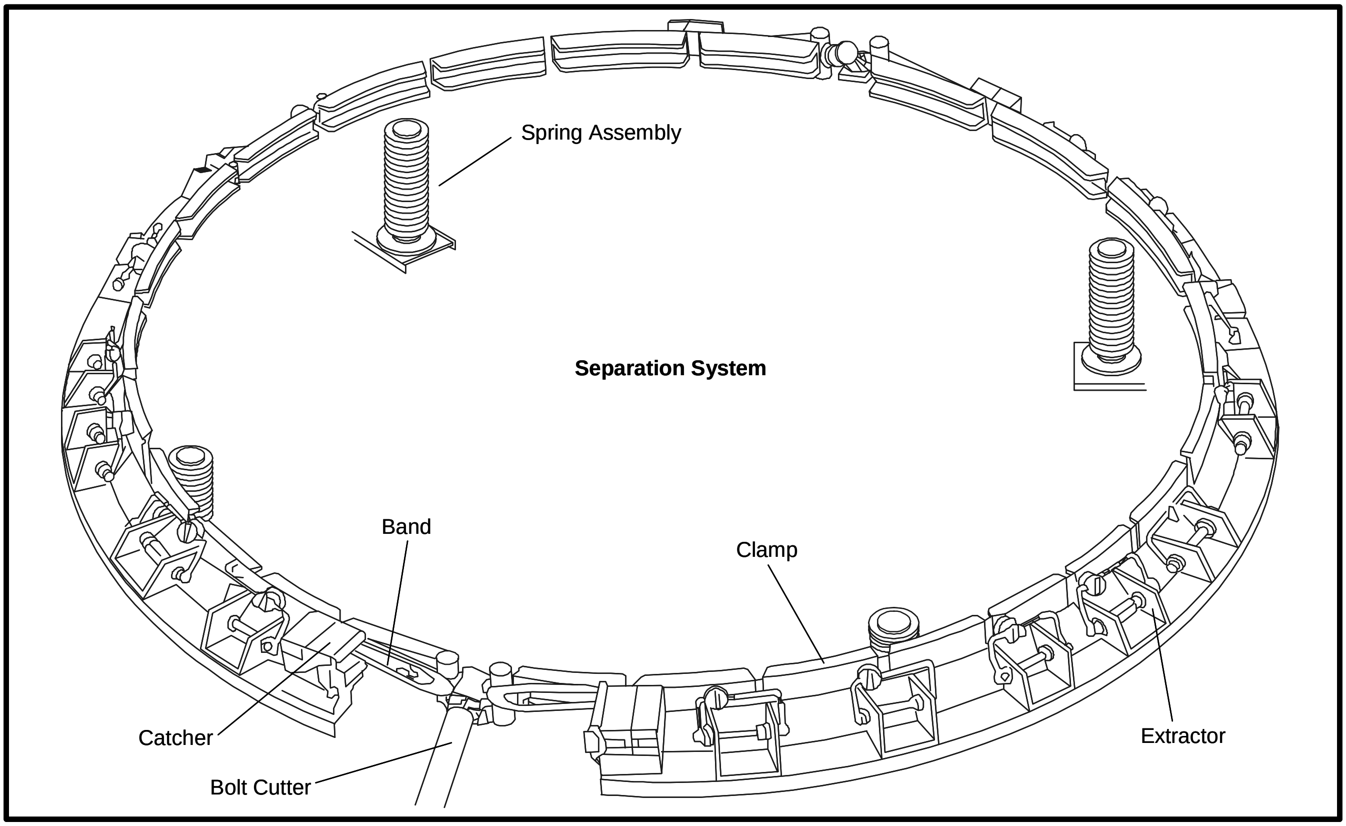


Figure E.3-2 PSS37 Payload Separation System

clampband on the adapter structure after separation. The clampband includes aluminum clamp segments that hold the payload adapter and spacecraft rings together and a two-piece stainless-steel retaining band that holds the clamp segments in place. The ends of the retaining band are held together by tension bolts. For separation, a pyrotechnically activated bolt-cutter severs the tension bolts allowing the end of the clamp segments to move apart and release the payload adapter and spacecraft mating rings.

Separation spring assemblies provide the necessary separation energy after the clampband is released and are mounted to the payload adapter forward ring and bear on the spacecraft aft ring. Positive spacecraft separation is detected through continuity loops installed in the spacecraft electrical connector and wired to the upper-stage instrumentation for monitoring and telemetry verification.

Payload Adapter Structural Capabilities—Allowable spacecraft weights and longitudinal centers of gravity for the Type A payload adapter/separation systems are shown in Figure E.3-3. These spacecraft mass and cg capabilities were determined using generic spacecraft interface ring geometry as shown in Figure E.3-4, and quasi-static load factors shown in Section 3.2.1. Actual spacecraft design allowables may vary depending on interface ring stiffness and results of spacecraft mission-specific coupled loads analyses. Coordination with the Atlas program is required to define appropriate structural capabilities for spacecraft designs that exceed these generic allowables.

Payload Adapter Interfaces—The primary structural interface between the launch vehicle and spacecraft occurs at the payload adapter forward ring. This ring interfaces with the spacecraft aft ring and a payload separation system holds the two rings together for the structural joint and provides the release mechanism for spacecraft separation. Electrical bonding is provided across all interface planes associated with these components. The payload adapter also provides mounting provisions for separation springs and supports interfacing components for electrical connectors between the launch vehicle and spacecraft. Interface requirements for these components are shown in Figures E.3-4 and

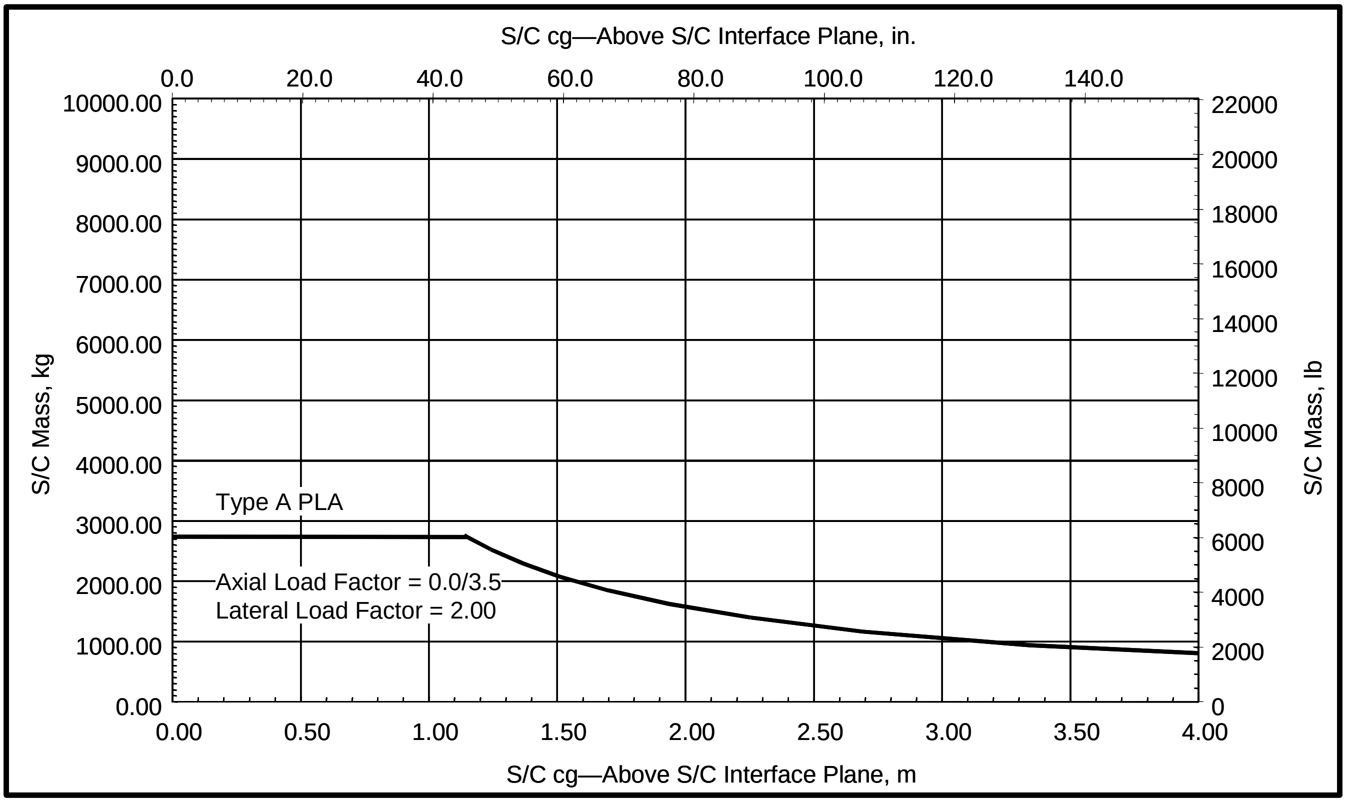


Figure E.3-3 Atlas Type A Payload Adapter Structural Capability

E.3-5. Additional mission-specific provisions, including spacecraft purge provisions, spacecraft range safety destruct units, and mission satisfaction kit instrumentation, may be added as necessary.

Static Payload Envelope—The usable volume for the spacecraft relative to the payload adapter is defined by the static payload envelope. This envelope represents the maximum allowable spacecraft static dimensions (including manufacturing tolerances) relative to the spacecraft/payload adapter interface. This envelope is designed to allow access to mating components and payload separation system for integration and installation operations, motion of the payload separation system during its operation, and the movement of the spacecraft and launch vehicle after separation of the spacecraft and payload. Clearance layouts and separation analyses are performed for each spacecraft configuration, and if necessary, critical clearance locations are measured during spacecraft-to-payload-adapter mate operations to ensure positive clearance during flight and separation. Detailed views of the static payload envelope for the Type A payload adapter are shown in Figure E.3-6.

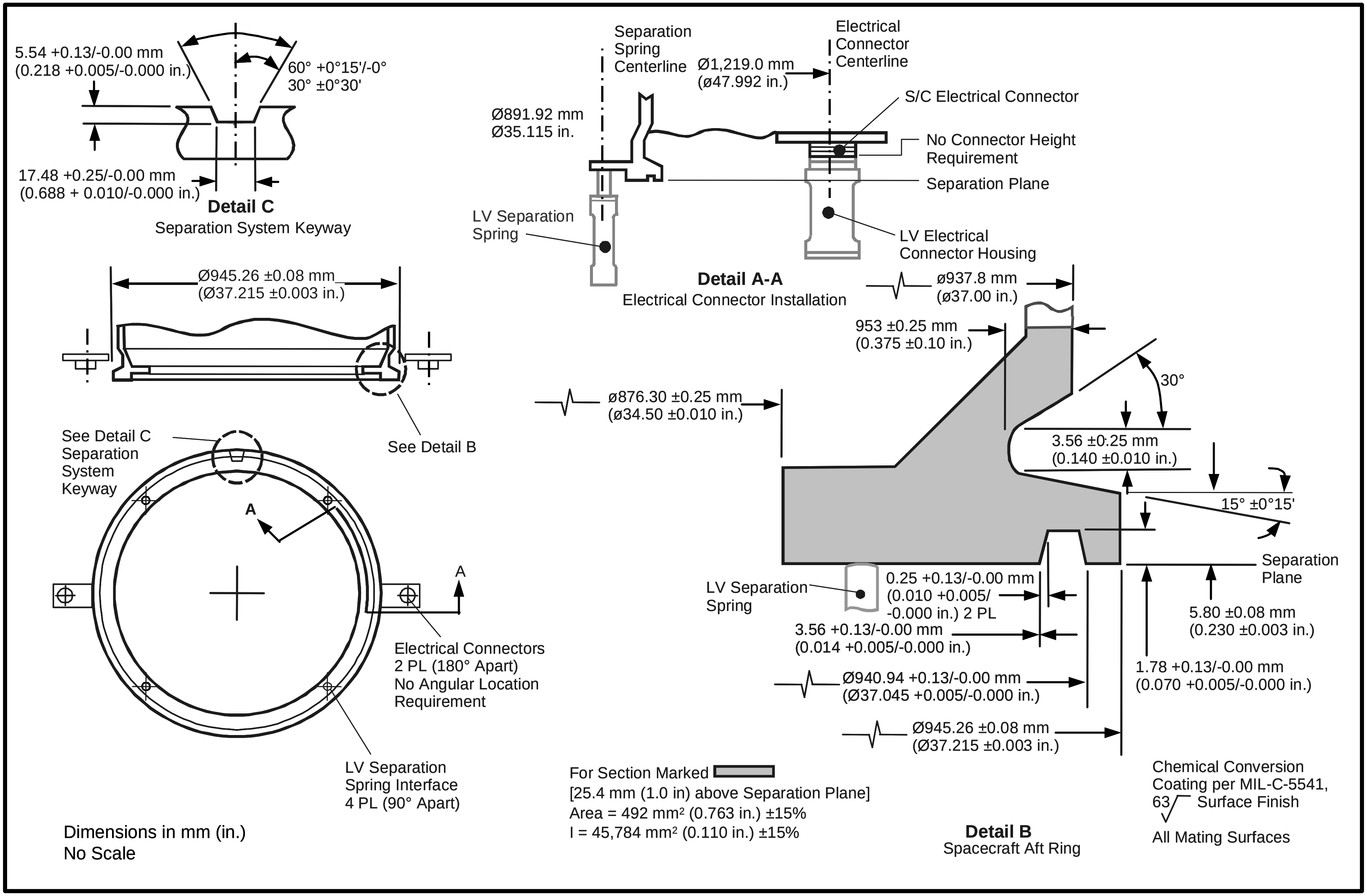


Figure E.3-4 Spacecraft Interface Requirements

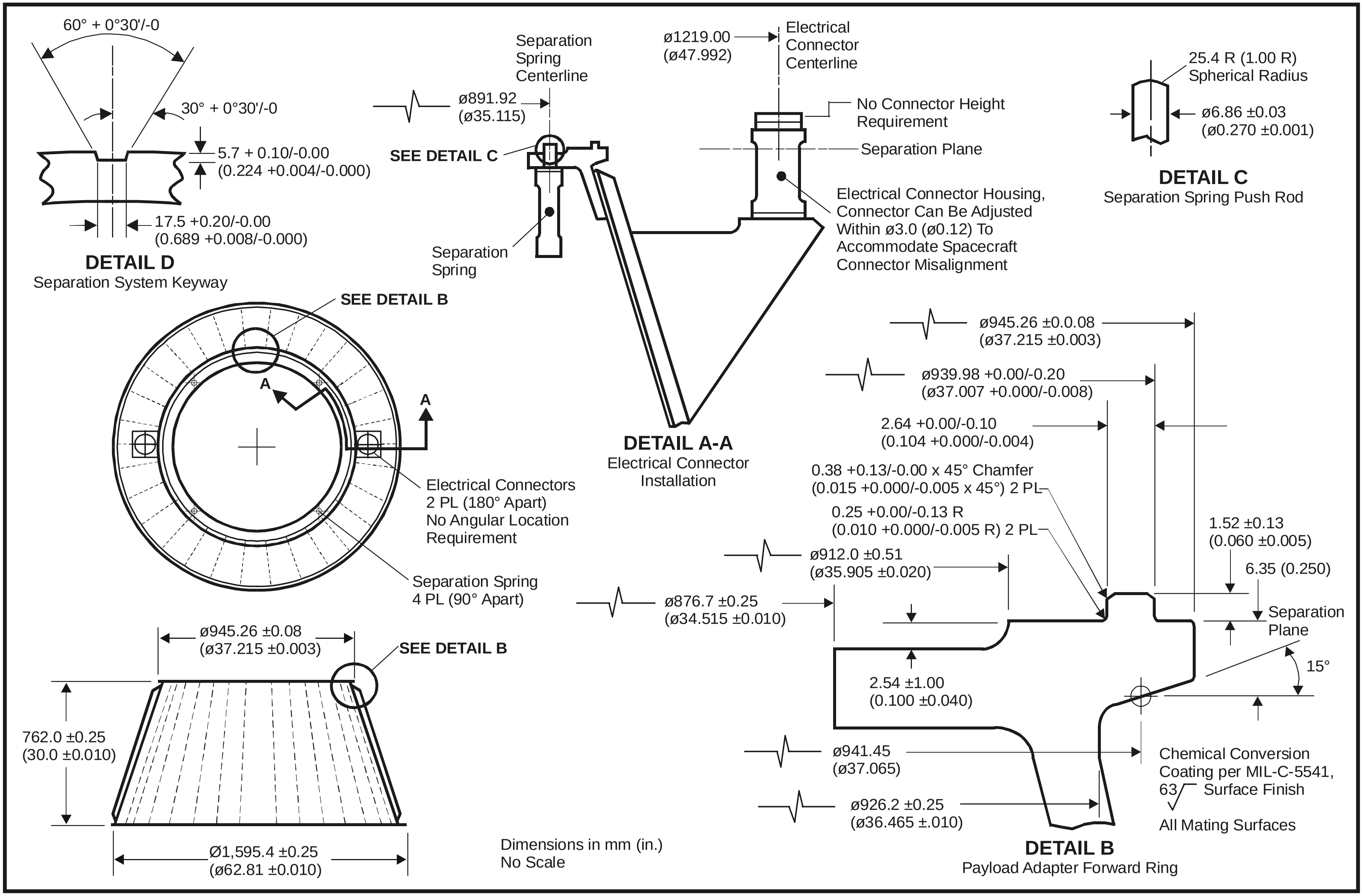


Figure E.3-5 Type A Payload Adapter Interface Requirements

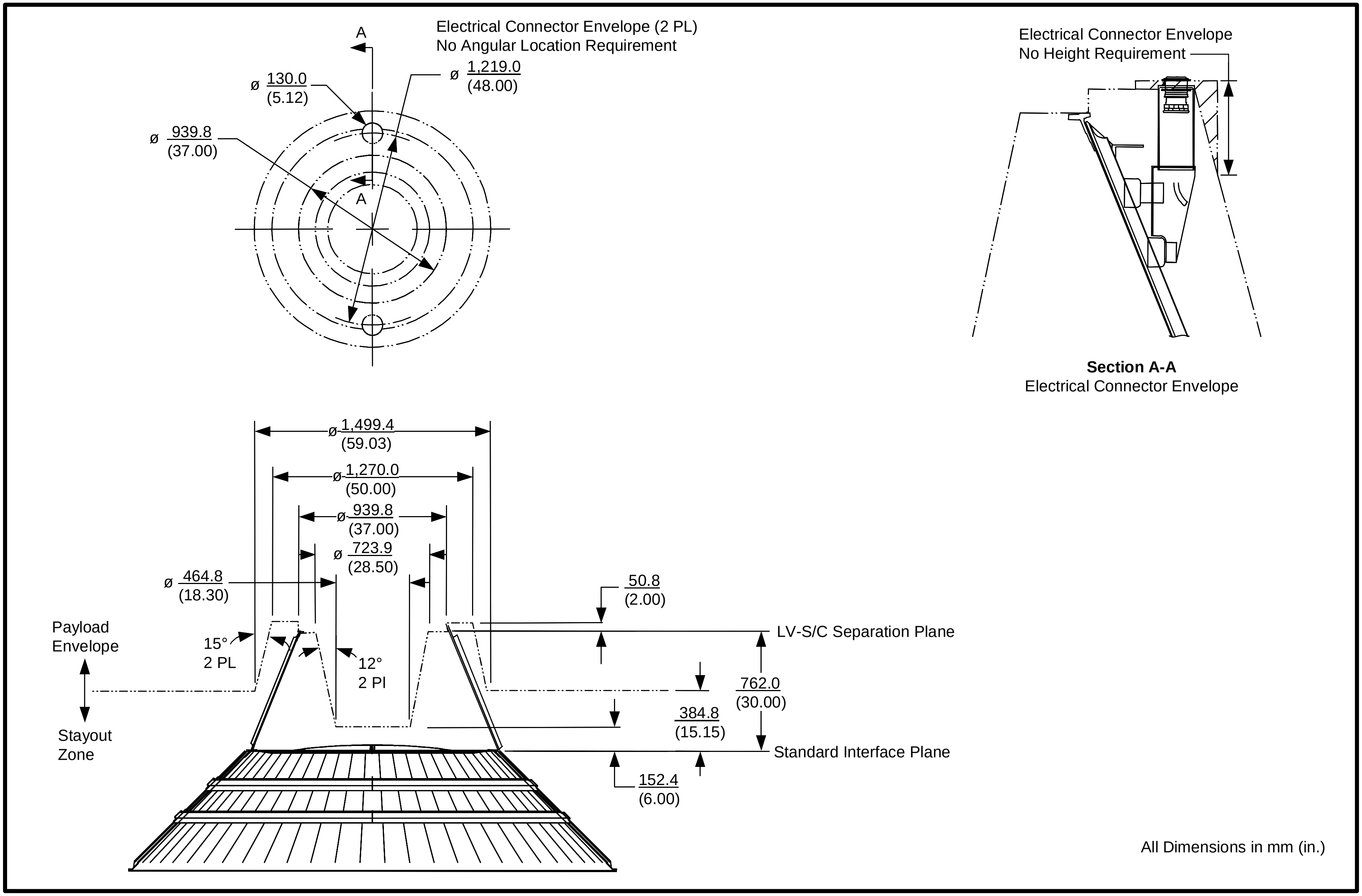


Figure E.3-6 Atlas Type A Payload Adapter Static Payload Envelope

E.4 ATLAS TYPE B1 PAYLOAD ADAPTER

The Atlas Type B1 payload adapter is designed to support spacecraft with an aft ring diameter of 1,194 mm (47 in.). Major characteristics of this payload adapter are summarized in Table E.4-1. This adapter is an aluminum skin, stringer, and frame construction with machined forward and aft rings that mate to the spacecraft and launch vehicle forward adapter (Fig. E.4-1). The forward ring has an outer diameter of 1,215 mm (47.835 in.) and forms the spacecraft separation plane. The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 121 holes that match up with Atlas standard interface plane requirements. The nominal height of the Type B1 payload adapter is 812.8 mm (32.00 in.). The Type B1 payload adapter supports all hardware that directly interfaces with spacecraft including the payload separation system, electrical connectors, and mission-specific options and includes all provisions for mating to the launch vehicle ground support equipment, including the torus arms and isolation diaphragm, used during ground processing operations. The currently available Type B1 payload adapter will be phased out and replaced by the higher capability, lower shock Type B1194VS payload adapter in the near future.

Table E.4-1 Atlas Type B1 Payload Adapter Characteristics

Atlas Type B1 Payload Adapter		
Construction	Aluminum Skin/Stringer/Frame Construction	
Mass Properties	52.6 kg	116 lb
Payload Capability	Figure E.4-3	
	3,800 kg at 1.53 m 8,383 lb at 60 in.	
P/L Separation Sys	PSS47	
Max Shock Levels	Section 3.2.4	
Clampband Preload—Installation	27.8 ± 0.1 kN	6,250 ± 22 lb
Clampband Preload—Flight	27.1 ± 0.5 kN	6,092 ± 112 lb
Separation Springs		
Number	4	
Force per Spring—Max	1 kN	225 lb

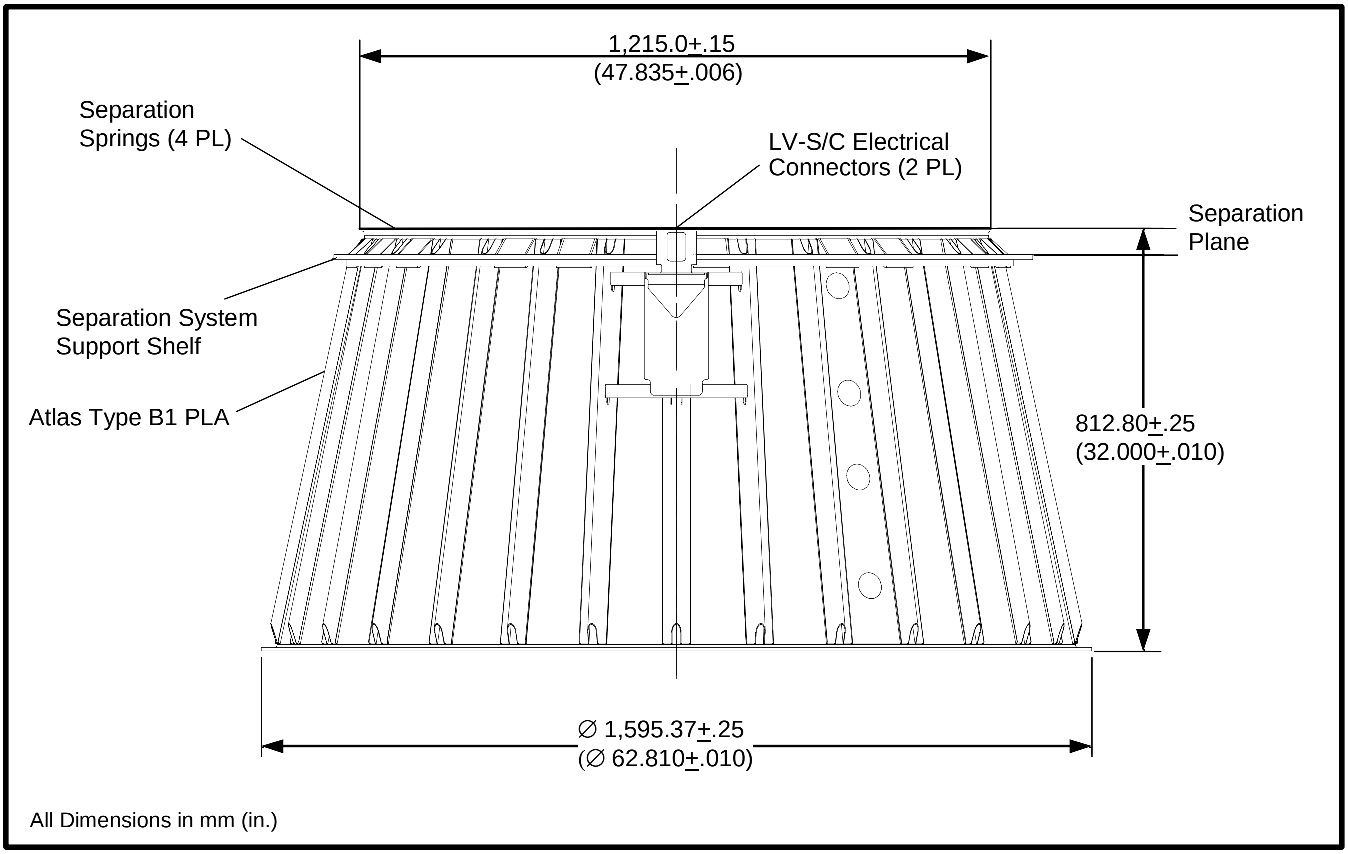


Figure E.4-1 Atlas Type B1 Payload Adapter

Payload Separation System—The Atlas Type B1 payload adapter uses a launch vehicle-provided Marmon-type clampband payload separation system. This separation system (Fig. E.4-2) consists of a clampband set, release mechanism, and separation springs. The clampband set consists of a clampband for holding the spacecraft and adapter rings together plus devices to extract, catch, and retain the clampband on the adapter structure after separation. The clampband includes aluminum clamp segments that hold the payload adapter and spacecraft rings together and a two-piece stainless-steel retaining band that holds the clamp segments in place. The ends of the retaining band are held together by tension bolts. For separation, a pyrotechnically activated bolt-cutter severs the tension bolts allowing the end of the clamp segments to move apart and release the payload adapter and spacecraft mating rings.

Separation spring assemblies provide the necessary separation energy after the clampband is released and are mounted to the payload adapter forward ring and bear on the spacecraft aft ring. Positive spacecraft separation is detected through continuity loops installed in the spacecraft electrical connector and wired to the upper-stage instrumentation for monitoring and telemetry verification.

Payload Adapter Structural Capabilities—Allowable spacecraft weights and longitudinal centers of gravity for the Type B1 payload adapter/separation systems are shown in Figure E.4-3. These spacecraft mass and cg capabilities were determined using generic spacecraft interface ring geometry as shown in Figure E.4-4, and quasi-static load factors shown in Section 3.2.1. Actual spacecraft design allowables may vary depending on interface ring stiffness and results of spacecraft mission-specific coupled loads analyses. Coordination with the Atlas program is required to define appropriate structural capabilities for spacecraft designs that exceed these generic allowables.

Payload Adapter Interfaces—The primary structural interface between the launch vehicle and spacecraft occurs at the payload adapter forward ring. This ring interfaces with the spacecraft aft ring and a payload separation system holds the two rings together for the structural joint and provides the

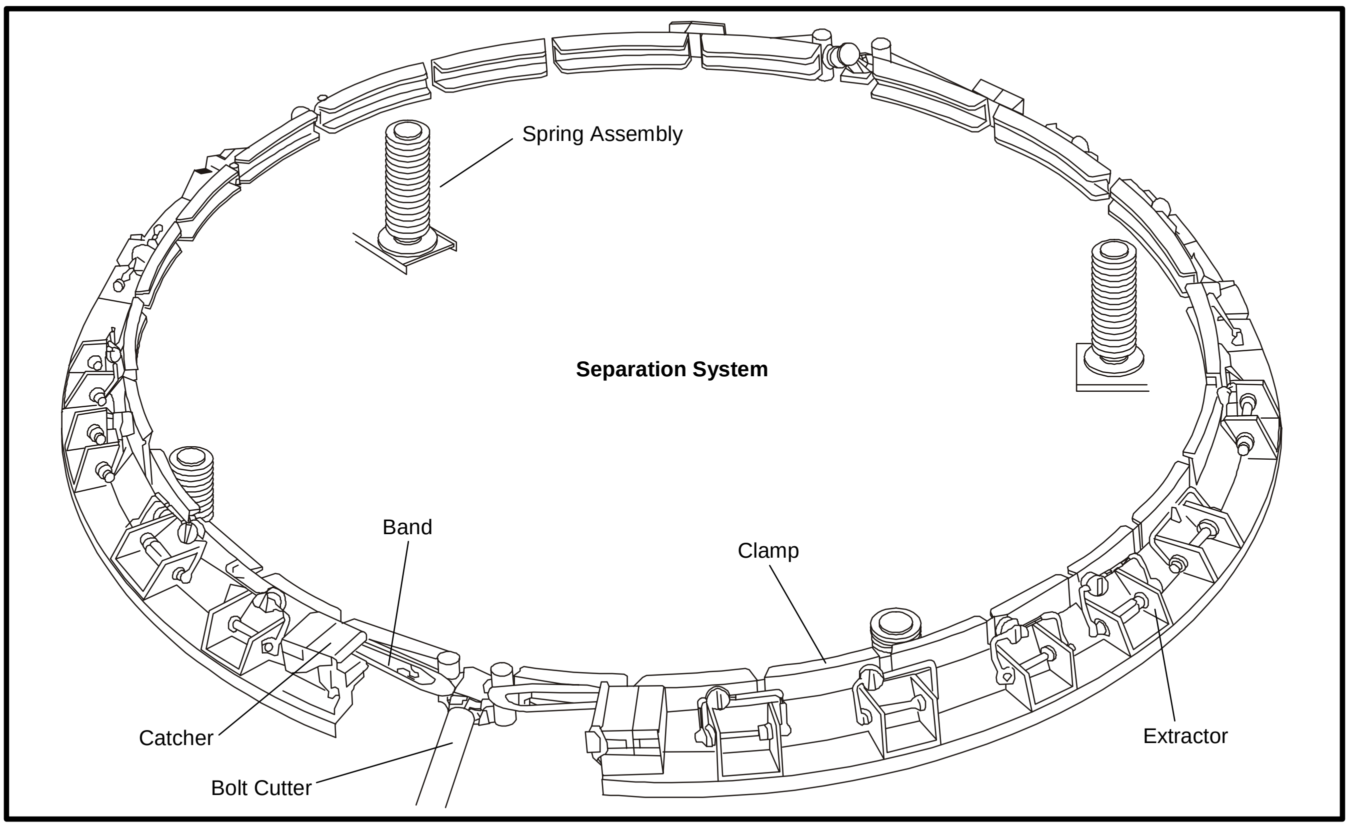


Figure E.4-2 PSS47 Payload Separation System

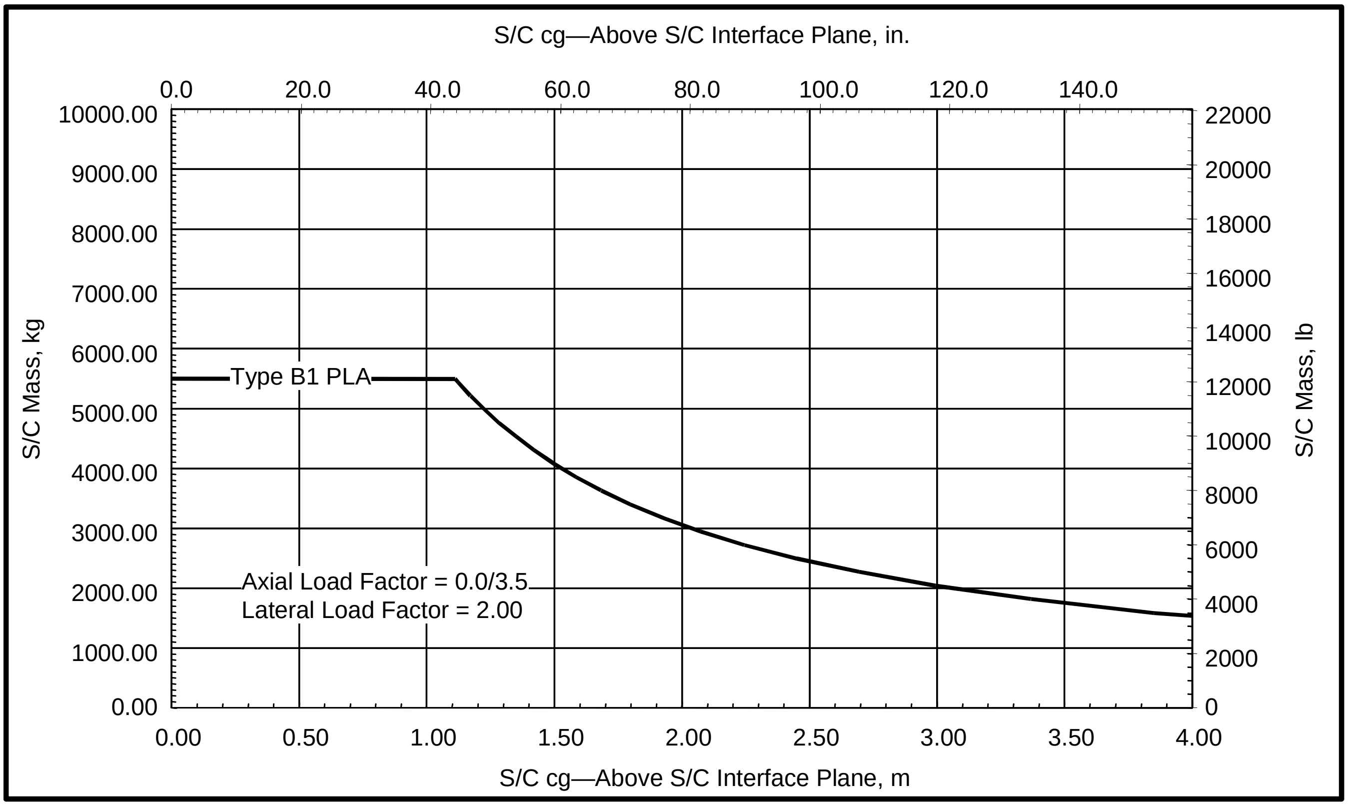


Figure E.4-3 Atlas Type B1 Payload Adapter Structural Capability

release mechanism for spacecraft separation. Electrical bonding is provided across all interface planes associated with these components. The payload adapter also provides mounting provisions for separation springs and supports interfacing components for electrical connectors between the launch vehicle and spacecraft. Interface requirements for these components are shown in Figures E.4-4 and E.4-5. Additional mission-specific provisions, including spacecraft purge provisions, spacecraft range safety destruct units, and mission satisfaction kit instrumentation, may be added as necessary.

Static Payload Envelope—The usable volume for the spacecraft relative to the payload adapter is defined by the static payload envelope. This envelope represents the maximum allowable spacecraft static dimensions (including manufacturing tolerances) relative to the spacecraft/payload adapter interface. This envelope is designed to allow access to mating components and payload separation system for integration and installation operations, motion of the payload separation system during its operation, and movement of the spacecraft and launch vehicle after separation of the spacecraft and payload. Clearance layouts and separation analyses are performed for each spacecraft configuration, and if necessary, critical clearance locations are measured during spacecraft-to-payload-adapter mate operations to ensure positive clearance during flight and separation. Detailed views of the static payload envelope for the Type B1 payload adapter are shown in Figure E.4-6.

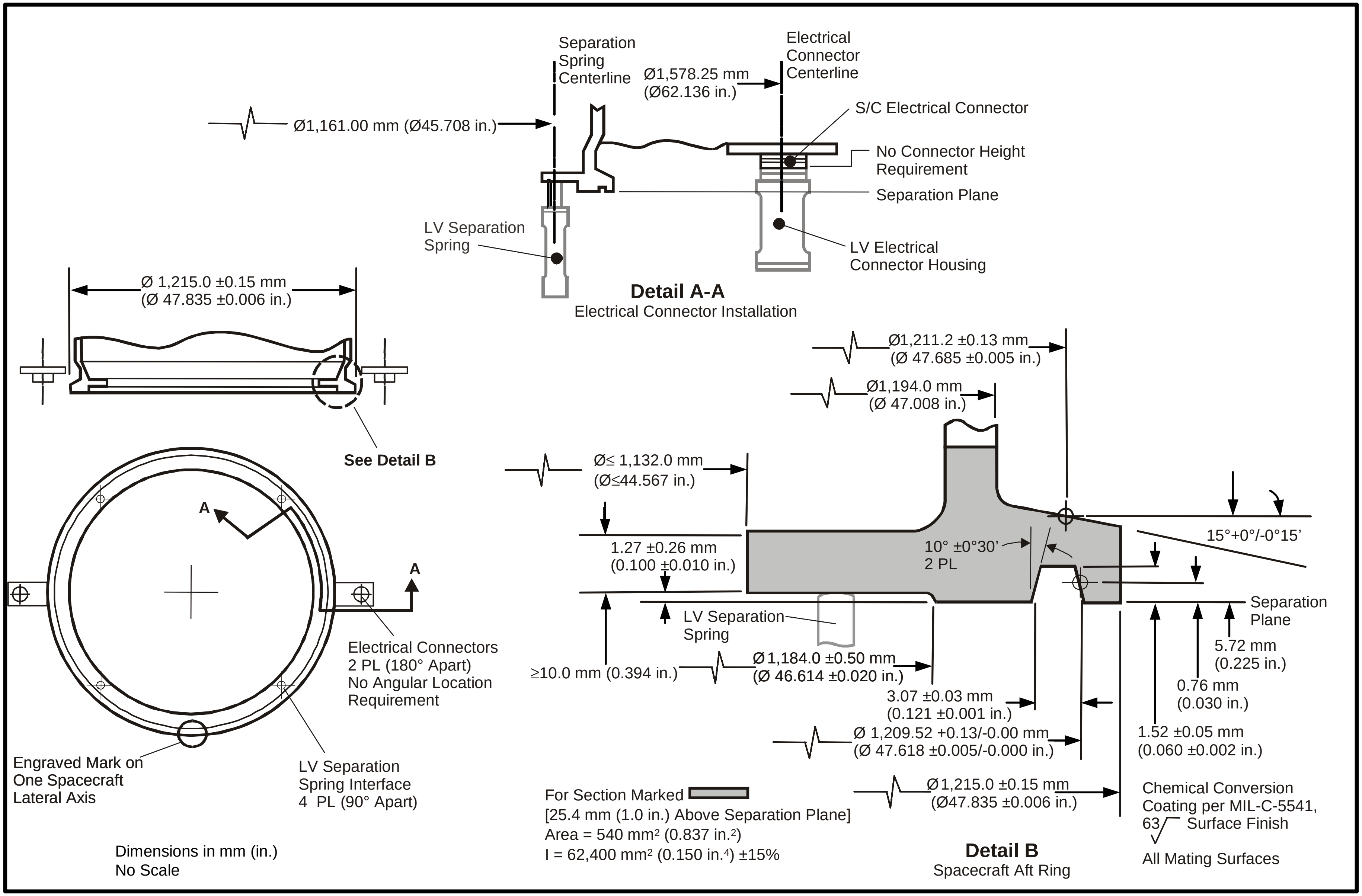


Figure E.4-4 Spacecraft Interface Requirements

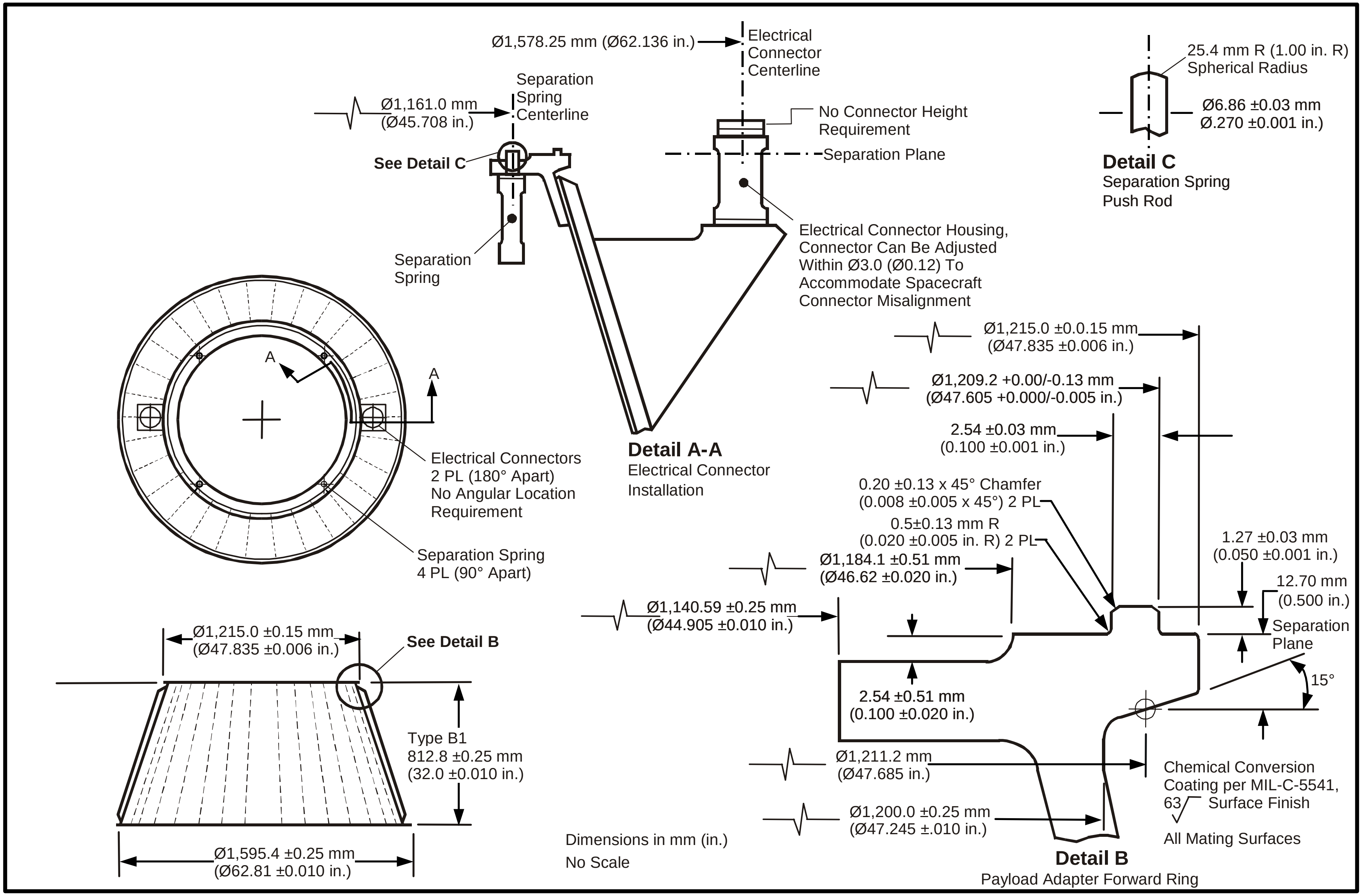


Figure E.4-5 Atlas Type B1 Payload Adapter Interface Requirements

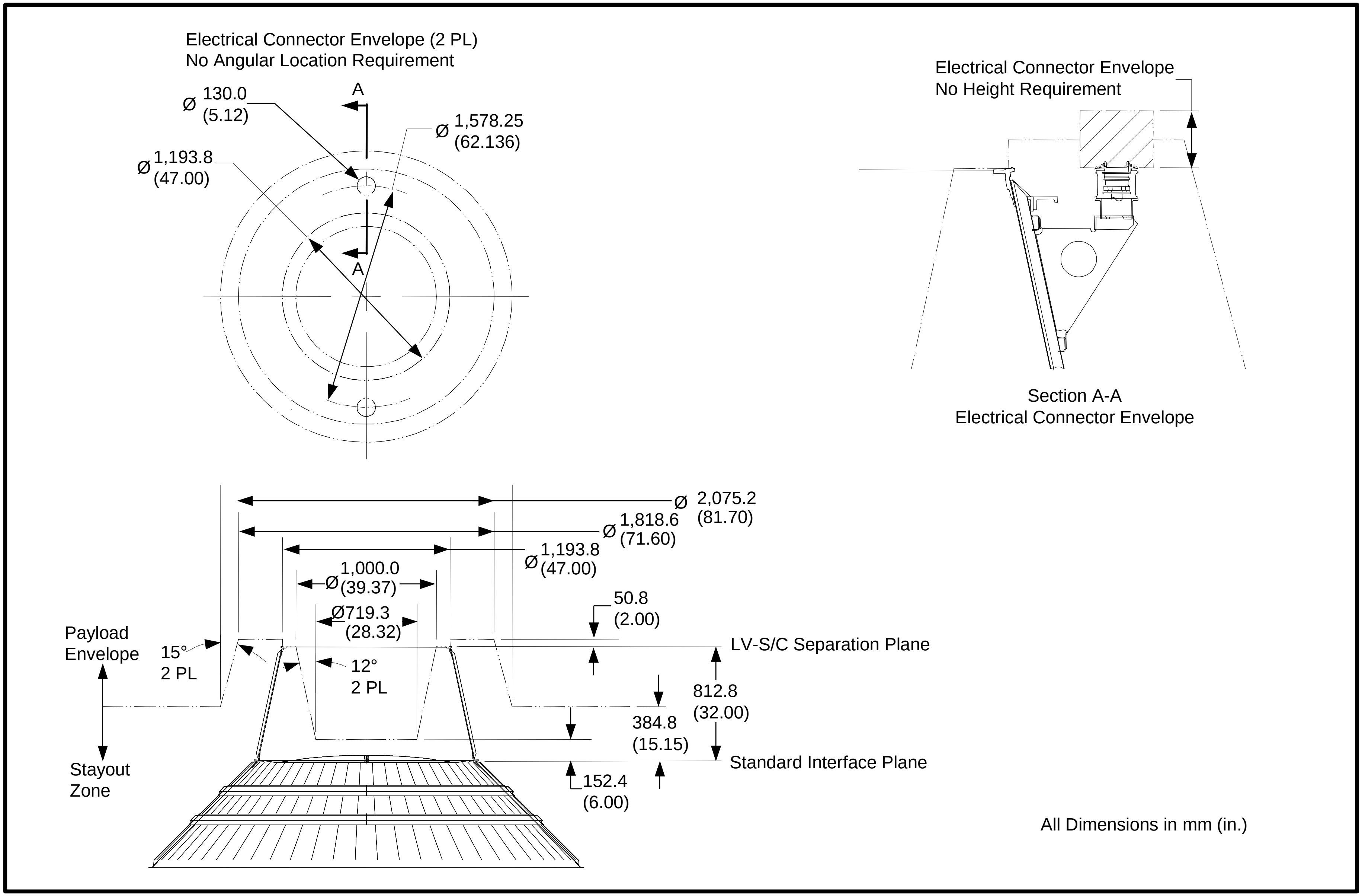


Figure E.4-6 Atlas Type B1 Payload Adapter Static Payload Envelope

E.5 ATLAS TYPE D PAYLOAD ADAPTER

The Atlas Type D payload adapter is designed to support spacecraft with an aft ring diameter of 1,666 mm (66 in.). Major characteristics of this payload adapter are summarized in Table E.5-1. This adapter is an aluminum skin, stringer, and frame construction with machined forward and aft rings that mate to the spacecraft and launch vehicle forward adapter (Fig. E.5-1). The forward ring has an outer diameter of 1,666.1 mm (65.594 in.) and forms the spacecraft separation plane. The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 121 holes that match up with Atlas standard interface plane requirements. The nominal height of the Type D payload adapter is 889 mm (35.00 in.). The Type D payload adapter supports all hardware that directly interfaces with spacecraft including the payload separation system, electrical connectors, and mission-specific options and includes all provisions for mating to the launch vehicle ground support equipment, including the torus arms and isolation diaphragm, used during ground processing operations. The current Type D payload adapter will be phased out and replaced by the higher capability, lower shock Type D1666VS payload adapter in the near future.

Table E.5-1 Atlas Type D Payload Adapter Characteristics

Atlas Type D Payload Adapter		
Construction	Aluminum Skin/Stringer/Frame Construction	
Mass Properties	54 kg	119 lb
Payload Capability	Figure E.5-3	
	3,492 kg at 1.02 m	7,700 lb at 40 in.
P/L Separation Sys	PSS66	
Max Shock Levels	Section 3.2.4	
Clampband Preload—Installation	30.7 ± 0.1 kN	6,902 ± 22 lb
Clampband Preload—Flight	30.0 ± 0.5 kN	6,744 ± 122 lb
Separation Springs		
Number	6	
Force per Spring—Max	1 kN	225 lb

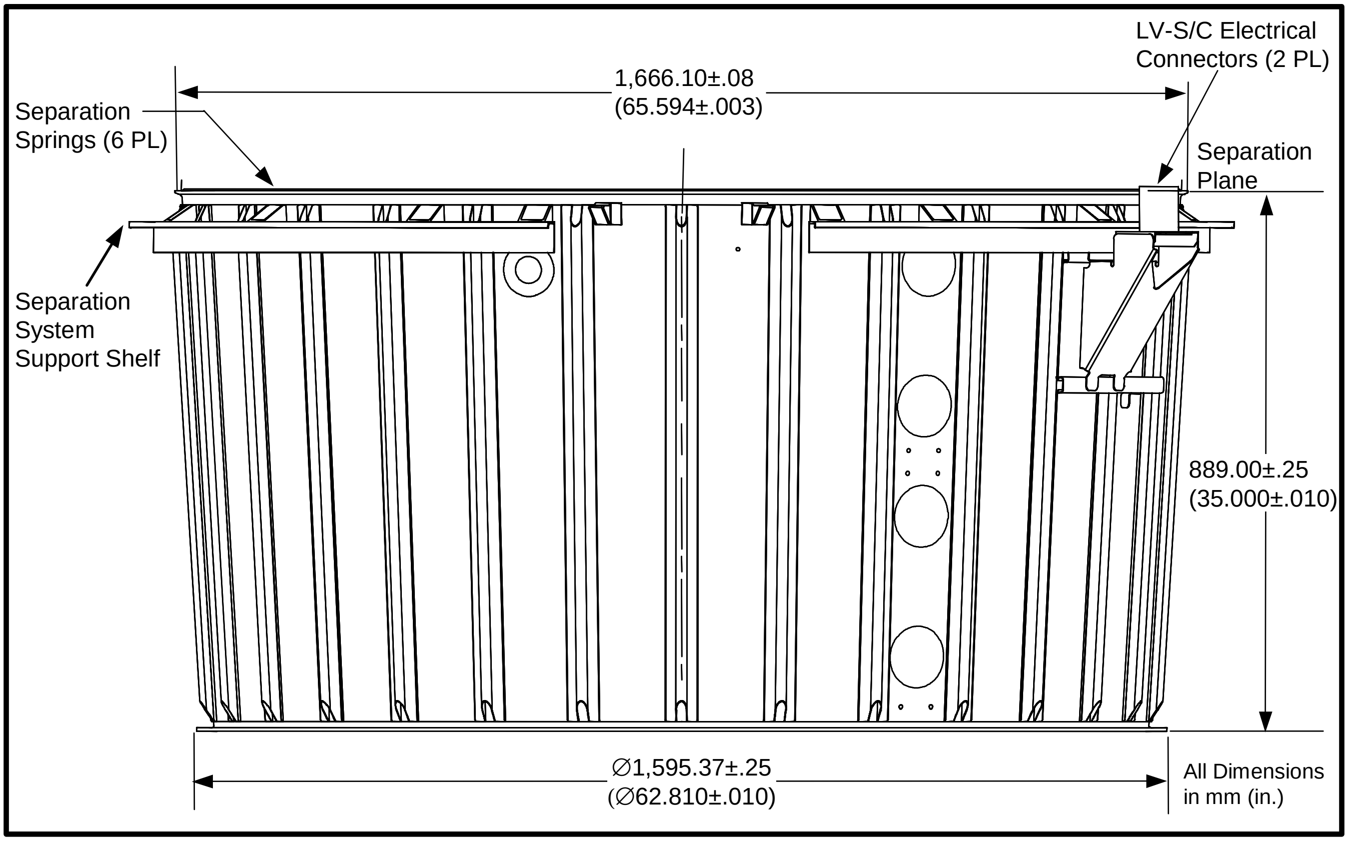


Figure E.5-1 Atlas Type D Payload Adapter

Payload Separation System—The Atlas Type D payload adapter uses a launch vehicle-provided Marmon-type clampband payload separation system. This separation system (Fig. E.5-2) consists of a clampband set, release mechanism, and separation springs. The clampband set consists of a clampband for holding the spacecraft and adapter rings together plus devices to extract, catch, and retain the clampband on the adapter structure after separation. The clampband includes aluminum clamp segments that hold the payload adapter and spacecraft rings together and a two-piece stainless-steel retaining band that holds the clamp segments in place. The clamp segments also hold shear pins at 12 locations that fit into shear slots on the spacecraft aft ring and payload adapter forward ring to increase the structural capability of this system. The shear slot pattern on the spacecraft and payload adapter rings for this interface is controlled by matched tooling. The ends of the retaining band are held together by tension bolts. For separation, a pyrotechnically activated bolt-cutter severs the tension bolts allowing the end of the clamp segments to move apart and release the payload adapter and spacecraft mating rings.

Separation spring assemblies provide the necessary separation energy after the clampband is released and are mounted to the payload adapter forward ring and bear on the spacecraft aft ring. Positive spacecraft separation is detected through continuity loops installed in the spacecraft electrical connector and wired to the upper-stage instrumentation for monitoring and telemetry verification.

Payload Adapter Structural Capabilities—Allowable spacecraft weights and longitudinal centers of gravity for the Type D payload adapter/separation systems are shown in Figure E.5-3. These spacecraft mass and cg capabilities were determined using generic spacecraft interface ring geometry as shown in Figure E.5-4, and quasi-static load factors shown in Section 3.2.1. Actual spacecraft design allowables may vary depending on interface ring stiffness and results of spacecraft mission-specific coupled loads analyses. Coordination with the Atlas program is required to define appropriate structural capabilities for spacecraft designs that exceed these generic allowables.

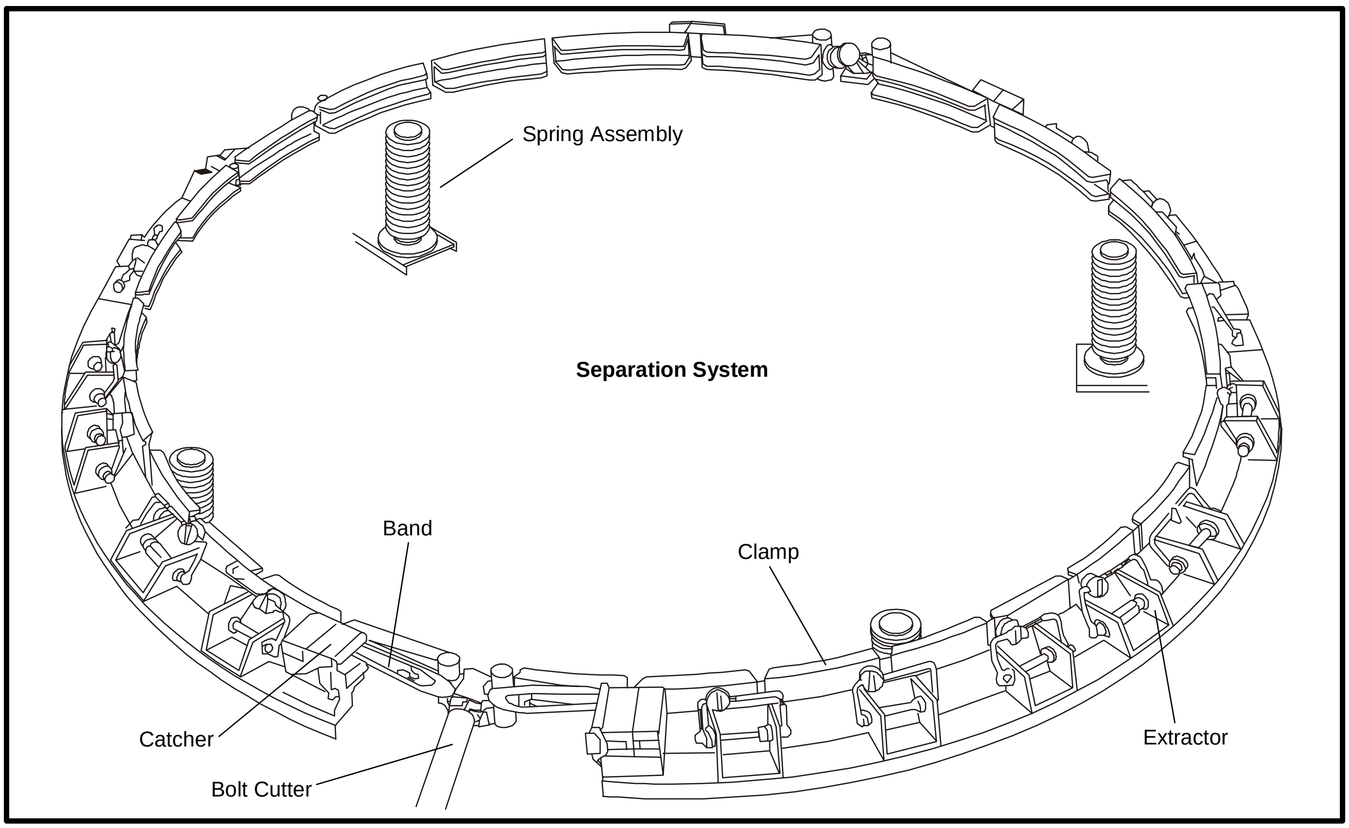


Figure E.5-2 PSS66 Payload Separation System

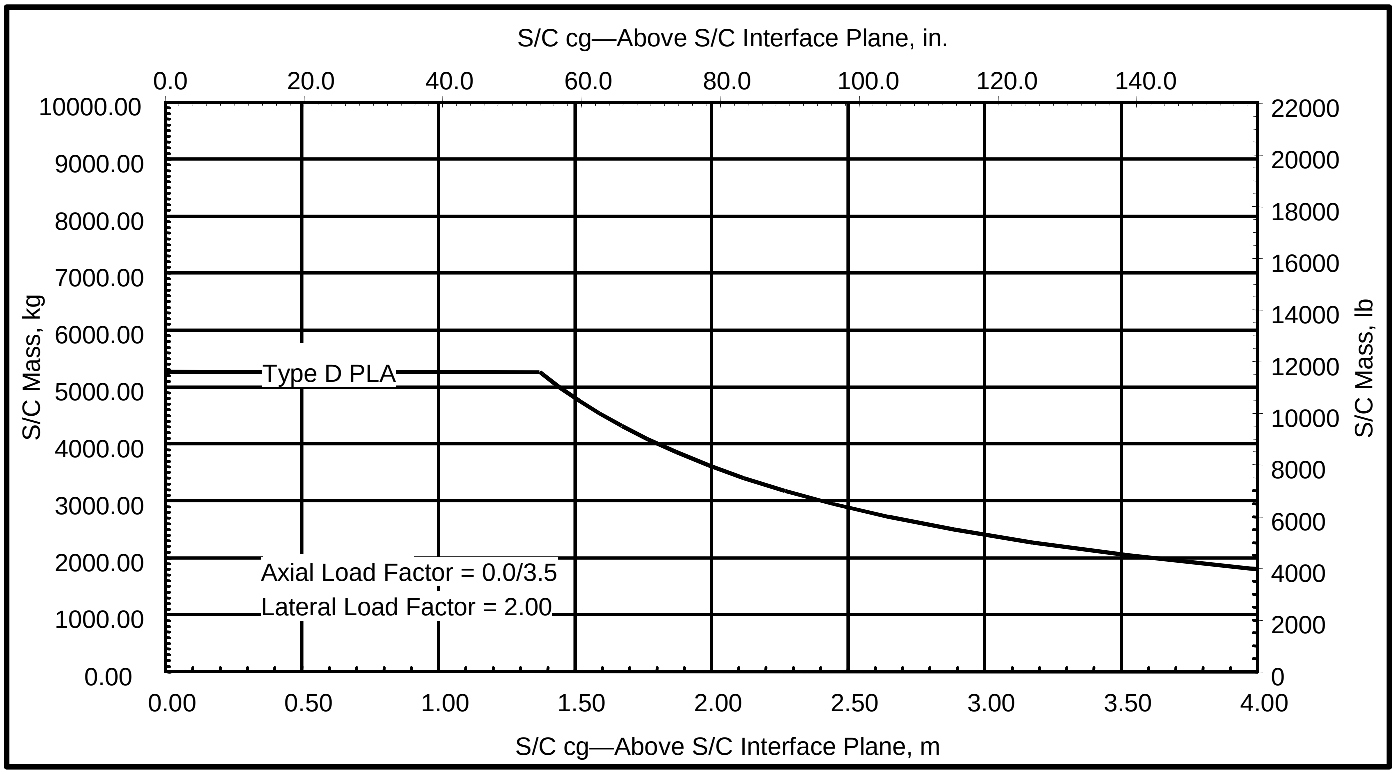


Figure E.5-3 Atlas Type D Payload Adapter Structural Capability

Payload Adapter Interfaces—The primary structural interface between the launch vehicle and spacecraft occurs at the payload adapter forward ring. This ring interfaces with the spacecraft aft ring and a payload separation system holds the two rings together for the structural joint and provides the release mechanism for spacecraft separation. Electrical bonding is provided across all interface planes associated with these components. The payload adapter also provides mounting provisions for separation springs and supports interfacing components for electrical connectors between the launch vehicle and spacecraft. Interface requirements for these components are shown in Figures E.5-4 and Figure E.5-5. Additional mission-specific provisions, including spacecraft purge provisions, spacecraft range safety destruct units, and mission satisfaction kit instrumentation, may be added as necessary.

Static Payload Envelope—The usable volume for the spacecraft relative to the payload adapter is defined by the static payload envelope. This envelope represents the maximum allowable spacecraft static dimensions (including manufacturing tolerances) relative to the spacecraft/payload adapter interface. This envelope is designed to allow access to the mating components and payload separation system for integration and installation operations, motion of the payload separation system during its operation, and movement of the spacecraft and launch vehicle after separation of the spacecraft and payload. Clearance layouts and separation analyses are performed for each spacecraft configuration, and if necessary, critical clearance locations are measured during spacecraft-to-payload-adapter mate operations to ensure positive clearance during flight and separation. Detailed views of the static payload envelope for the Type D payload adapter are shown in Figure E.5-6.

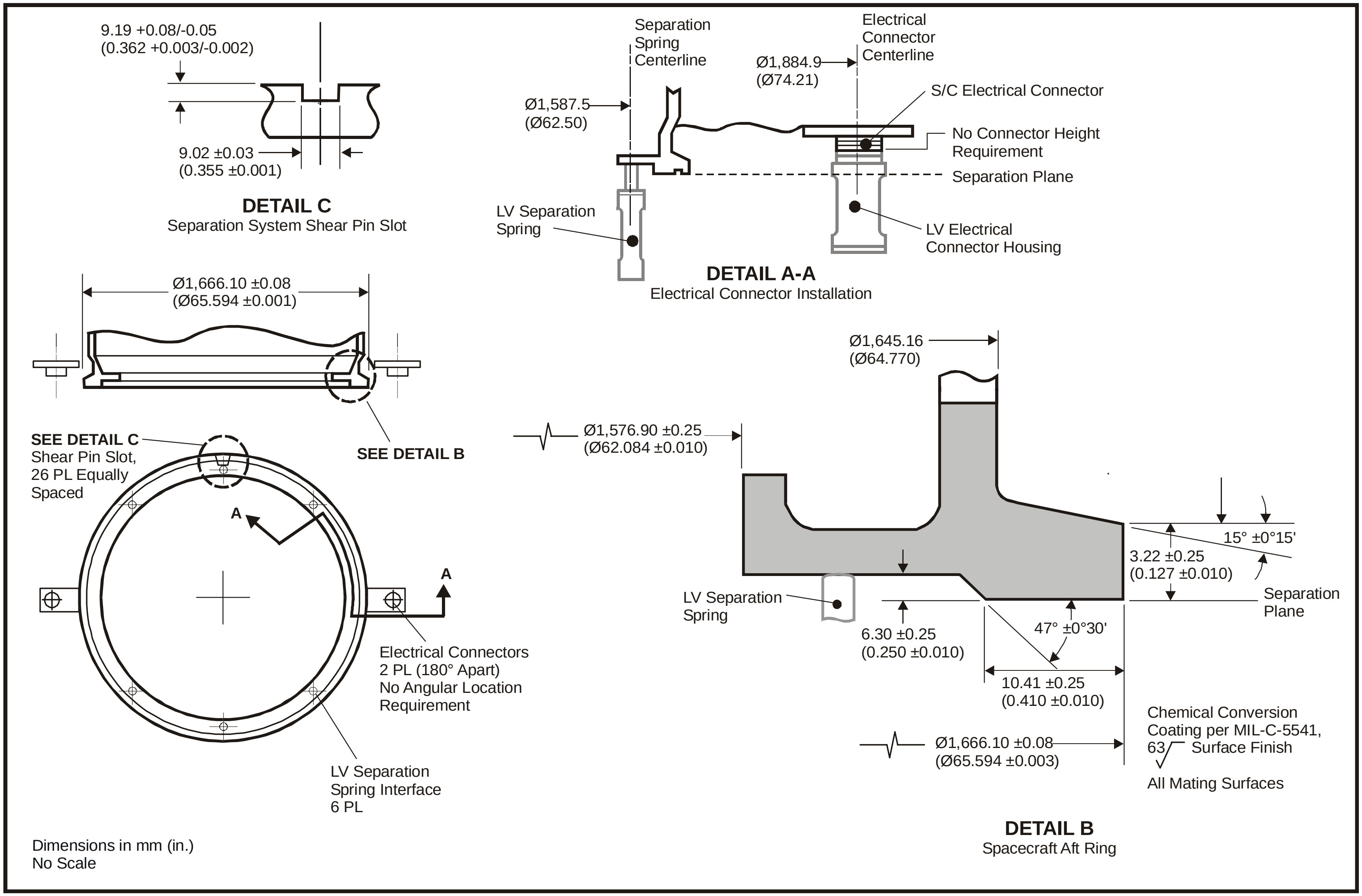


Figure E.5-4 Spacecraft Interface Requirements

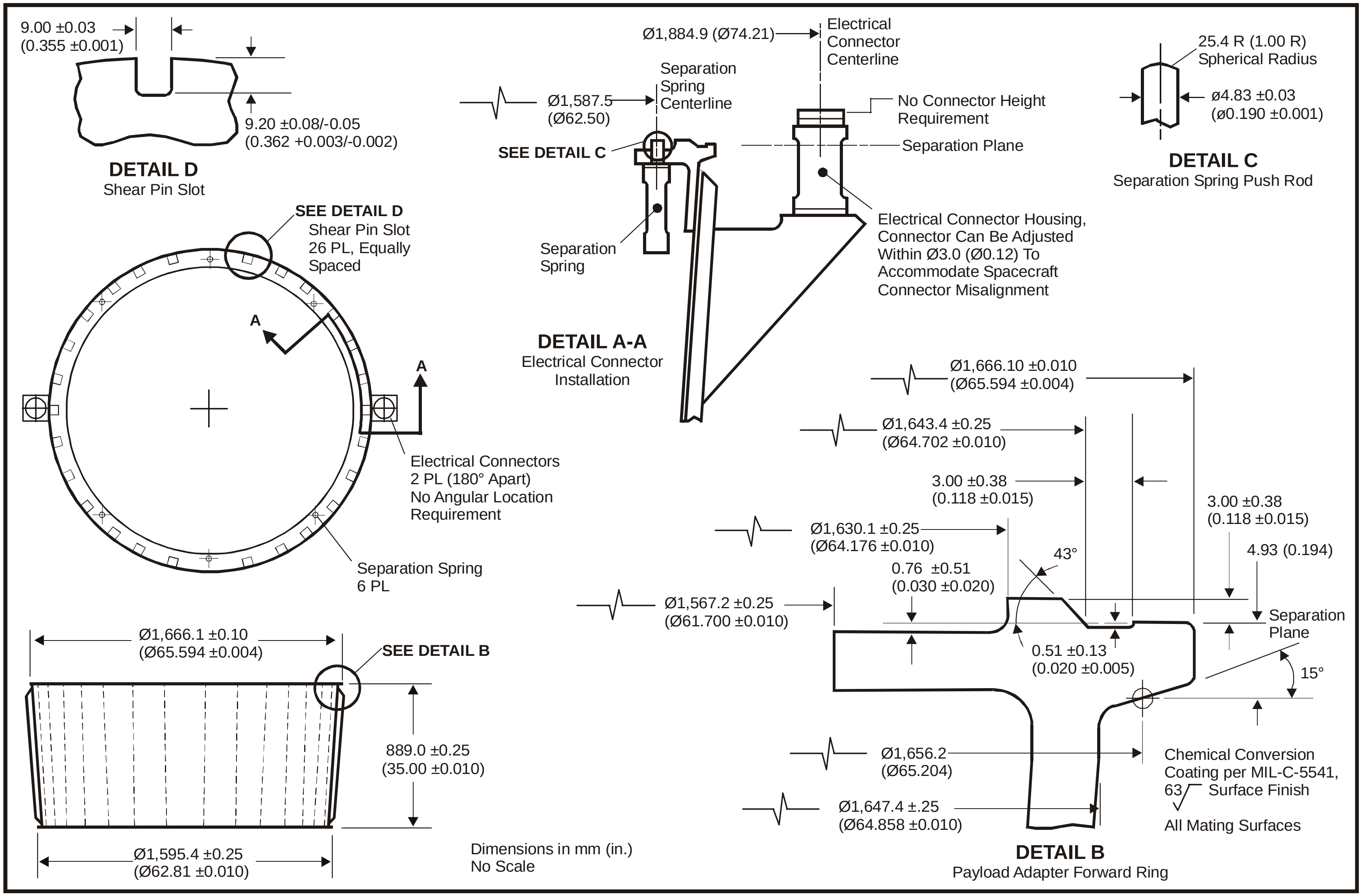


Figure E.5-5 Atlas Type D Payload Adapter Interface Requirements

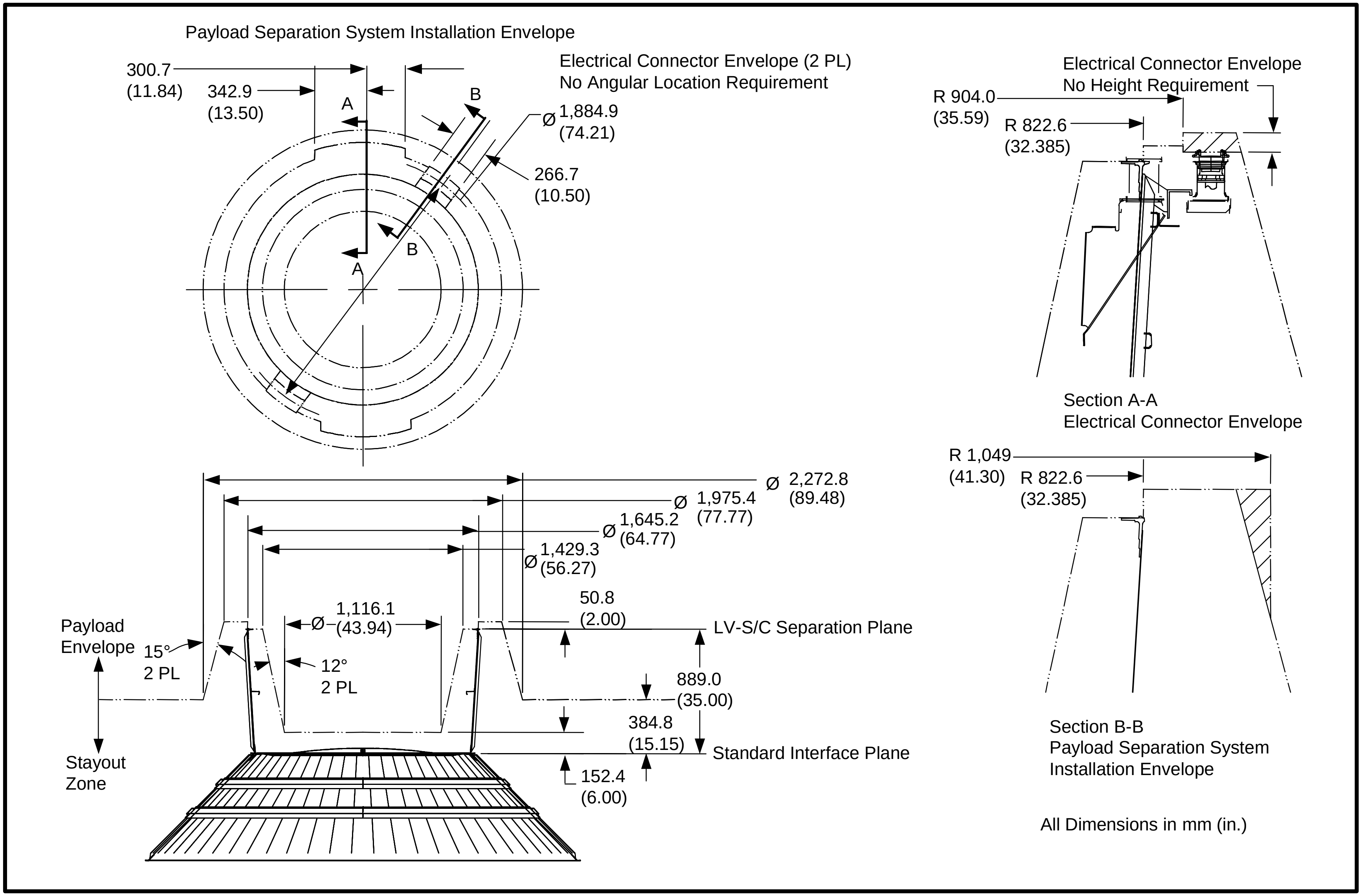


Figure E.5-6 Atlas Type D Payload Adapter Static Payload Envelope

E.6 ATLAS TYPE E PAYLOAD ADAPTER

The Atlas Type E adapter is designed for spacecraft that mate to the launch vehicle using six-discrete hard points on an interface diameter of 1,956 mm (77 in.). Major characteristics of this payload adapter are summarized in Table E.6-1. This adapter is an aluminum skin, stringer, and frame construction with machined forward and aft rings that mate to the spacecraft and launch vehicle forward adapter (Fig. E.6-1). The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 121 holes that match up with Atlas standard interface plane requirements. The nominal height of the Type E payload adapter is 422.9 mm (16.65 in.). The Type E payload adapter supports all hardware that directly interfaces with spacecraft including the payload separation system, electrical connectors, and mission-specific options and includes all provisions for mating to the launch vehicle ground support equipment, including the torus arms and isolation diaphragm, used during ground processing operations.

Table E.6-1 Atlas Type E Payload Adapter Characteristics

Atlas Type E Payload Adapter		
Construction	Aluminum Skin/Stringer/Frame Construction	
Mass Properties	98 kg	215 lb
Payload Capability	Figure E.6-3	
	6,800 kg at 3.05 m	15,000 lb at 120 in.
P/L Separation Sys	Separation Nut	
Max Shock Levels	Section 3.2.4	
Separation Nut Preload	92 kN	20,700 lb
Separation Springs		
Number	6	
Force per Spring—Max	1 kN	225 lb

Payload Separation System—The Atlas Type E payload adapter uses a launch vehicle-provided six-separation nut set that attaches the spacecraft to the forward ring of the payload adapter, and a separation spring set that provides the necessary separation energy after the separation nut is actuated. The separation nut set consists of the separation stud, separation nut, and stud catcher (Fig. E.6-2). The separation stud forms the mechanical connection between the spacecraft and payload adapter. The separation nut holds the end of the stud on the payload adapter side and contains a pyrotechnically activated mechanism to release the stud for separation. The stud catcher captures the stud on the spacecraft structure after separation. For spacecraft requiring a lower shock environment, the option exists of using a fast-acting shockless separation nut (FASSN) with the Type E adapter. The FASSN system is designed to rapidly separate a bolted joint while producing minimal shock. Separation springs are mounted on the aft side of the payload adapter forward ring, and the pushrod of each spring acts through a hole in the forward ring to push against the payload aft structure. Positive spacecraft separation is detected through continuity loops installed in the spacecraft interface connector and wired to the upper-stage instrumentation for monitoring and telemetry verification. Coordinated tooling between the spacecraft and payload adapter is required for this system.

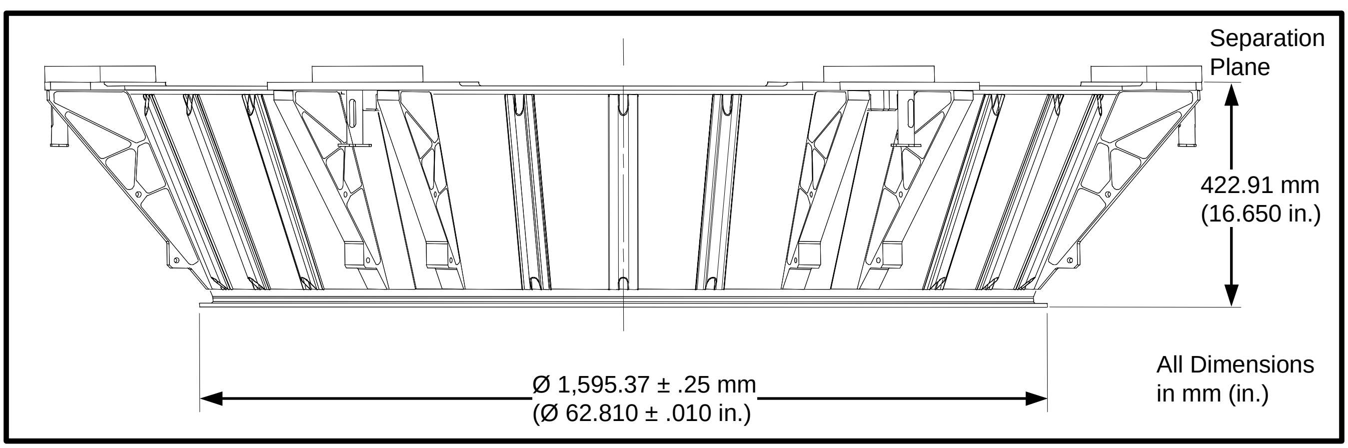


Figure E.6-1 Atlas Type E Payload Adapter

Payload Adapter Structural Capabilities—Allowable spacecraft weights and longitudinal centers of gravity for Type E payload adapter/separation systems are shown in Figure E.6-3. These spacecraft mass and cg capabilities were determined using quasi-static load factors shown in Section 3.2.1. Actual spacecraft design allowables may vary depending on interface component stiffness and results of spacecraft mission-specific coupled loads analyses. Coordination with the Atlas program is required to define appropriate structural capabilities for spacecraft designs that exceed these generic allowables.

Payload Adapter Interfaces—The primary structural interface between the launch vehicle and spacecraft occurs at the bearing surfaces around the six separation nuts. These surfaces interface with the equivalent surfaces on the spacecraft aft end. Electrical bonding is provided across all interface planes associated with these components. Interface requirements for these components are shown in Figure E.6-4. The payload adapter also provides mounting provisions for separation springs and supports interfacing components for electrical connectors between the launch vehicle and spacecraft. Additional mission-specific provisions, including spacecraft purge provisions, spacecraft range safety destruct units, and mission satisfaction kit instrumentation, may be added as necessary.

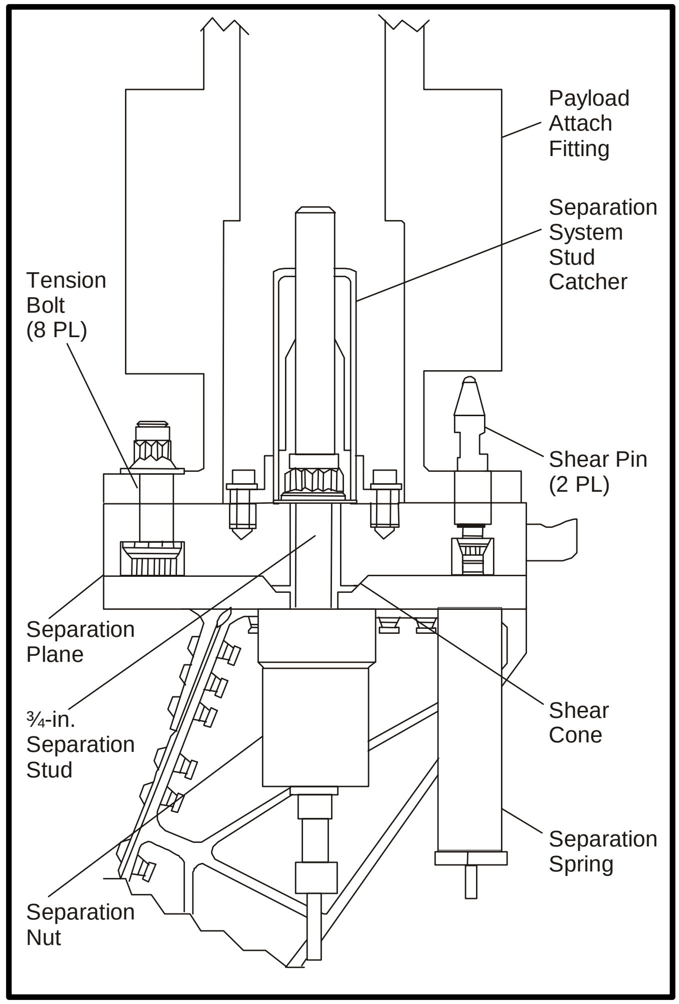


Figure E.6-2 PSS77 Payload Separation Nut Set

Static Payload Envelope—The usable volume for the spacecraft relative to the payload adapter is defined by the static payload envelope. This envelope represents the maximum allowable spacecraft static dimensions (including manufacturing tolerances) relative to the spacecraft/payload adapter interface. This envelope is designed to allow access to the mating components and payload separation system for integration and installation operations, motion of the payload separation system during its operation, and movement of the spacecraft and launch vehicle after separation of the spacecraft and payload. Clearance layouts and separation analyses are performed for each spacecraft configuration, and if necessary, critical clearance locations are measured during spacecraft-to-payload-adapter mate operations to ensure positive clearance during flight and separation. Detailed views of the static payload envelope for the Type E payload adapter are shown in Figure E.6-5.

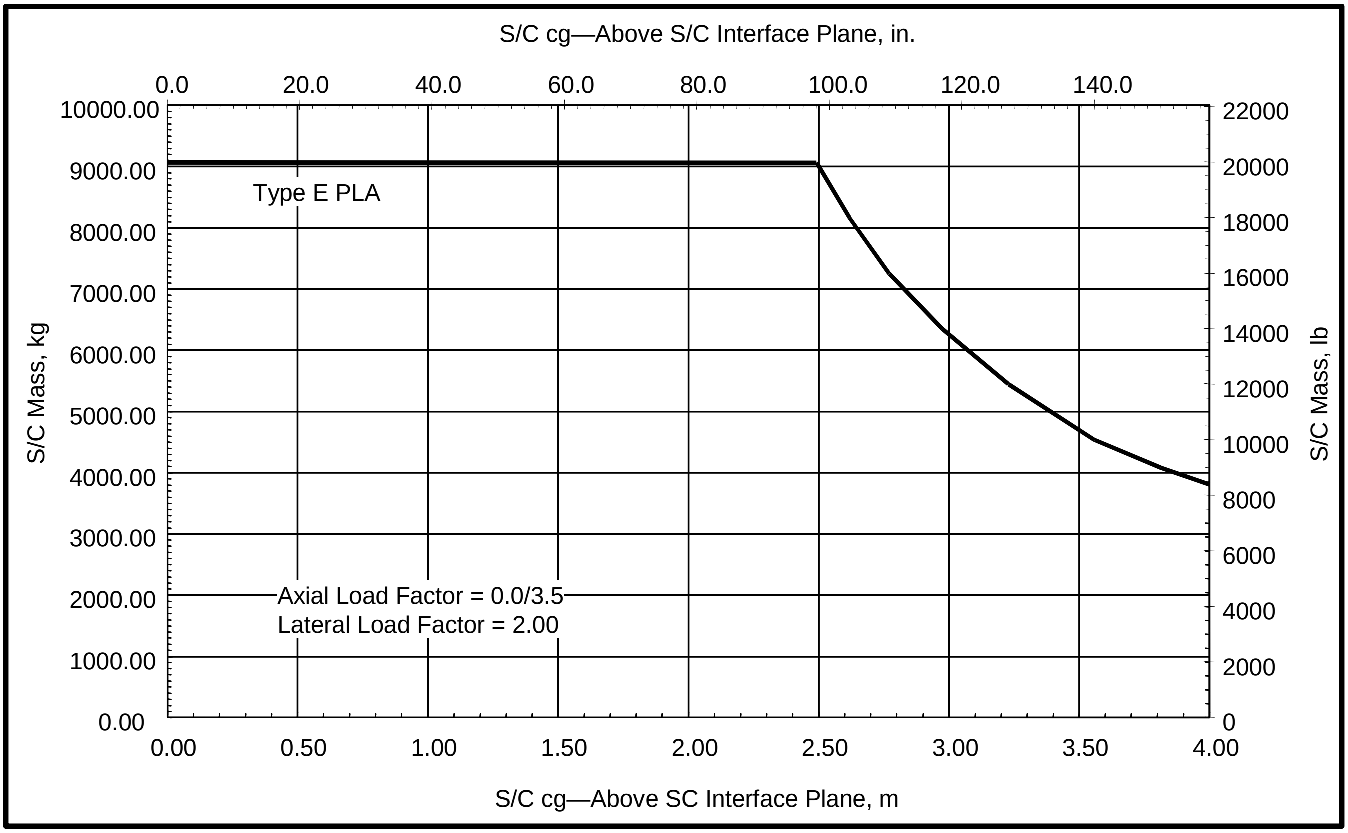


Figure E.6-3 Atlas Type E Payload Adapter Structural Capability

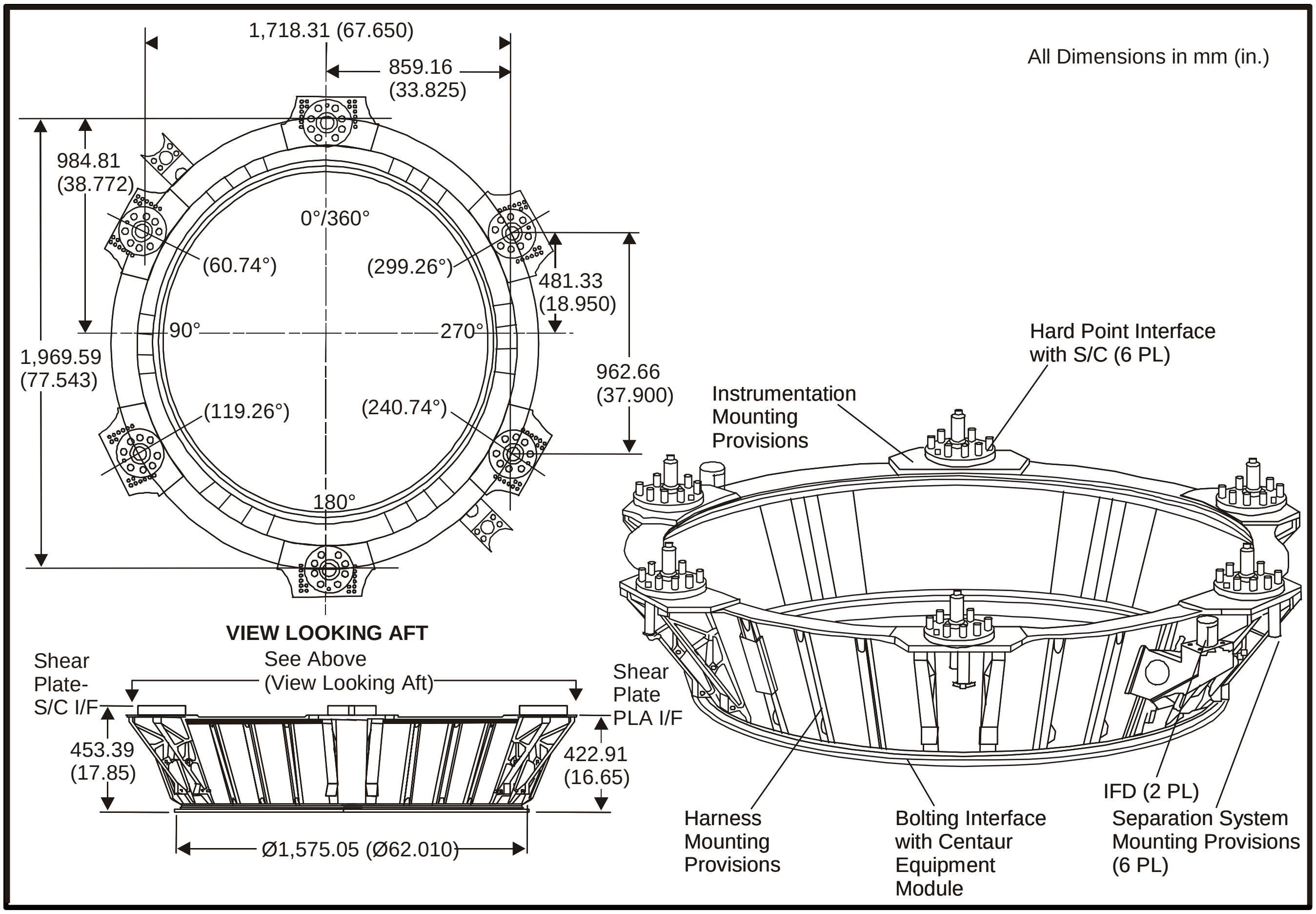


Figure E.6-4 Atlas Type E Payload Adapter Interface Requirements

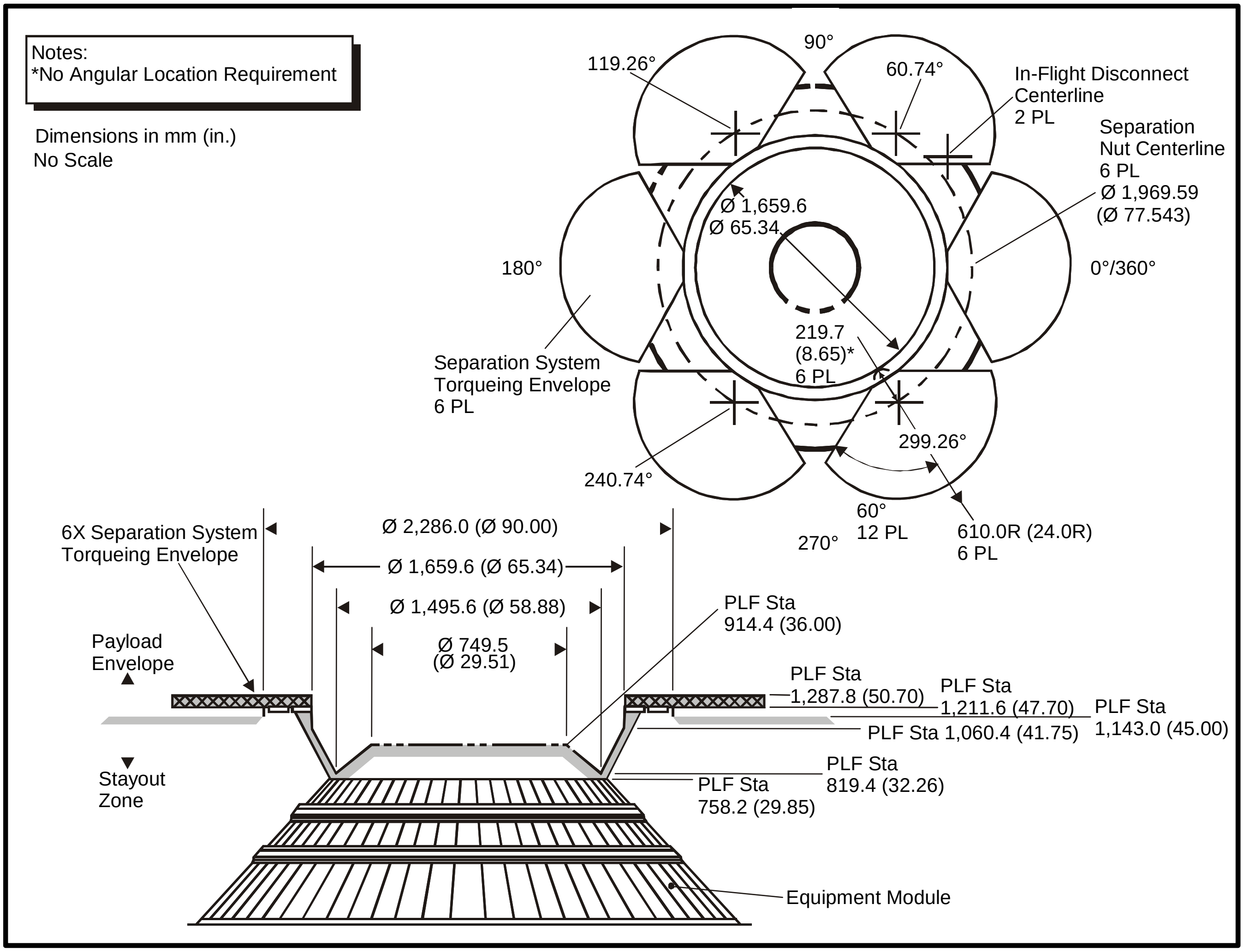


Figure E.6-5 Atlas Type E Payload Adapter Static Payload Envelope

E.7 ATLAS TYPE B1194VS PAYLOAD ADAPTER

The Atlas Type B1194VS payload adapter is designed to support spacecraft with an aft ring diameter of 1,194 mm (47 in.). Major characteristics of this payload adapter are summarized in Table E.7-1. This payload adapter consists of two major sections: the payload separation ring and the launch vehicle adapter (Fig. E.7-1). The payload separation ring is a machined aluminum component in the form of a 254-mm (10-in.) high truncated cone. The forward ring has an outer diameter of 1,215 mm (47.835 in.) and forms the spacecraft separation plane. The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 120 evenly spaced holes that allow it to be joined to the launch vehicle adapter. This symmetrical hole pattern allows the payload separation ring and attached spacecraft to be rotated relative to the launch vehicle in 3° increments to meet mission-specific requirements. The payload separation ring supports all hardware that directly interfaces with spacecraft, including the payload separation system, electrical connectors, and mission-specific options.

Table E.7-1 Atlas Type B1194 VS Payload Adapter Characteristics

Atlas Type B1194VS Payload Adapter		
Construction	Two-Piece, Integrally Machined Aluminum Construction	
Mass Properties	91 kg	200 lb
Payload Capability	Figure E.7-3	
	8,000 kg at 2.5 m	17,640 lb at 98 in.
P/L Separation Sys	PSS47VS	
Max Shock Levels	Available on Request	
Clampband Preload—Installation	66.0 ± 0.5 kN	14,840 ± 22 lb
Clampband Preload—Flight	60.0 ± 0.5 kN	13,490 ± 122 lb
Separation Springs		
Number	4–8	
Force per Spring—Max	1 kN	225 lb

The launch vehicle adapter is a machined aluminum component in the form of an integrally stiffened cylinder. The forward ring has an outer diameter of 1,595 mm (62.81 in.) and contains 120 holes spaced evenly every 3° that allow it to be joined to the payload separation ring. The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 121 holes that match up with Atlas standard interface plane requirements. The nominal height of the launch vehicle adapter is 558.8 mm (22.00 in.), but this height may be varied from 330.2 mm (13.00 in.) to 736.6 mm (29.00 in.) to meet mission-specific requirements. The launch vehicle adapter includes all provisions for mating to the launch vehicle ground

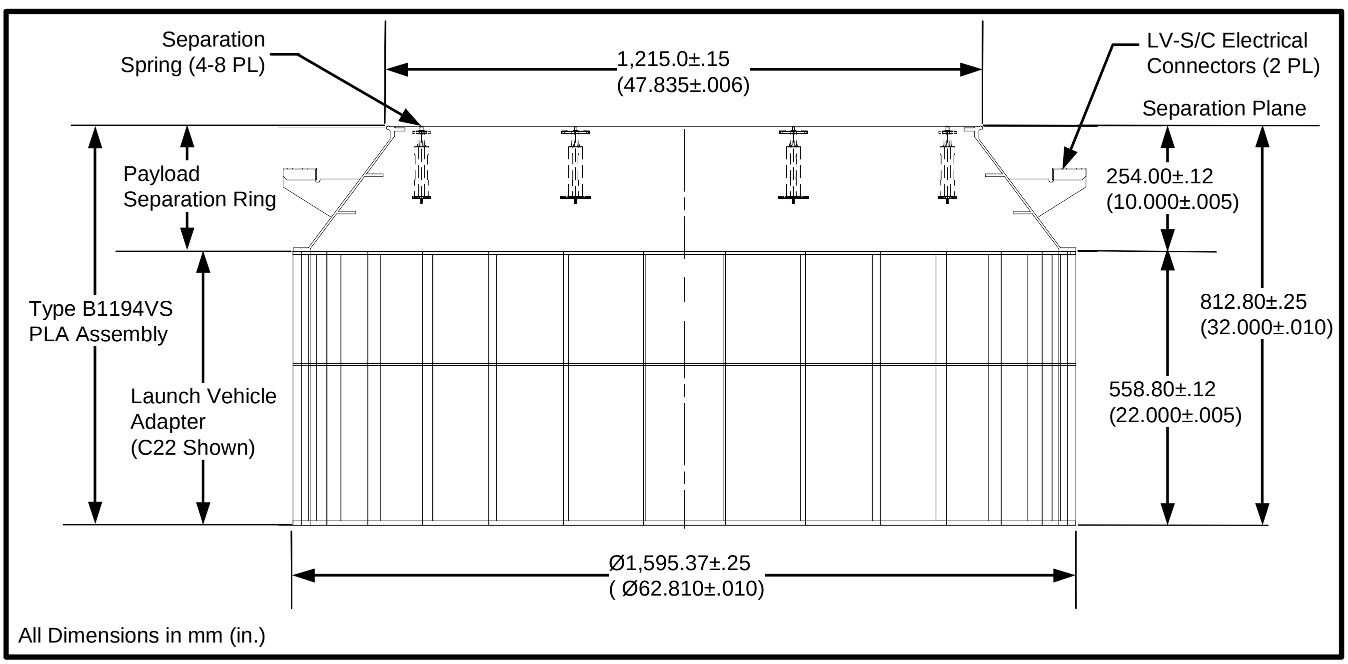


Figure E.7-1 Atlas Type B1194VS Payload Adapter

support equipment, including the torus arms and isolation diaphragm, used during ground processing operations.

Payload Separation System—The Atlas Type B1194VS payload adapter uses a launch vehicle-provided Marmon-type clampband payload separation system. This separation system (Fig. E.7-2) consists of a clampband set, release mechanism, and separation springs. The clampband set consists of a clampband for holding the spacecraft and adapter rings together plus devices to catch and retain the clampband on the adapter structure after separation. The clampband includes aluminum clamp segments that hold the payload adapter and spacecraft rings together and a single-piece aluminum retaining band that holds the clamp segments in place.

The ends of the retaining band are held together by the low-shock clampband opening device (CBOD). The CBOD includes release bolts that engage the ends of the clampband. These release bolts are threaded into a flywheel mechanism. During installation and flight, the flywheel is restrained against rotation by a restraining pin. For separation, a pyrotechnically activated pin-puller retracts this pin from the flywheel, allowing it to rotate and eject the release bolts. This system significantly reduces shock compared to a conventional bolt-cutter system and is resetable allowing actual flight hardware to be tested during component acceptance testing.

Separation spring assemblies provide the necessary separation energy after the clampband is released and are mounted to the payload adapter forward ring and bear on the spacecraft aft ring. Positive spacecraft separation is detected through continuity loops installed in the spacecraft electrical connector and wired to the upper-stage instrumentation for monitoring and telemetry verification.

Payload Adapter Structural Capabilities—Allowable spacecraft weights and longitudinal centers of gravity for the Type B1194VS payload adapter/separation systems are shown in Figure E.7-3. These spacecraft mass and center of gravity capabilities were determined using generic spacecraft interface ring geometry as shown in Figure E.7-4, and quasi-static load factors shown in Section 3.2.1. Actual spacecraft design allowables may vary depending on interface ring stiffness and results of spacecraft mission-specific coupled loads analyses. Coordination with the Atlas program is required to define appropriate structural capabilities for spacecraft designs that exceed these generic allowables.

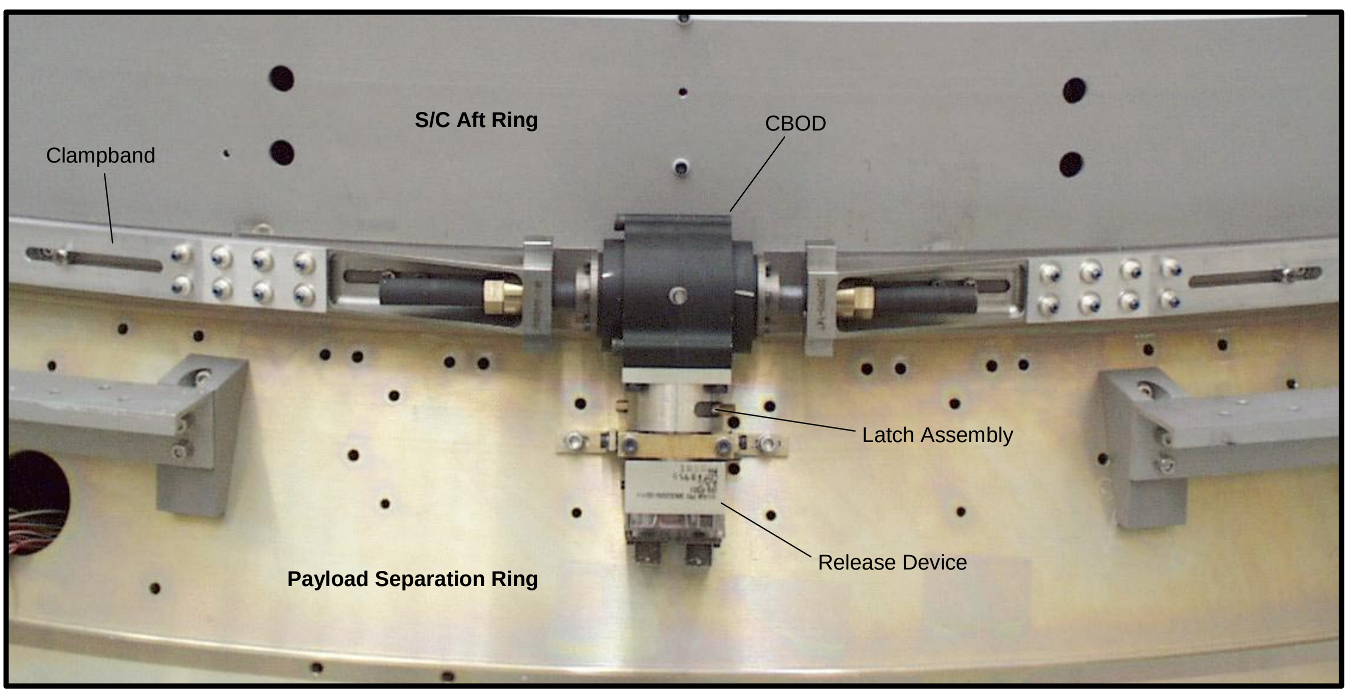


Figure E.7-2 PSS47VS Payload Separation System

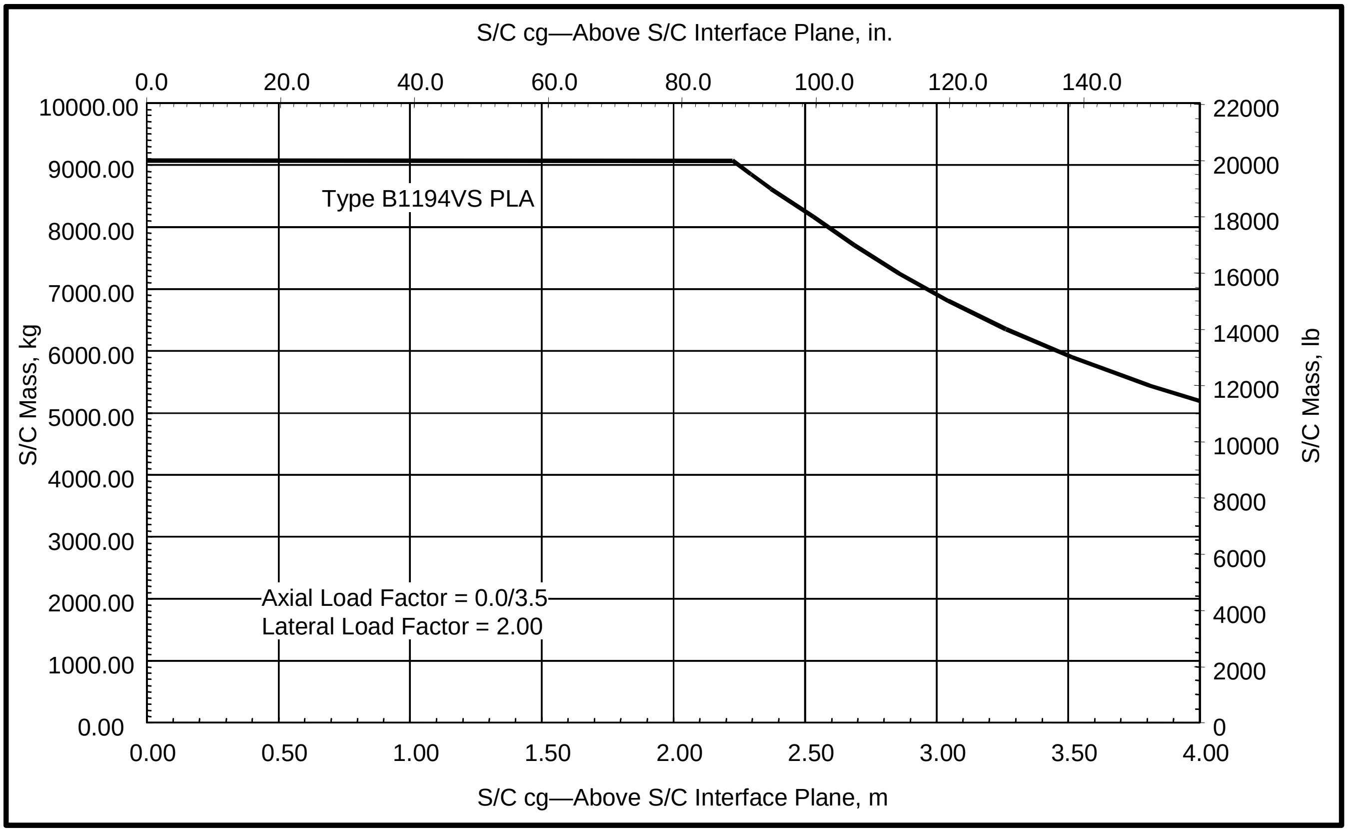


Figure E.7-3 Atlas Type B1194VS Payload Adapter Structural Capability

Payload Adapter Interfaces—The primary structural interface between the launch vehicle and spacecraft occurs at the payload adapter forward ring. This ring interfaces with the spacecraft aft ring and a payload separation system holds the two rings together for the structural joint and provides the release mechanism for spacecraft separation. Electrical bonding is provided across all interface planes associated with these components. The payload adapter also provides mounting provisions for separation springs and supports interfacing components for electrical connectors between the launch vehicle and spacecraft. Interface requirements for these components are shown in Figures E.7-4 and E.7-5. Additional mission-specific provisions, including spacecraft purge provisions, spacecraft range safety destruct units, and mission satisfaction kit instrumentation, may be added as necessary.

Static Payload Envelope—The usable volume for the spacecraft relative to the payload adapter is defined by the static payload envelope. This envelope represents the maximum allowable spacecraft static dimensions (including manufacturing tolerances) relative to the spacecraft/payload adapter interface. This envelope is designed to allow access to the mating components and payload separation system for integration and installation operations, motion of the payload separation system during its operation, and movement of the spacecraft and launch vehicle after separation of the spacecraft and payload. Clearance layouts and separation analyses are performed for each spacecraft configuration, and if necessary, critical clearance locations are measured during spacecraft-to-payload-adapter mate operations to ensure positive clearance during flight and separation. Detailed views of the static payload envelope for the B1194VS payload adapter are shown in Figure E.7-6.

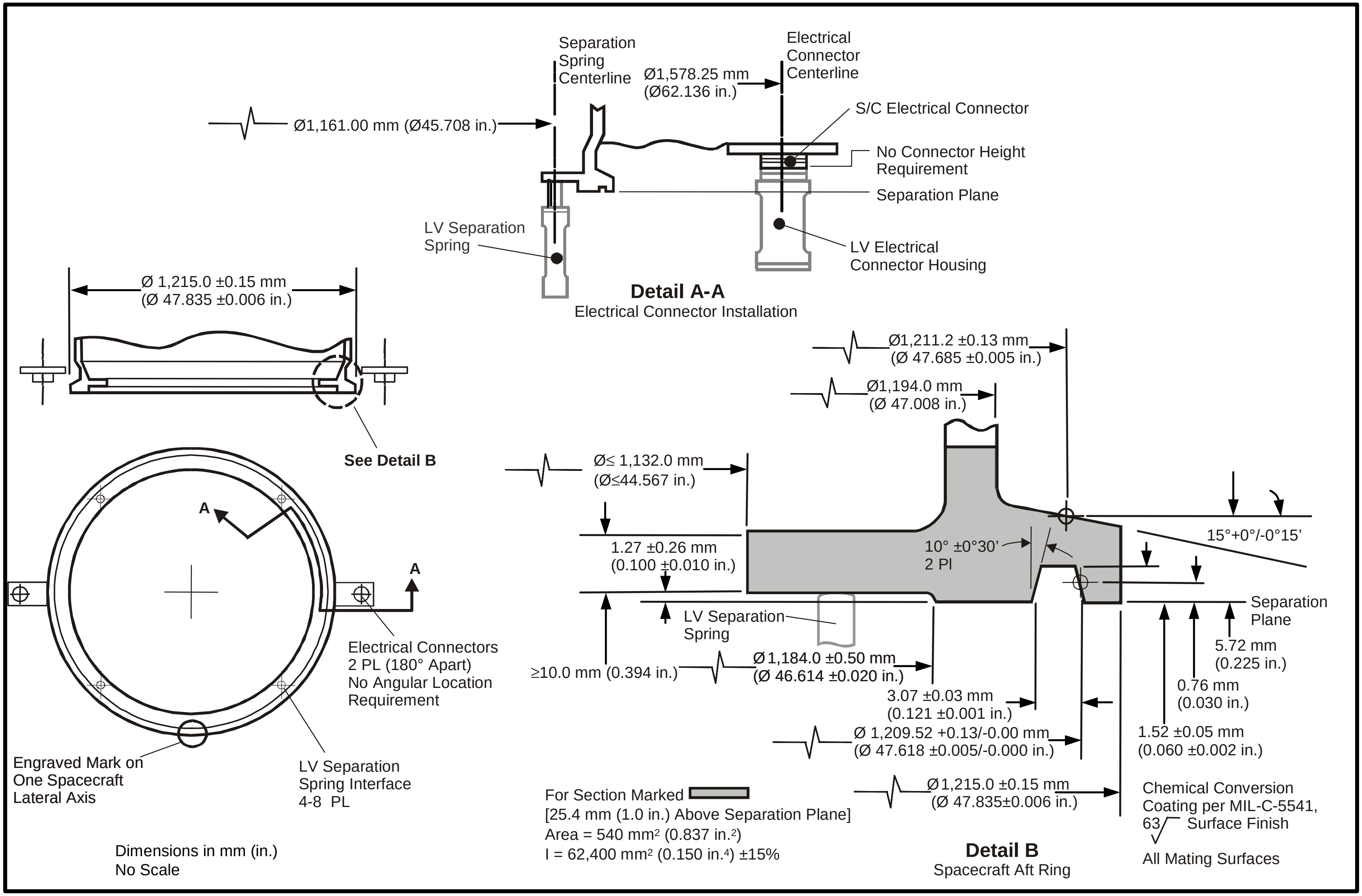


Figure E.7-4 Spacecraft Interface Requirements

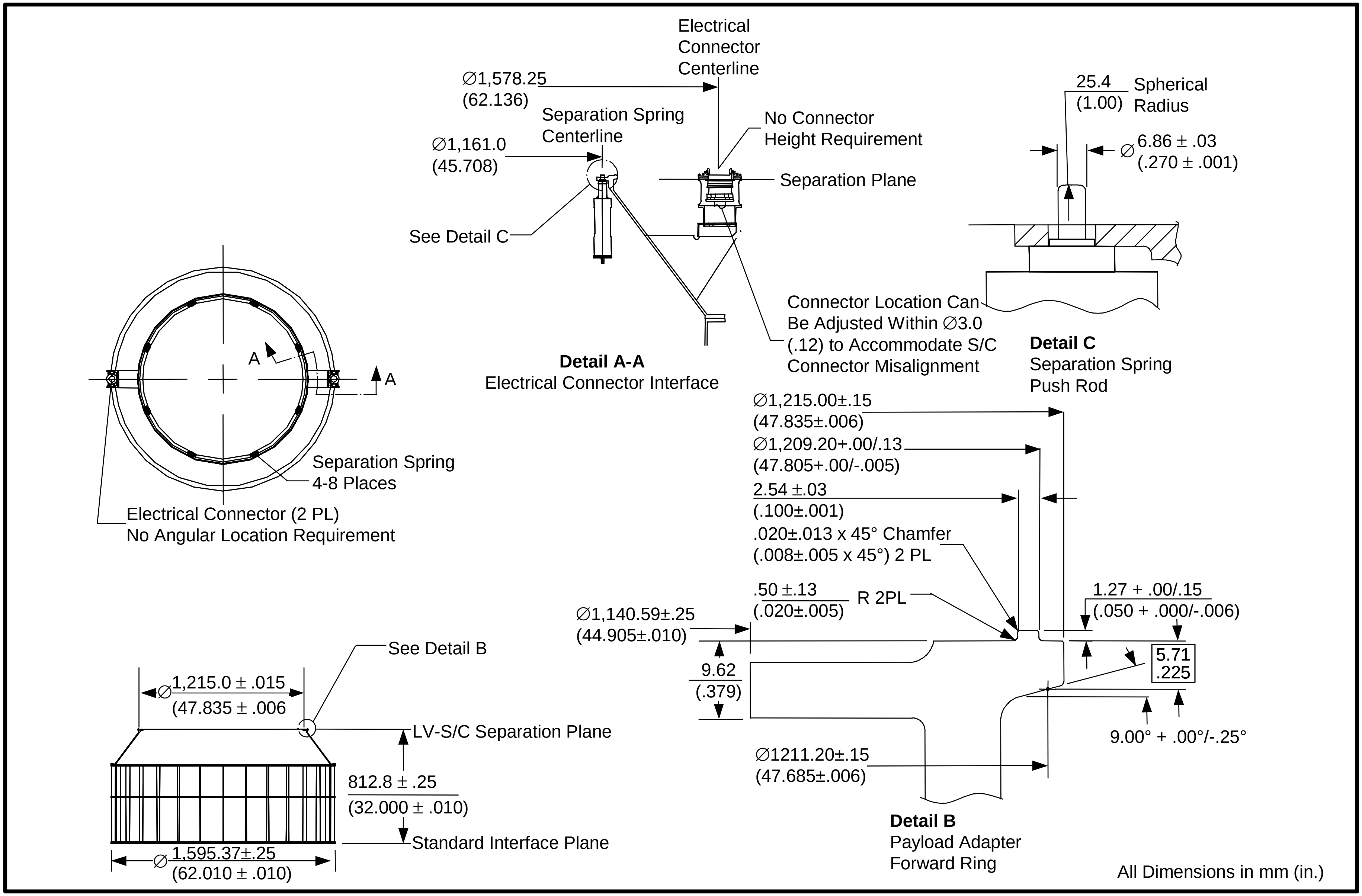


Figure E.7-5 Atlas Type B1194VS Payload Adapter Interface Requirements

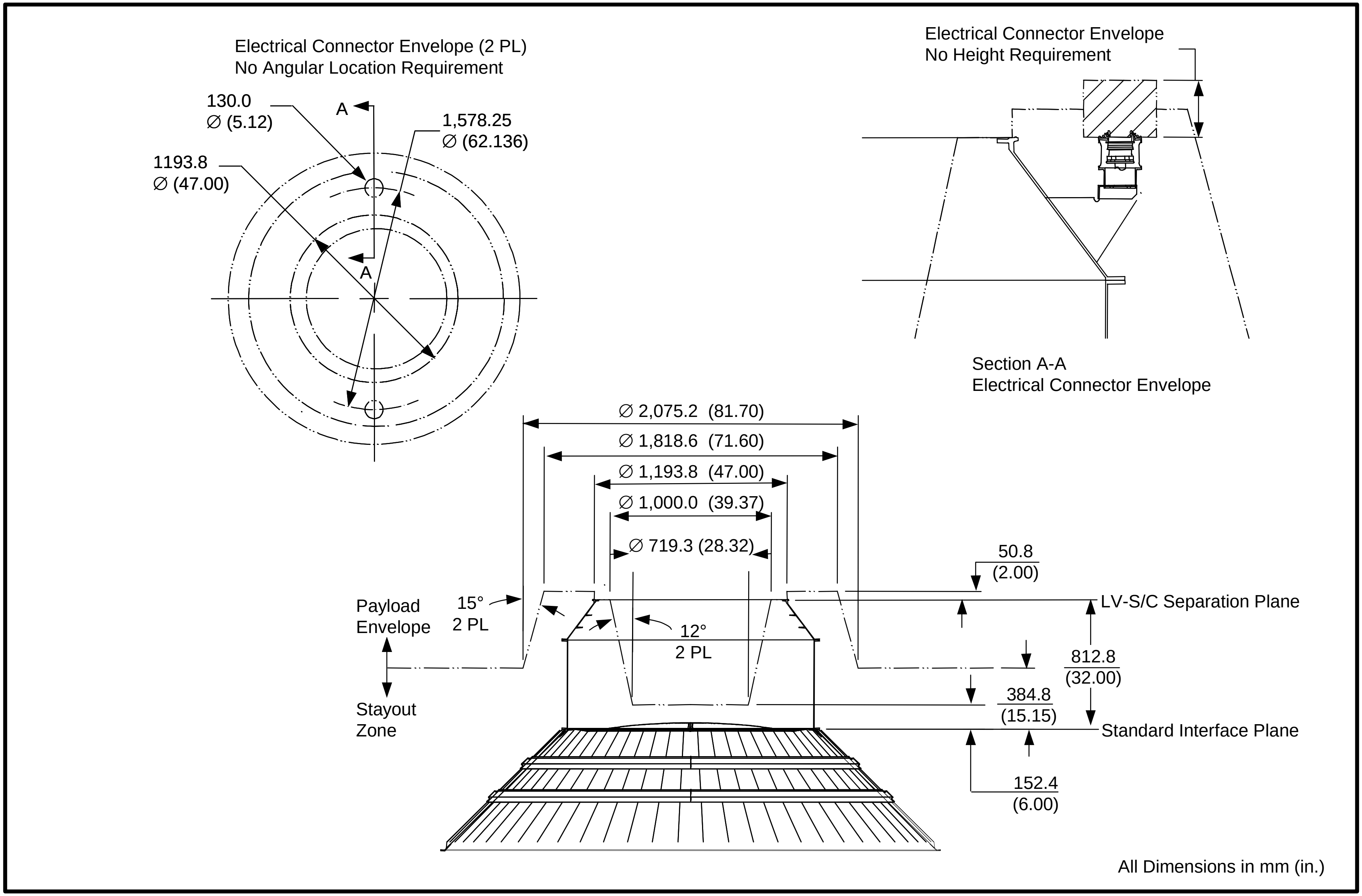


Figure E.7-6 Atlas Type B1194VS Payload Adapter Static Payload Envelope

E.8 ATLAS TYPE D1666VS PAYLOAD ADAPTER

The Atlas Type D1666VS payload adapter is designed to support spacecraft with an aft ring diameter of 1,666 mm (66 in.). Major characteristics of this payload adapter are summarized in Table E.8-1. This payload adapter consists of two major sections: the payload separation ring and the launch vehicle adapter (Fig. E.8-1). The payload separation ring is a machined aluminum component in the form of a 330.2-mm (13-in.) high truncated cone. The forward ring has an outer diameter of 1,666.1 mm (65.594 in.) and forms the spacecraft separation plane. The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 120 evenly spaced holes that allow it to be joined to the launch vehicle adapter. This symmetrical hole pattern allows the payload separation ring and attached spacecraft to be rotated relative to the launch vehicle in 3° increments to meet mission-specific requirements. The payload separation ring supports all hardware that directly interfaces with spacecraft, including the payload separation system, electrical connectors, and mission-specific options.

Table E.8-1 Atlas Type D1666VS Payload Adapter Characteristics

Atlas Type D1666VS Payload Adapter		
Construction	Two-Piece, Integrally Machined Aluminum	
Mass Properties	91 kg	200 lb
Payload Capability	Figure E.8-3	
	8,000 kg at 2.5 m	17,640 lb at 98 in.
P/L Separation Sys	PSS66VS	
Max Shock Levels	Available on Request	
Clampband Preload—Installation	43.8 ± 0.1kN	9,770 ± 22 lb
Clampband Preload—Flight	40.0 ± 0.5 kN	8,990 ± 112 lb
Separation Springs		
Number	4–8	
Force per Spring—Max	1 kN	225 lb

The launch vehicle adapter is a machined aluminum component in the form of an integrally stiffened cylinder. The forward ring has an outer diameter of 1,595 mm (62.81 in.) and contains 120 holes spaced evenly every 3° that allow it to be joined to the payload separation ring. The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 121 holes that match up with Atlas standard interface plane requirements. The nominal height of the launch vehicle adapter is 558.8 mm (22.00 in.), but this height may be varied from 330.2 mm (13.00 in.) to 736.6 mm (29.00 in.) to meet mission-specific requirements. The launch vehicle adapter includes all provisions for mating to the launch vehicle ground

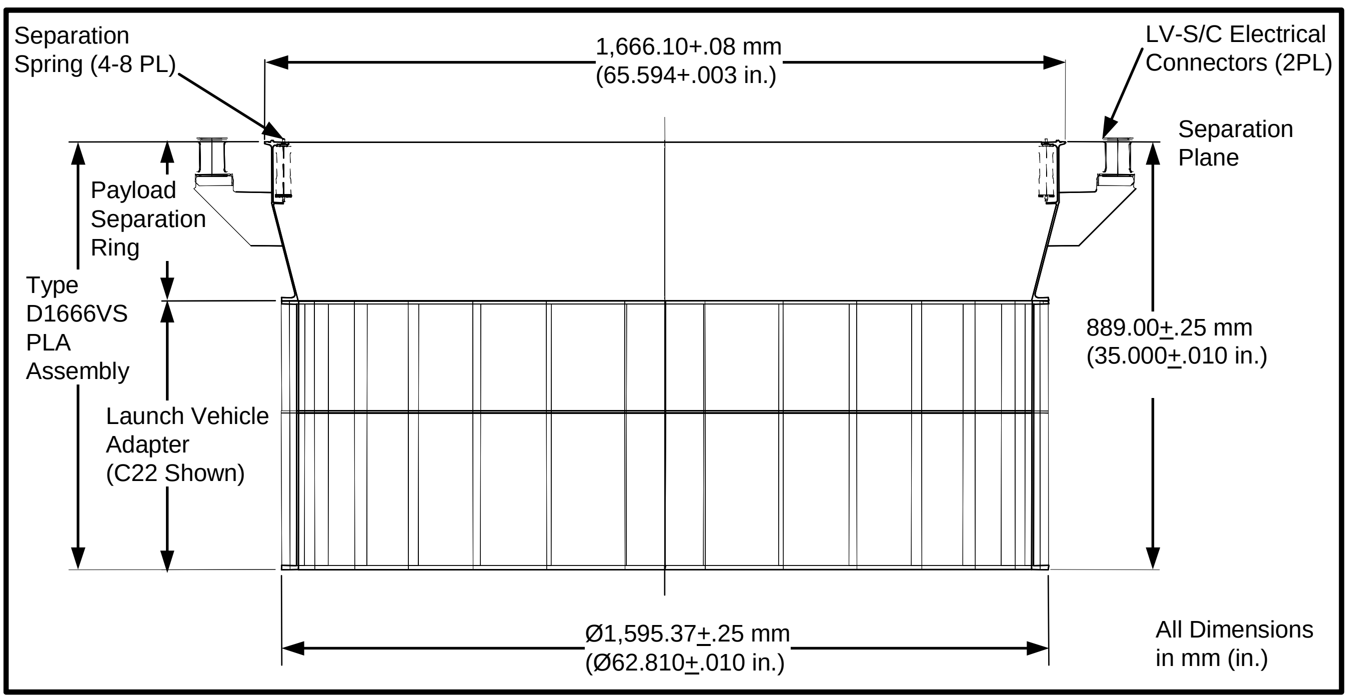


Figure E.8-1 Atlas Type D1666VS Payload Adapter

support equipment, including the torus arms and isolation diaphragm, used during ground processing operations.

Payload Separation System—The Atlas Type D1666VS payload adapter uses a launch vehicle-provided Marmon-type clampband payload separation system. This separation system (Fig. E.8-2) consists of a clampband set, release mechanism, and separation springs. The clampband set consists of a clampband for holding the spacecraft and adapter rings together plus devices to catch and retain the clampband on the adapter structure after separation. The clampband includes aluminum clamp segments that hold the payload adapter and spacecraft rings together and a single-piece aluminum retaining band that holds the clamp segments in place. The clamp segments also hold shear pins at 22 locations that fit into shear slots on the spacecraft aft ring and payload adapter forward ring to increase the structural capability of this system. The shear slot pattern on the spacecraft and payload adapter rings for this interface is controlled by matched tooling.

The ends of the retaining band are held together by the low-shock CBOD. The CBOD includes release bolts that engage the ends of the clampband. These release bolts are threaded into a flywheel mechanism. During installation and flight, the flywheel is restrained against rotation by a restraining pin. For separation, a pyrotechnically activated pin-puller retracts this pin from the flywheel, allowing it to rotate and eject the release bolts. This system significantly reduces shock compared to a conventional bolt-cutter system and is resetable, allowing actual flight hardware to be tested during component acceptance testing.

Separation spring assemblies provide the necessary separation energy after the clampband is released and are mounted to the payload adapter forward ring and bear on the spacecraft aft ring. Positive spacecraft separation is detected through continuity loops installed in the spacecraft electrical connector and wired to the upper-stage instrumentation for monitoring and telemetry verification.

Payload Adapter Structural Capabilities—Allowable spacecraft weights and longitudinal centers of gravity for the Type D1666VS payload adapter/separation systems are shown in Figure E.8-3. These spacecraft mass and center of gravity capabilities were determined using generic spacecraft interface ring geometry as shown in Figure E.8-4, and quasi-static load factors shown in Section 3.2.1. Actual

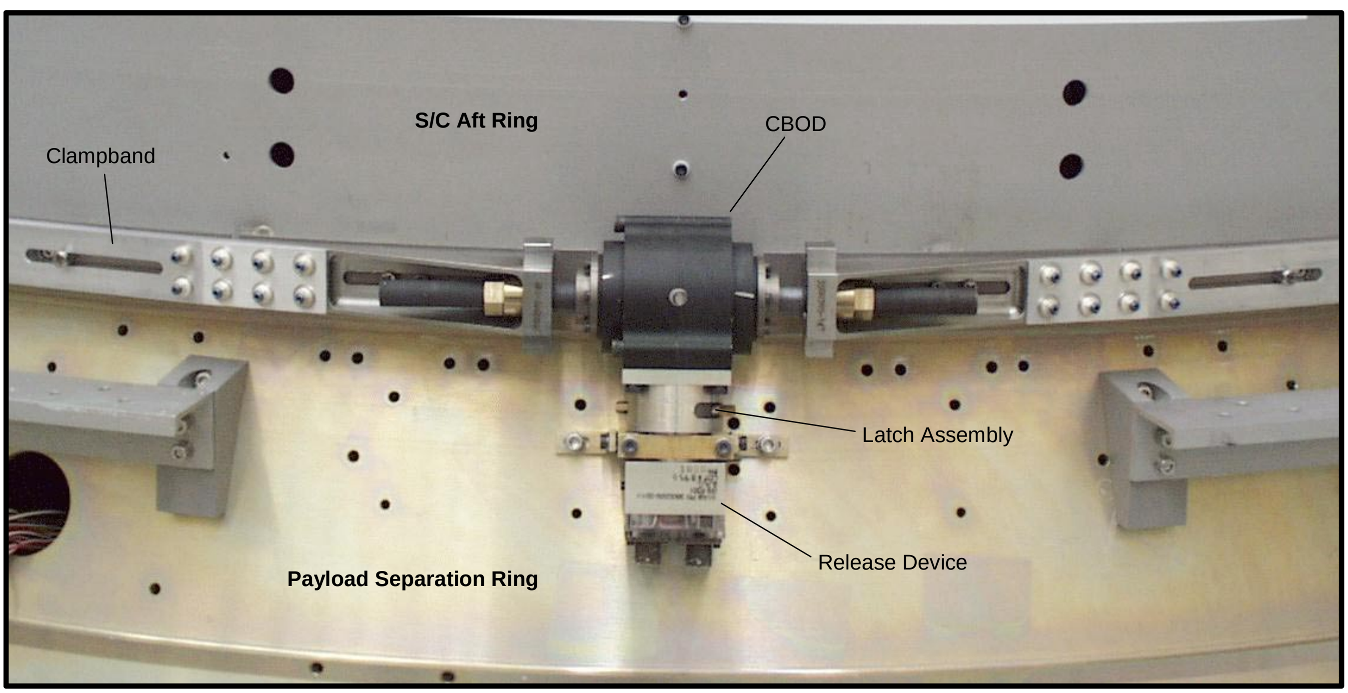


Figure E.8-2 PSS66VS Payload Separation System

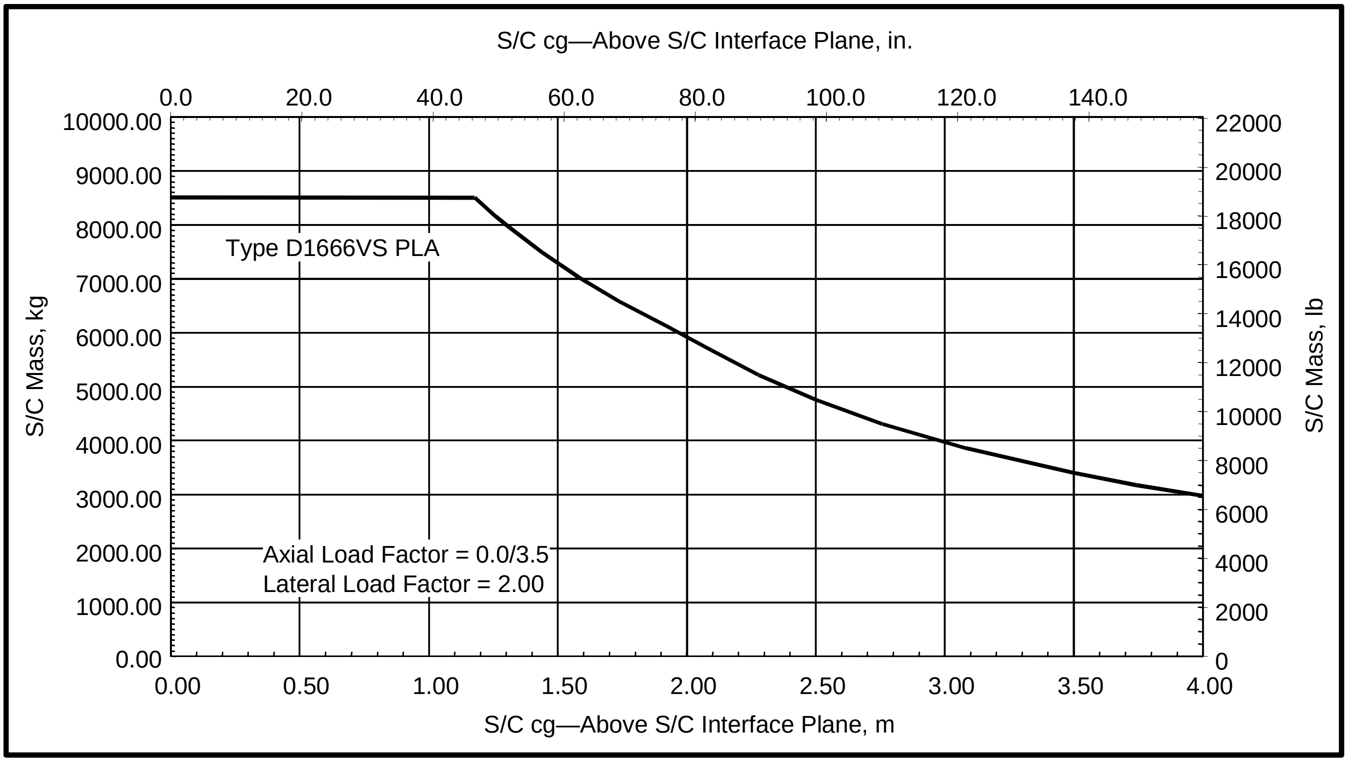


Figure E.8-3 Atlas Type D1666VS Payload Adapter Structural Capability

spacecraft design allowables may vary depending on interface ring stiffness and results of spacecraft mission-specific coupled loads analyses. Coordination with the Atlas program is required to define appropriate structural capabilities for spacecraft designs that exceed these generic allowables.

Payload Adapter Interfaces—The primary structural interface between the launch vehicle and spacecraft occurs at the payload adapter forward ring. This ring interfaces with the spacecraft aft ring and a payload separation system holds the two rings together for the structural joint and provides the release mechanism for spacecraft separation. Electrical bonding is provided across all interface planes associated with these components. Interface requirements for these components are shown in Figures E.8-4 and E.8-5. The payload adapter also provides mounting provisions for separation springs and supports interfacing components for electrical connectors between the launch vehicle and spacecraft. Interface requirements for these components are shown in Figures E.8-4 and E.8-5. Additional mission-specific provisions, including spacecraft purge provisions, spacecraft range safety destruct units, and mission satisfaction kit instrumentation, may be added as necessary.

Static Payload Envelope—The usable volume for the spacecraft relative to the payload adapter is defined by the static payload envelope. This envelope represents the maximum allowable spacecraft static dimensions (including manufacturing tolerances) relative to the spacecraft/payload adapter interface. This envelope is designed to allow access to the mating components and payload separation system for integration and installation operations, motion of the payload separation system during its operation, and movement of the spacecraft and launch vehicle after separation of the spacecraft and payload. Clearance layouts and separation analyses are performed for each spacecraft configuration, and if necessary, critical clearance locations are measured during spacecraft-to-payload-adapter mate operations to ensure positive clearance during flight and separation. Detailed views of the static payload envelope for the D1666VS payload adapter are shown in Figure E.8-6.

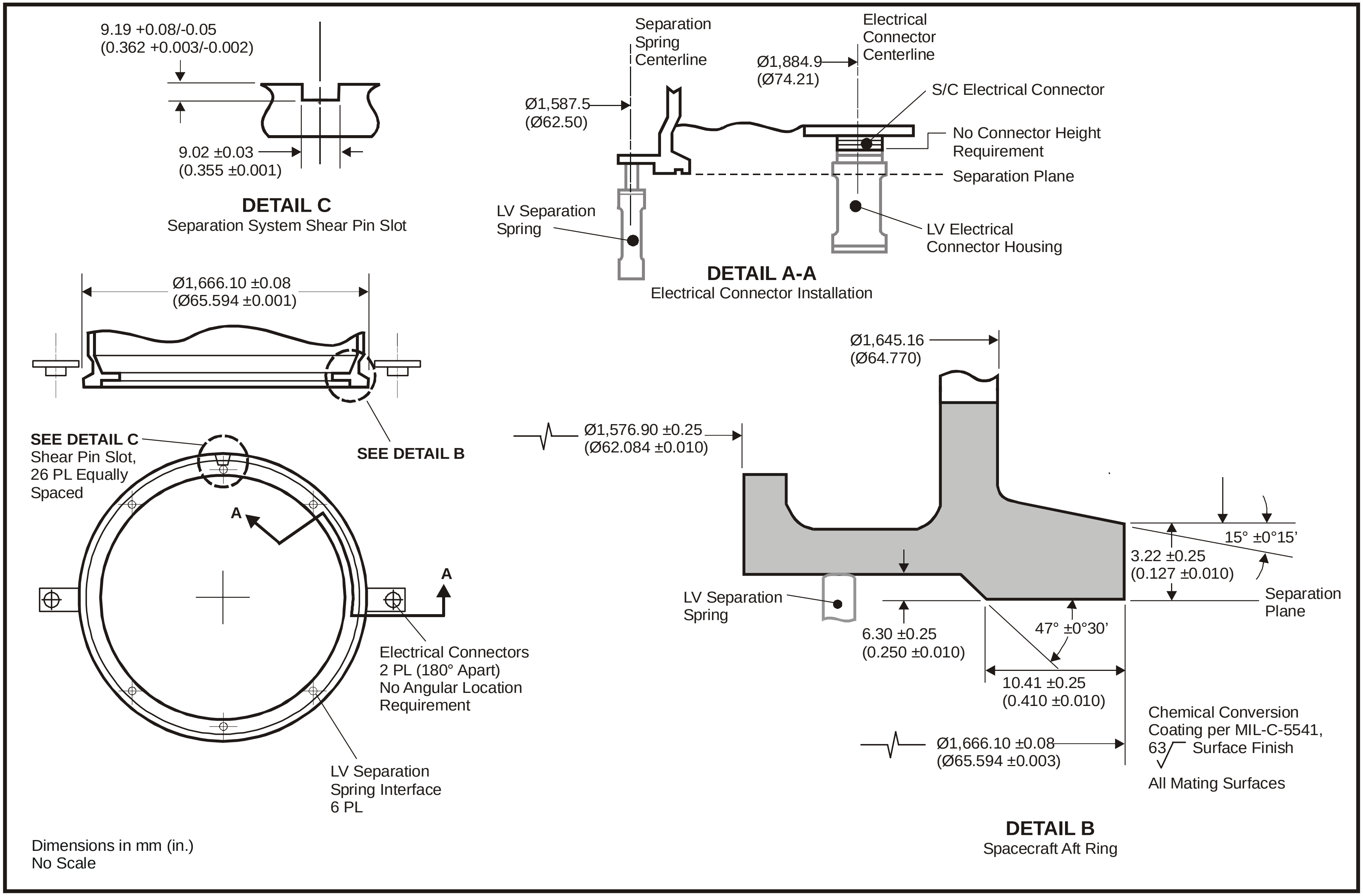


Figure E.8-4 Type D Adapter Spacecraft Interface Requirements

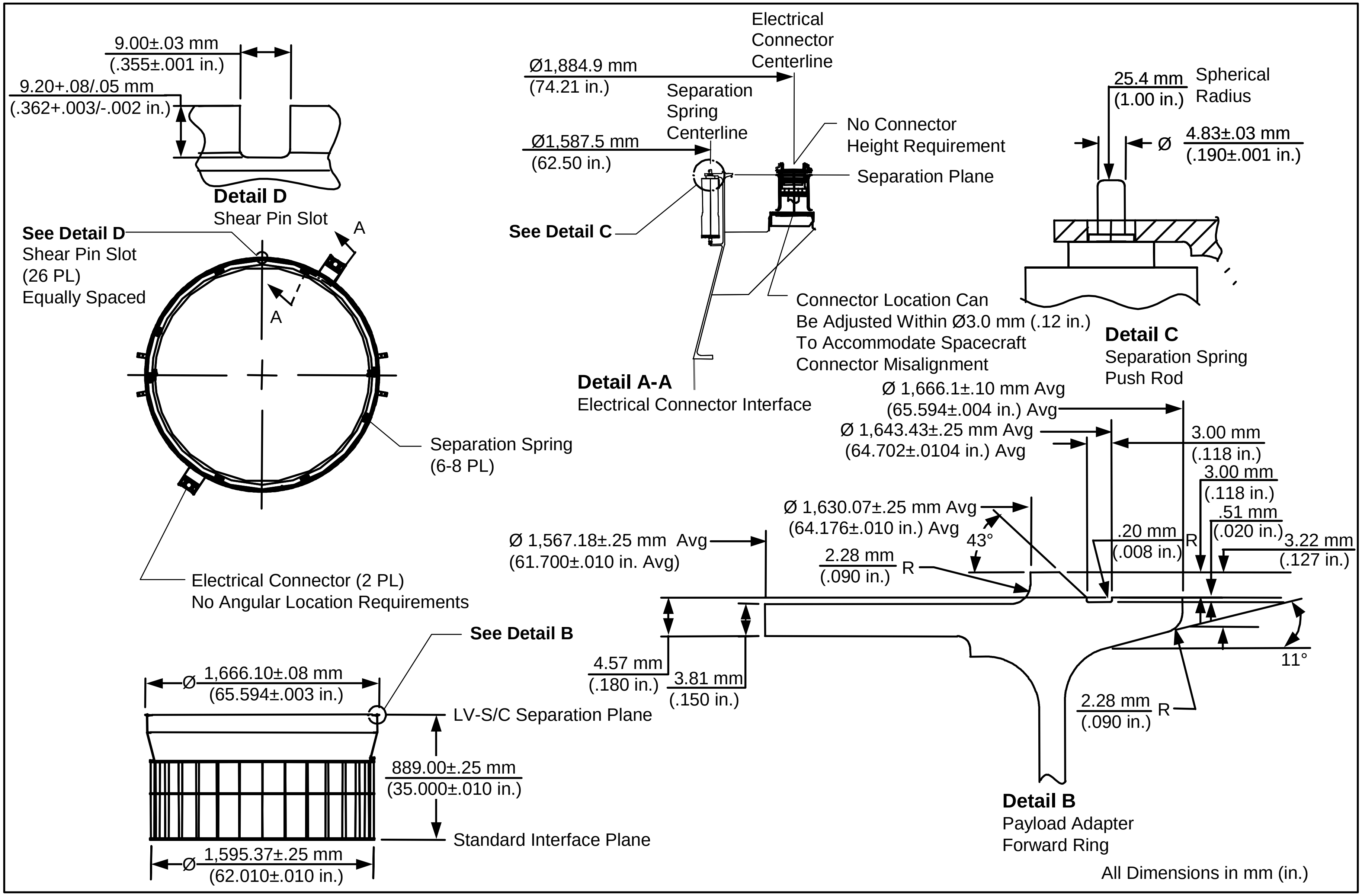


Figure E.8-5 Atlas Type D1666VS Payload Adapter Interface Requirements

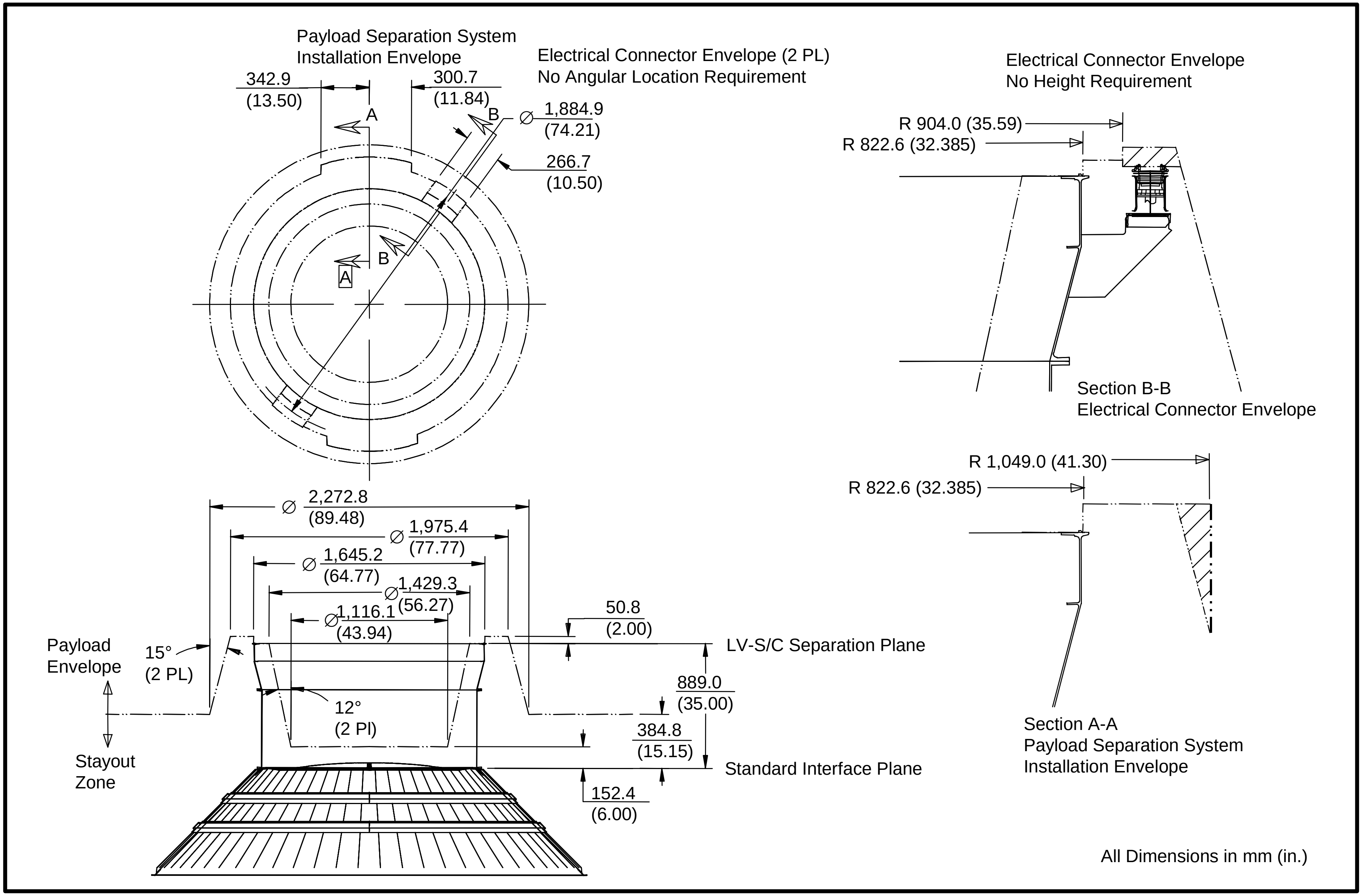


Figure E.8-6 Atlas Type D1666VS Payload Adapter Static Payload Envelope

E.9 ATLAS TYPE F1663 PAYLOAD ADAPTER

The Atlas Type F1663 adapter is being developed to support a payload using a four-hard point interface on a diameter of 1,663 mm (65.50 in.). Major characteristics of this payload adapter are summarized in Table E.9-1. This payload adapter consists of two major sections: the payload separation ring and the launch vehicle adapter (Fig. E.9-1). The payload separation ring is a machined aluminum component in the form of a 330.2-mm (13.00-in.) high truncated cone. The forward ring has four bearing surfaces that incorporate shear cones and bolt holes on an interface diameter of 1,663 mm (65.50 in.) that form the spacecraft separation plane. The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 120 evenly spaced holes that allow it to be joined to the launch vehicle adapter. This symmetrical hole pattern allows the payload separation ring and attached spacecraft to be rotated relative to the launch vehicle in 3° increments to meet mission-specific

Table E.9-1 Atlas Type F1663 Payload Adapter Characteristics

Atlas Type F1663 Payload Adapter		
Construction	Two-Piece, Integrally Machined Aluminum Construction	
Mass	91 kg	200 lb
Payload Capability	Figure E.9-3	
	6,700 kg at 1.75 m	14,800 lb at 69.3 in.
P/L Separation Sys	FASSN (4 Places)	
Max Shock Levels	Available on Request	
Separation Bolt Preload	222 kN	50,000 lb
Separation Springs		
Number	4	
Force per Spring—Max	1 kN	225 lb

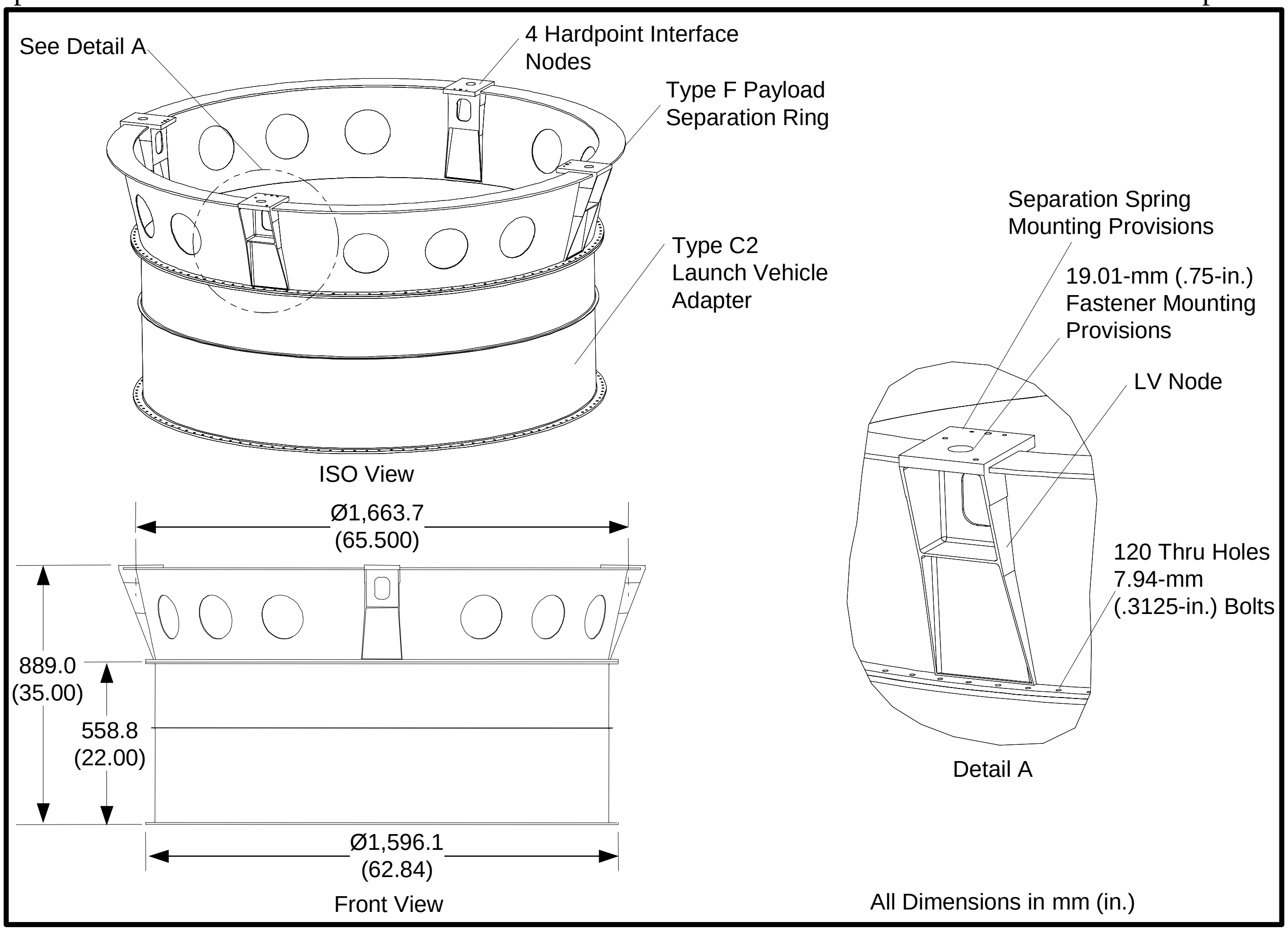


Figure E.9-1 Atlas Type F1663 Payload Adapter

requirements. The payload separation ring supports all hardware that directly interfaces with spacecraft, including the payload separation system, electrical connectors, and mission-specific options.

The launch vehicle adapter is a machined aluminum component in the form of an integrally stiffened cylinder. The forward ring has an outer diameter of 1,595 mm (62.81 in.) and contains 120 holes spaced evenly every 3° that allow it to be joined to the payload separation ring. The aft ring has an outer diameter of 1,595 mm (62.81 in.) and contains 121 holes that match up with Atlas standard interface plane requirements. The nominal height of the launch vehicle adapter is 558.8 mm (22.00 in.), but this height may be varied from 330.2 mm (13.00 in.) to 736.6 mm (29.00 in.) to meet mission-specific requirements. The launch vehicle adapter includes all provisions for mating to the launch vehicle ground support equipment, including the torus arms and isolation diaphragm, used during ground processing operations.

The Type F1663 payload adapter is in development. Information on the payload adapter and spacecraft interface requirements is available upon request to the Atlas program.

Payload Separation System—The Atlas Type F1663 payload adapter uses a launch vehicle-provided FASSN system that attaches the spacecraft to the forward ring of the payload adapter, and a separation spring set that provides the necessary separation energy after the separation nut is actuated (Fig. E.9-2). The FASSN set consists of a bolt, actuator, and retractor. The bolt forms the mechanical connection between the spacecraft and payload adapter. The actuator includes a flywheel mechanism that the bolt is threaded into. During installation and flight, the flywheel is restrained against rotation. For separation, an electromechanically activated device releases the flywheel allowing it to rotate and eject the bolts. This system significantly reduces shock compared to a conventional separation nut system and is resetable, allowing actual flight hardware to be tested during component acceptance testing. The retractor captures the bolt on the spacecraft structure after separation. Coordinated tooling between the spacecraft and payload adapter is required for this system.

Separation spring assemblies provide the necessary separation energy after the clampband is released and are mounted to the payload adapter forward ring and bear on the spacecraft aft ring. Positive spacecraft separation is detected through continuity loops installed in the spacecraft electrical connector and wired to the upper-stage instrumentation for monitoring and telemetry verification.

Payload Adapter Structural Capabilities—Allowable spacecraft weights and longitudinal centers of gravity for the Type F1663 payload adapter/separation systems are shown in Figure E.9-3. These spacecraft mass and center of gravity capabilities were determined using quasi-static load factors shown in Section 3.2.1. Actual spacecraft design allowables may vary depending on interface ring stiffness and results of spacecraft mission-specific coupled loads analyses. Coordination with the Atlas program is required to define appropriate structural capabilities for spacecraft designs that exceed these generic allowables.

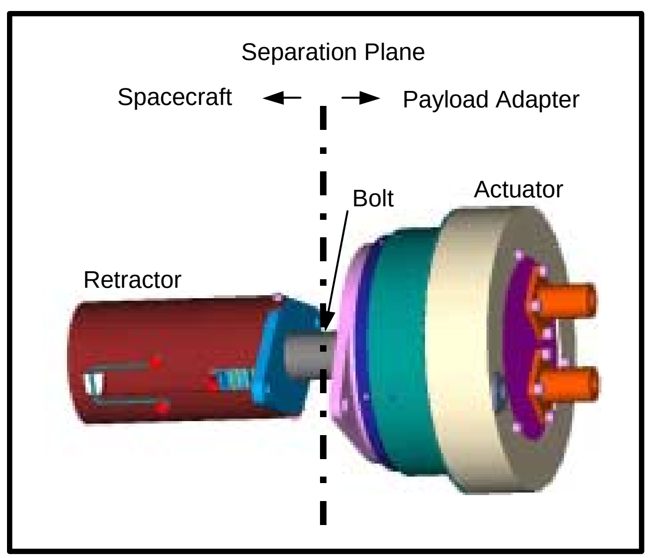


Figure E.9-2 Fast Acting Shockless Separation Nut (FASSN) Payload Separation System

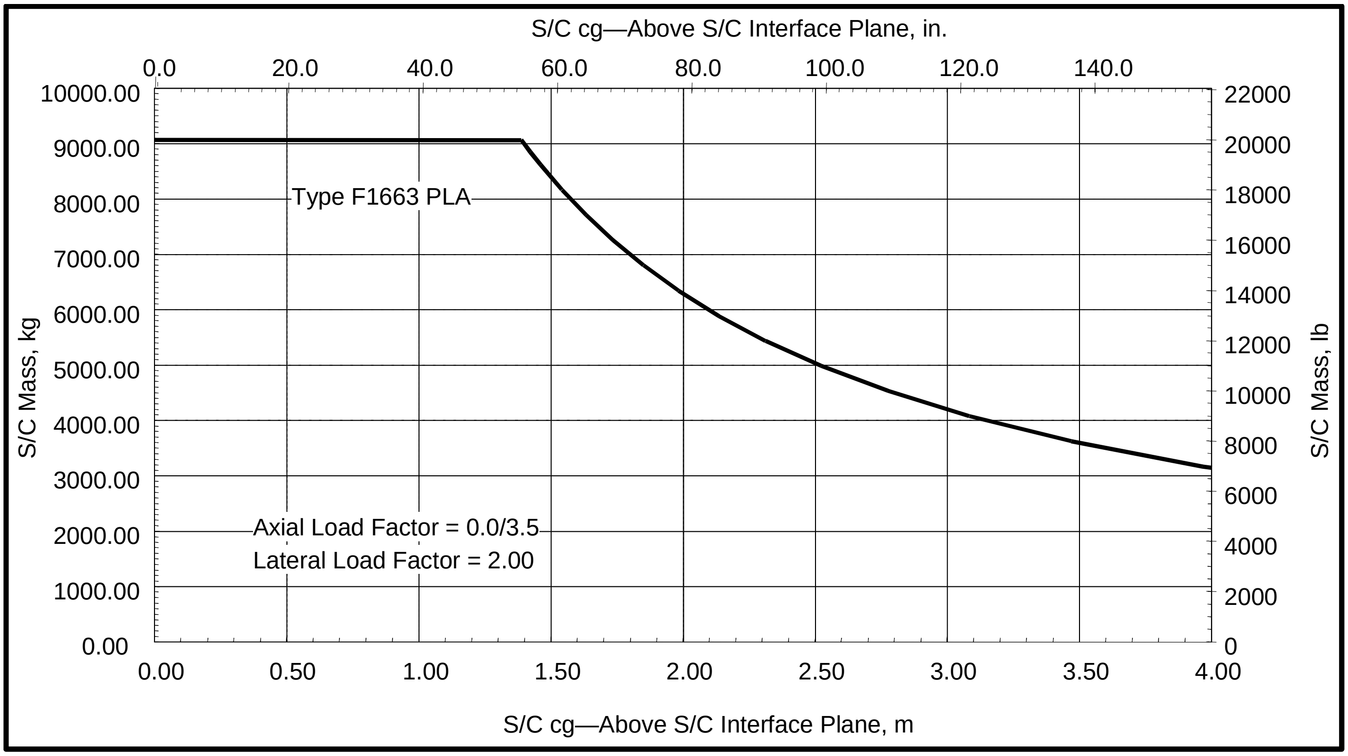


Figure E.9-3 Atlas Type F1663 Payload Adapter Structural Capability

GLOSSARY

3-D	Three Dimensional
A	ampere(s)
Å	angstrom(s)
ac	Alternating Current
A/C	Air Conditioning
ADDJUST	Automatic Determination and Dissemination of Just-Updated Steering Terms
ADJ	Attach, Disconnect, and Jettison
AFB	Air Force Base
AFM	Air Force Manual
AFQA	Air Force Quality Assurance
AFR	Air Force Regulation
AFSCF	Air Force Satellite Control Facility
AFSCN	Air Force Space Control Network
AGE	Aerospace Ground Equipment
AHU	Air Handling Unit
AMPG	*Atlas Mission Planner's Guide*
AMSC	American Mobile Satellite Corporation
ARCM	Atlas Roll-Control Module
A-RCU	Atlas Remote Control Unit
AS	Atlas Station
ASO	Astrotech Space Operations
ASOC	Atlas V Spaceflight Operations Center
ATP	Authority To Proceed
ATS	Advanced Technology Satellite
ATS	Aft Transition Skirt
AVE	Aerospace Vehicle Equipment
AWG	American Wire Gauge
Batt	Battery
BECO	Booster Engine Cutoff
BETC	Booster Engine Thrust Coupling
BOS	Booster on Stand
BPJ	Booster Package Jettison
BPSK	Binary Phase-Shift Keying
BRCU	Booster Remote Control Unit
BSI	British Standards Institute
Bstr Sep	Booster Separation
Btu	British thermal unit(s)
°C	Degree(s) Celsius
CA	Corrective Action
CAD	Computer-Aided Design
CBOD	Clampband Opening Device

CCAFS	Cape Canaveral Air Force Station
CCAM	Collision and Contamination Avoidance Maneuver
CCAPS	Computer-Controlled Atlas Pressurization System
CCB	Common Core Booster™
CCLS	Computer-Controlled Launch Set
CCTV	Closed-Circuit Television
CCVAPS	Computer-Controlled Vent and Pressurization System
CDR	Critical Design Review
CEM	Common Equipment Module
CERT	Combined Electrical Readiness Test
CFLR	Centaur Forward Load Reactor
cfm	cubic feet per minute
cg	center of gravity(ies)
CIB	Change Integration Board
C-ISA	Centaur Interstage Adapter
CLA	Coupled Loads Analysis
CLE	Centaur Longitudinal Event
CLS	Commercial Launch Services
cm	centimeter(s)
Cmd	Command
C/O	Checkout
CPM	Common Payload Module
C-RCU	Centaur Remote Control Unit
CRES	Corrosion-Resistant Steel
CRRES	Combined Release and Radiation Effects Satellite
CS	Centaur Station
CSC	Customer Support Center
CSF	Customer Support Facility
CSO	Complex Safety Officer
CT	Command Transmitter
CW	Continuous Wave
CX	Complex
DAS	Data Acquisition System
dB	decibel(s)
DBS	Direct Broadcast Satellite
dc	Direct Current
DCMA	Defense Contract Management Agency
DEC	Dual-Engine Centaur
Dia	Diameter
DLF	Design Load Factor
DMSP	Defense Meteorological Satellite Program
DOF	Degree of Freedom
DOP	Dioctyl Phthalate
DPC	Dual Payload Carrier

DPF	Defense System Communications Satellite (DSCS) Processing Facility
DPM	Director Program Management
DSCS	Defense System Communications Satellite
Dsn	Design
DSN	Deep Space Network
DUF	Dynamic Uncertainty Factor
EB	East Bay
ECA	Environmentally Controlled Area
ECS	Environmentally Controlled System
ECU	Electronic Control Unit
EED	Electroexplosive Device
EELV	Evolved Expendable Launch Vehicle
EGSE	Electrical Ground Support Equipment
EHA	Electrical Hydraulic Actuator
EHF	Extreme High Frequency
EIA	Electronics Industry Association
EICD	Electrical Interface Control Drawing
EM	Electromagnetic
EMA	Electromechanical Actuator
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EMI/EMC	Electromagnetic Interference/Electromagnetic Compatibility
EMK	Extended Mission Kit
Eng	Engine
EOC	Engineering Operations Center
EPA	Environmental Protective Agency
EPF	Extended Payload Fairing
ER	Eastern Range
ERB	Engineering Review Board
ERR	Eastern Range Regulation
ESMCR	Eastern Space and Missiles Control Regulation
EUTELSAT	European Telecommunications Satellite
EVCF	Eastern Vehicle Checkout Facility
EWR	Eastern/Western Range
°F	Degree(s) Fahrenheit
FAA	Federal Aviation Authority
FAB	Final Assembly Building
FAP	Fairing Acoustic Protection
FASSN	Fast-Acting Shockless Separation Nut
FC	Facility Console
FCDC	Flexible Confined Detonating Cord
FCS	Flight Control Subsystem
FDLC	Final Design Loads Cycle

Fig.	Figure
FLR	Forward Load Reactor
Flt Contl	Flight Control
FLTSATCOM	Fleet Satellite Communications
FM	Flight Model
FMH	Free Molecular Heating
F/O	Follow-on
FO	Fiber Optic
FPA	Flight Plan Approval
FPR	Flight Performance Reserve
FS	Factor of Safety
FSO	Flight Safety Officer
ft	foot (feet)
ft^2	square feet
FTS	Flight Termination System
g	gravity
G&C	Guidance and Control
GC^3	Ground, Command, Control, and Communication
GG	Gas Generator
GHe	Gaseous Helium
GIDEP	Government-Industry Data Exchange Program
G&N	Guidance and Navigation
GN_2	Gaseous Nitrogen
Gnd	Ground
GOES	Geostationary Operational Environmental Satellite
GORR	Ground Operations Readiness Review
GOWG	Ground Operations Working Group
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
GSO	Geosynchronous Orbit
GSTDN	Ground Spaceflight Tracking and Data Network
GTO	Geosynchronous Transfer Orbit
GTR	Gantry Test Rack
GTS	Ground Telemetry Station
GTV	Ground Transport Vehicle
GVSC	Generic Very-High-Speed Integrated Circuit
Haz Gas	Hazardous Gas
He	Helium
HEAO	High Energy Astronomy Observatory
HEPA	High-Efficiency Particulate Air
HGDS	Hazardous Gas Detection System
HLV	Heavy Lift Vehicle
hp	horsepower

HPF	Hazardous Processing Facility
hr	hour(s)
Hz	hertz
ICBM	Intercontinental Ballistic Missile
ICD	Interface Control Document
I/F	Interface
IFD	In-Flight Disconnect
ILC	Initial Launch Capability
ILS	International Launch Services
IMS	Inertial Measurement System
in.	inch(es)
INTELSAT	International Telecommunications Satellite
INU	Inertial Navigation Unit
IRD	Integrated Requirements Document
IRD	Interface Requirements Document
ISA	Interstage Adapter
ISDS	Inadvertent Separation and Destruct System
I_{SP}	Specific Impulse
IST	Integrated System Test
ITA	Integrated Thermal Analysis
ITO	Intermediate Transfer Orbit
ITP	Integrated Test Plan
IVA	Integrated Valve Assembly
JCSAT	Japan Communications Satellite
JPL	Jet Propulsion Laboratory
kg	kilogram(s)
klb	kilo-pound(s)
km	kilometer(s)
kN	kilonewton(s)
KPa	kilopascal(s)
KSC	Kennedy Space Center
kVA	kilovolt-ampere(s)
kV/m	kilovolt(s) per meter
LAN	Local Area Network
lb	pound(s)
LC	Launch Complex
LCC	Launch Control Center
LD	Launch Director
LDA	Launch Site Authority
LEO	Low-Earth Orbit
LH_2	Liquid Hydrogen

LKEI	Khrunichev-Energia International Incorporated
LM	Lockheed Martin
LMAO	Lockheed Martin Astronautics Operations
LMCLS	Lockheed Martin Commercial Launch Services
LMD	Lockheed Martin Mission Director
LN_2	Liquid Nitrogen
LO_2	Liquid Oxygen
LOB	Launch Operations Building
LOC	Launch Operations Center
LPF	Large Payload Fairing
LR	Load Ratio
LRR	Launch Readiness Review
LSB	Launch and Service Building
LSC	Linear-Shaped Charge
LV	Launch Vehicle
LVCE	Launch Vehicle Chief Engineer
LVDC	Launch Vehicle Data Center
LVRT	Launch Vehicle Readiness Test
m	meter(s)
m^2	square meter(s)
M	Million(s)
MAPS	Mission Air Purge System
MDC	Mission Director's Center
MDU	Master Data Unit
MECO	Main Engine Cutoff
MES	Main Engine Start
MHz	MegaHertz
MICD	Mechanical Interface Control Drawing
MILA	Merritt Island Launch Area
MLP	Mobile Launch Platform
MLV	Medium Launch Vehicle
MOC	Mission Operations Center
MP	Mission Peculiar
MPDR	Mission-Peculiar Design Review
MR	Mixture Ratio
MRB	Material Review Board
MS&PA	Mission Success and Product Assurance
MSPSP	Missile System Prelaunch Safety Package
MSR	Mission Support Room
MST	Mobile Service Tower
N	Newton(s)
NASA	National Aeronautics and Space Administration
N_2H_4	Hydrazine

NIOSH	National Institute for Occupational Safety and Health
nmi	nautical mile(s)
NOAA	National Oceanic and Atmospheric Administration
NRZ-L	Nonreturn-to-Zero Level
NVR	Nonvolatile Residue
OAO	Orbiting Astronomical Observatory
OASPL	Overall Sound Pressure Level
OCC	Operations Communication Center
OD	Outer Diameter
OH	Overhead
Ops	Operations
ORCA	Ordnance Remote Control Assembly
ORD	Operations Requirements Document
OSHA	Occupational Health and Safety Administration
PA	Public Address
PAFB	Patrick Air Force Base
PB	Preburner
PCM	Pulse-Code Modulation
PCOS	Power Changeover Switch
PDLC	Preliminary Design Loads Cycle
PDR	Preliminary Design Review
PDSM	*Product Delivery System Manual*
PEB	Payload Equipment Building
Perf	Performance
PFJ	Payload Fairing Jettison
PFM	Protoflight Model
PHSF	Payload Hazardous Servicing Facility
PIU	Pyroinhibit Unit
Pk	Peak
P/L	Payload
PLA	Payload Adapter
PLCP	Propellant Leak Contingency Plan
PLCU	Propellant Loading Control Unit
PLF	Payload Fairing
PLIS	Propellant-Level Indicating System
PLRR	President's Launch Readiness Review
PMP	Parts, Materials, and Processes
PMPCM	Parts, Materials, and Processes Control Board
PMR	Preliminary Material Review
POD	Program Office Directive
POL	Paints, Oils, and Lubricants
PPF	Payload Processing Facility
PRD	Program Requirements Document

psi	pound(s) per square inch
psig	pound(s) per square inch, gage
PSS	Payload Separation System
PST	Product Support Team
PSW	Payload Systems Weight
PTC	Payload Test Conductor
PU	Propellant Utilization
PVA	Perigee Velocity Augmentation
PVan	Payload Van
PVC	Polyvinyl Chloride
P&W	Pratt & Whitney
Pwr	Power
PYC	Pyrotechnic Control
R	Recorder
RAAN	Right Ascension of Ascending Node
RC	Range Coordinator
RCS	Reaction Control System
RCU	Remote Control Unit
R&D	Research and Development
RDU	Remote Data Unit
RF	Radio Frequency
RFTS	Radio Frequency Test Set
RGU	Rate Gyro Unit
RH	Relative Humidity
RLCC	Remote Launch Control Center
Rm	Room
RNCO	Range Noncommissioned Officer
ROCC	Range Operations Control Center
RP	Rocket Propellant
RPO	Radiation Protection Officer
RSC	Range Safety Console
RSSR	Range Safety System Report
RTS	Remote Tracking Stations
s	Second(s)
S/A	Safe and Arm
SAEF	Spacecraft Assembly and Encapsulation Facility
SAI	Safe/Arm Initiator
SAR	Safety Assessment Report
SAR	Software Anomaly Report
SASA	Safe/Arm and Securing Unit
S/C	Spacecraft
SCAPE	Self-Contained Atmospheric-Protective Ensemble
SCC	Spacecraft Control Center

SCTC	Spacecraft Test Conductor
SEC	Single-Engine Centaur
SECO	Sustainer Engine Cutoff
SERB	Systems Engineering Review Board
SFC	Spacecraft Facility Controller
SIL	Systems Integration Laboratory
SIP	Standard Interface Plan
SIU	Servo Inverter Unit
SL	Sea Level
SLC	Space Launch Complex
SLC	Spacecraft Launch Conductor
SLV	Space Launch Vehicle
SOC	Spacecraft Operations Center
SOHO	Solar and Heliospheric Observatory
SOW	Statement of Work
SPF	Spacecraft Processing Facility
SPL	Sound Pressure Level
SPRB	Space Program Reliability Board
SRB	Solid Rocket Booster
SRM	Solid Rocket Motor
Sta	Station
STC	Satellite Test Center
STE	Special Test Equipment
STM	Structural Test Model
STV	Spacecraft Transport Vehicle
T	Table
TC	Test Conductor
TCO	Thrust Cutoff
TDRS	Tracking and Data Relay Satellite
TDRSS	Tracking and Data Relay Satellite System
TEMP	Test and Evaluation Master Plan
Tlm	Telemetry
TMRSS	Triple Modular Redundant
TOPS	Transistorized Operational Phone System
TRAJEX	Trajectory Simulation Program
TVC	Thrust Vector Control
TVCF	Transportable Vehicle Checkout Facility
UHF	Ultra-High Frequency
Umb	Umbilical
UPS	Uninterruptible Power System
USAF	United States Air Force
USG	United States Government
USSN	Universal Spacecraft Separation Node

UT	Umbilical Tower
V	volt(s)
VAB	Vehicle Assembly Building (Kennedy Space Center)
Vac	volt(s), alternating current
VAFB	Vandenberg Air Force Base
Vdc	volt(s) direct current
VGP	Virtual Ground Plane
VIF	Vertical Integration Facility
V/m	volt(s) per meter
VTF	Vertical Test Facility
W	watt(s)
WB	West Bay
WDR	Wet Dress Rehearsal
WSMCR	Western Space and Missiles Control Regulation
Xdcr	Transducer
$\omega\rho$	Argument of Perigee